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# COLLECTED WORKS

OF

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# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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## TECHNICAL NOTE NO. 711

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### AN APPROXIMATE SPIN DESIGN CRITERION FOR MONOPLANES

By Oscar Seidman and Charles J. Donlan

#### SUMMARY

A quantitative criterion of merit has been needed to assist airplane designers to incorporate satisfactory spinning characteristics into new designs. An approximate empirical criterion, based on the projected side area and the mass distribution of the airplane, has been formulated in a recent British report. In the present paper, the British results have been analyzed and applied to American designs. A simpler design criterion, based solely on the type and the dimensions of the tail, has been developed; it is useful in a rapid estimation of whether a new design is likely to comply with the minimum requirements for safety in spinning.

#### INTRODUCTION

A considerable amount of information concerning the effects of dimensional and inertial design characteristics exists in the literature on spinning. In general, however, the data are so presented that they are not directly and quantitatively applicable to new designs. There is need for a satisfactory quantitative criterion to indicate whether a new design is likely to comply with the minimum requirements for safety in spinning.

Such a criterion is developed in a recent British publication (reference 1). The present report is concerned with the application of the British criterion to American airplanes. An analysis of the results is presented and a simplified criterion of spinning merit developed that, as far as American designs are concerned, conforms better with full-scale and model spinning data than the original English criterion.



## BASIS OF ENGLISH CRITERION

The complete development of the British criterion is given in detail in reference 1. The basic considerations underlying the development are reviewed briefly in the following paragraphs.

The spinning characteristics of an airplane are considered to be affected by three major design factors. These factors are:

- (1) The longitudinal distribution of mass as measured by the difference  $I_Z - I_X$  and expressed nondimensionally as  $\frac{I_Z - I_X}{\rho S (b/2)^3}$ , where  $I_Z$  and  $I_X$  are the moments of inertia about the Z and X body axes, respectively;  $\rho$  is the density of the air;  $S$ , the wing area; and  $b/2$ , the semispan. The value of air density,  $\rho$ , used in this report is that corresponding to 15,000 feet standard altitude.
- (2) The resistance offered by the fuselage side area (exclusive of the rudder) while the airplane is spinning, which is measured by  $\Sigma Ax^2$ , where  $A$  is an elementary area located at a distance  $x$  from the center of gravity of the airplane. Because of its greater effectiveness, the area beneath the horizontal tail plane is multiplied by 2. (For conventional tail planes, this area is measured between the most forward and the most rearward portions of the tail plane.) The resistance of the fuselage to rotation is expressed in the form of a nondimensional "body damping ratio," defined as  $\frac{\Sigma Ax^2}{S (b/2)^3}$ , where  $S$  and  $b/2$  denote the wing area and the semispan, respectively.
- (3) The unshielded rudder area, expressed nondimensionally as an "unshielded rudder volume coefficient" is equal to  $\frac{\text{unshielded rudder area} \times l}{S (b/2)}$ , where  $l$  is the distance from the centroid of the unshielded rudder area to the center of gravity of the airplane.

In the computation of the body damping ratio (BDR) and the unshielded rudder volume coefficient (URVC), it was assumed that the relative wind strikes the horizontal tail surfaces from below at an angle of  $45^\circ$  and that the air flow diverges  $\pm 15^\circ$  after passing the tail plane. (This assumption regarding the divergence of the air flow above the tail plane is verified by the flight tests described in reference 2.) Any area of the vertical tail within this divergent wake is disregarded in the computations. Figure 1 illustrates the method used in evaluating BDR and URVC.

A "damping-power factor" (DPF) is defined as the product  $\text{BDR} \times \text{URVC}$ , and this factor is plotted against the pitching parameter  $\frac{I_Z - I_X}{\rho S(b/2)^3}$ . The relative magnitude of the slope,  $\frac{(\text{DPF}) [\rho S(b/2)^3]}{I_Z - I_X}$ , is used by the British as a figure of merit.

#### COMPARISON OF BRITISH AND AMERICAN RESULTS

Figure 2, which is taken from reference 1, is a plot of DPF against  $\frac{I_Z - I_X}{\rho S(b/2)^3}$  for the 22 British monoplane designs submitted for testing in the British free-spinning wind tunnel. The models are rated as either "passed" or "failed," depending on their ability to meet the requirements of a standard British model recovery test. In most instances where an initial design is represented as unsatisfactory, a point will be found representing the final modified version of that design. It is obvious from the dispersion of points that secondary factors not included in the analysis influence the ability of a model to pass the spin test. Nevertheless, a line has been drawn such that no pass point lies below it (although failures may lie above it) and defines the minimum requirement for safety in spinning. It is implied that any design the characteristic point of which lies beneath this line (i.e., for which the ratio  $\frac{(\text{DPF}) [\rho S(b/2)^3]}{I_Z - I_X}$  is less than 0.001) will probably give unsatisfactory recoveries from a spin;

whereas, if its characteristic point lies above this line, recoveries may be either satisfactory or unsatisfactory.

Figure 3 is a plot of DPF against  $\frac{I_z - I_x}{\rho S(b/2)^3}$  for 14 American monoplanes tested in the N.A.C.A. free-spinning wind tunnel. The N.A.C.A. having no unique criterion of spinning merit, the designs considered here have been denoted as being "good" or "poor" spinners, partly on the basis of spin-tunnel results and partly on the basis of pilots' reports. The fact that the English standard of recovery was not utilized in the American classification may account for the relatively greater percentage of American airplanes appearing as satisfactory spinners. It will be noted that two good points fall below the British line for minimum safety in spinning. If, however, a line (dotted line on fig. 3) is drawn through the point for airplane 1 a separation of the American airplanes is effected; this line has about one-half the slope of the British line.

#### ANALYSIS OF RESULTS

A detailed analysis of both the British and the American results was then made with the purpose of obtaining a simpler and more effective criterion.

The individual factors that constitute the British DPF are plotted in figures 4 and 5. The close grouping of points in figure 4 discourages the establishment of a spin criterion on the basis of the BDR alone. Figure 5, on the other hand, shows a greater dispersion of points and the dotted horizontal line drawn through a value of URVC of 0.013 effects a separation of passed and failed points that is comparable with the separation previously noted on the DPF chart (fig. 2). This result suggests that the URVC alone might prove as satisfactory a criterion as the more complex DPF, which necessitates the consideration of BDR and  $\frac{I_z - I_x}{\rho S(b/2)^3}$ .

It would appear that an alternative conclusion to the British report might have stated that any model possessing a value for URVC of less than 0.013 would be unlikely to pass the model spinning requirements. This condition would have eliminated the necessity of considering the body damping ratio and the inertia pitching parameter.

This discussion concludes the analysis of the British data. The rest of the report is concerned with the analysis of the results for the American monoplanes and the formulation of a criterion based on the unshielded rudder area and the fuselage area directly below the horizontal tail surfaces.

Figures 6 and 7 are plots of the factors constituting the DRF for the American monoplanes. Figure 6 could be used to segregate the good from the poor spinners but, as will be shown later, the segregation is largely attributable to the area beneath the horizontal tail surfaces and not to the BDR as a whole. In figure 7, the dotted horizontal line drawn through the value of URVC of 0.01 indicates that satisfactory separation of good and poor spinners can be obtained for the American airplanes by considering the URVC alone, although the line of separation is lower than that used for the British models in figure 5. Thus the value  $URVC = 0.01$  might be used to separate new designs into two classifications; designs having a value of URVC less than 0.01 may be considered unlikely to pass the spinning requirements.

In reference 1, the importance of the fixed area below the horizontal tail surfaces has been recognized by arbitrarily doubling its contribution to the body damping ratio. Its influence is obscured, however, because its contribution may be small even when doubled as compared with the body damping ratio. In order further to emphasize its importance, the contribution to the body damping ratio of the fixed area below the horizontal tail surfaces has been considered separately for the American airplanes;

it is expressed as a tail damping ratio, 
$$TDR = \frac{FL^2}{S(b/2)^2}$$
 where  $F$  is the total fixed area below the horizontal tail and  $L$  is the distance from the centroid of this area to the center of gravity of the airplane.

Values of the TDR for American monoplanes are shown in figure 8. It will be noted that a separation of the good from the poor spinners can be effected by using the value  $TDR = 0.015$ . Figures 7 and 8 show that the URVC and the TDR taken separately effect similar separations of the American designs into two groups. It is obvious that many possible combinations of these two factors could be used in devising an empirical criterion to segregate the poor spinners. In order to emphasize the importance of

having both unshielded rudder area and fixed area below the horizontal surfaces, it was decided to use the product  $URVC \times TDR$  as a criterion of merit. It is appreciated that the results may not be valid for unconventional designs.

Figure 9 illustrates the method used in evaluating a tail damping-power factor (TDPF) defined as the product of TDR and URVC. This TDPF is plotted in figure 10 for 14 American monoplanes and effects a satisfactory separation of good and poor spinners.

It may be concluded on the basis of figure 10 that monoplanes possessing a TDPF of less than, say, 0.00015 are not likely to exhibit satisfactory recovery characteristics. On the other hand, it is felt that a TDPF in excess of 0.00015 is, in itself, insufficient to insure satisfactory spin characteristics.

Similar results were obtained for American biplanes, but the critical value of the TDPF appeared to be somewhat lower and less distinct than for the monoplanes.

Lack of sufficient data prevented the calculation of the TDPF for the British designs.

#### DISCUSSION AND CONCLUSIONS

In the present state of knowledge, no criterion is available that will infallibly predict the recovery characteristics of a new airplane design. As shown in the text, however, it is possible to formulate empirical criteria that are helpful in establishing the minimum design requirements for safety in spinning. It is believed that the tail damping-power factor (TDPF) developed in the text is a simple practical method for rapidly estimating whether a new design is likely to comply with the minimum requirements for safety in spinning and it is recommended that no new monoplane design be constructed which possesses a TDPF of less than 0.00015. It should not be assumed, however, that a design which has a satisfactory TDPF will necessarily exhibit good recovery characteristics, as other factors not herein considered may influence the results.

Langley Memorial Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., May 1, 1939.

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1. Finn, E.: Analysis of Routine Tests of Monoplanes in the Royal Aircraft Establishment Free Spinning Tunnel. R. & M. No. 1810, British A.R.C., 1937.
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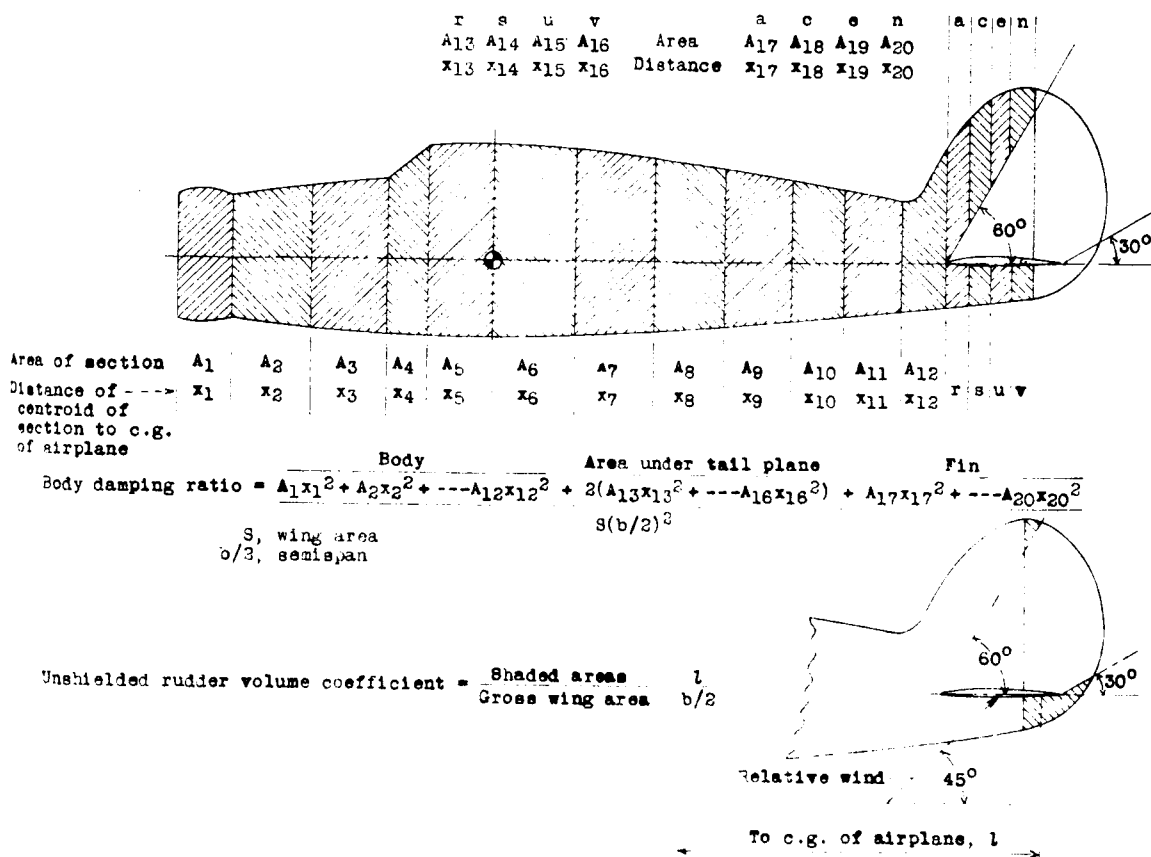


Figure 1.- Method of computation of damping coefficients.

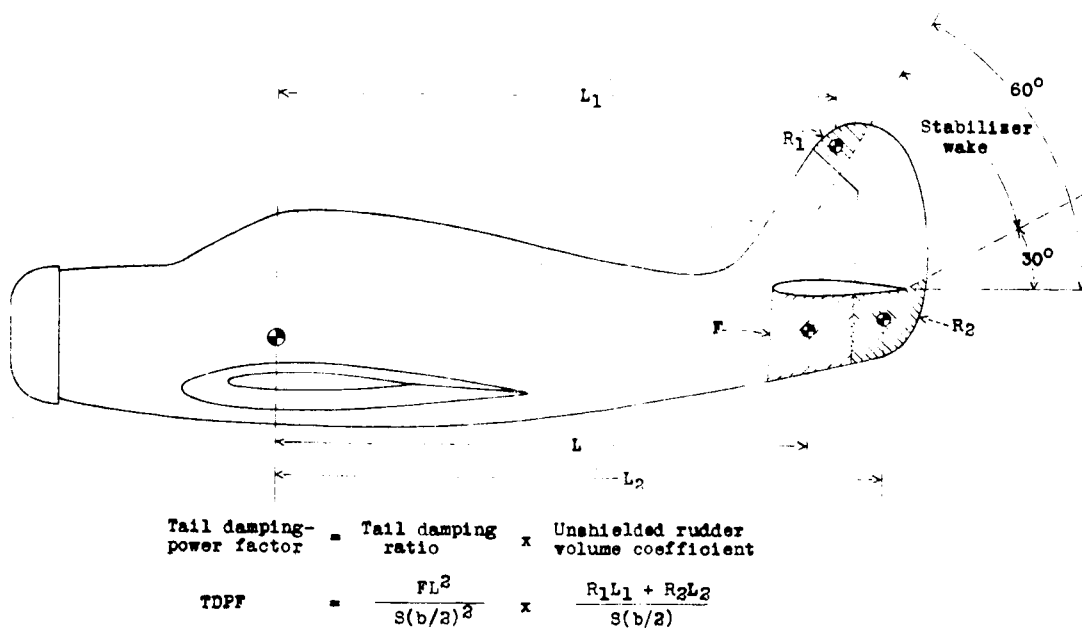


Figure 9.- Method of computing tail damping-power factor.

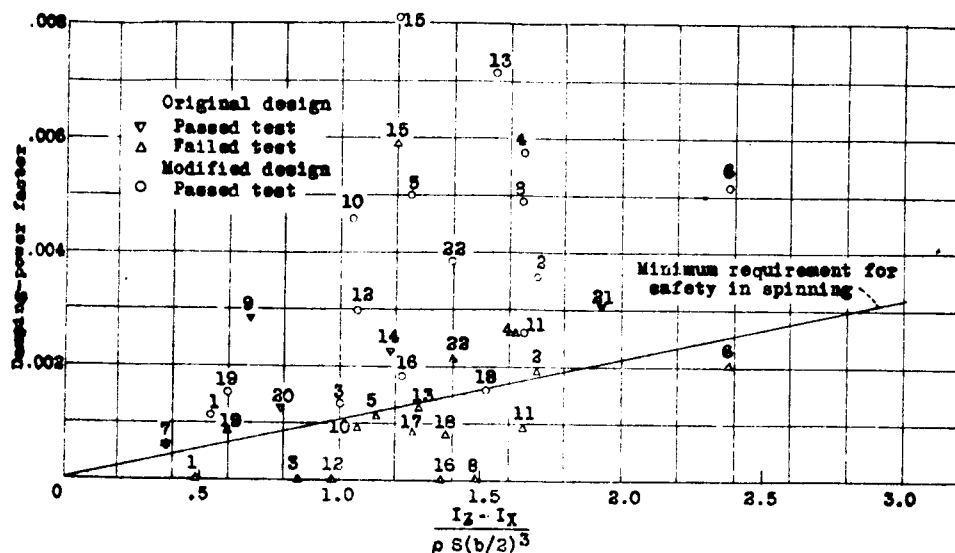


Figure 2.- Variation of damping-power factor with inertia pitching parameter for 23 British monoplanes (From reference 1).

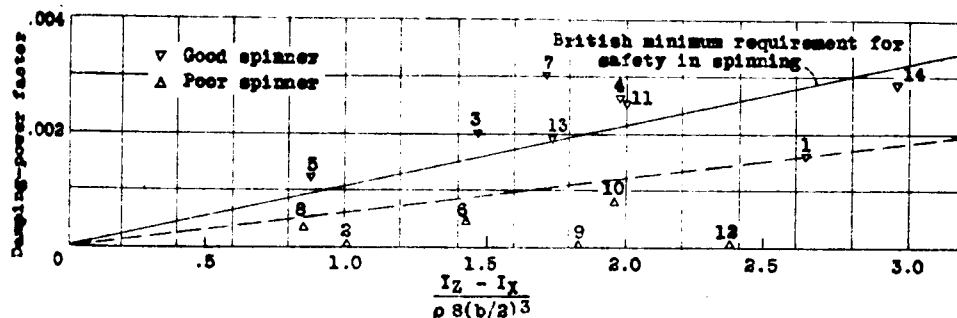


Figure 3.- Variation of damping-power factor with inertia pitching parameter for 14 American monoplanes.

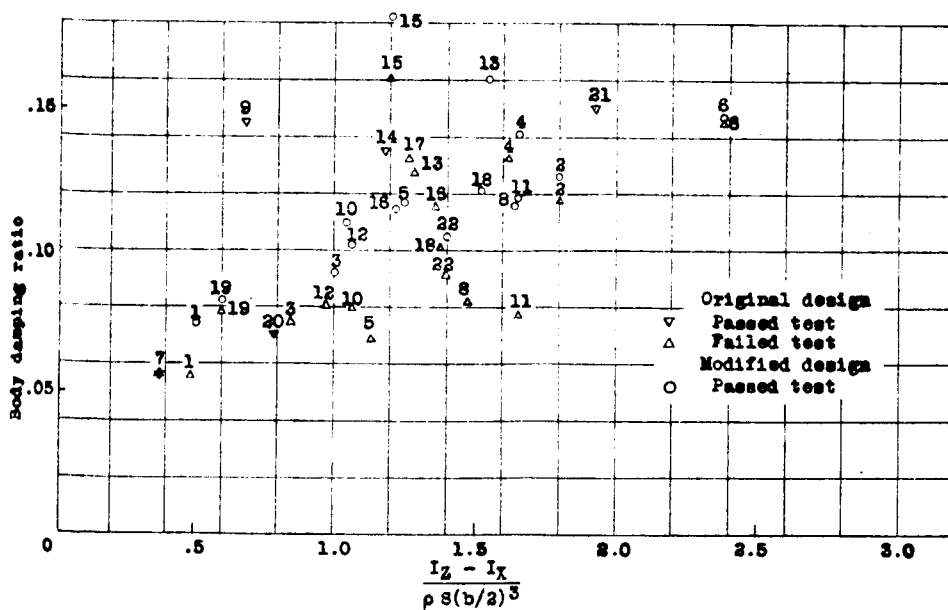


Figure 4.- Variation of body damping ratio with inertia pitching parameter for 23 British monoplanes (From reference 1).



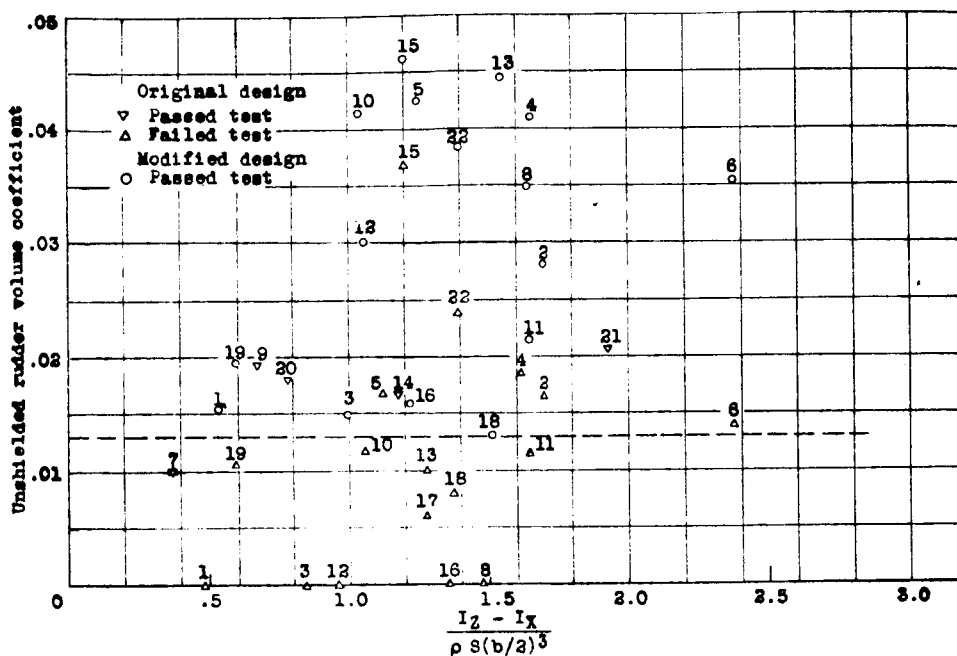


Figure 5.- Variation of unshielded rudder volume coefficient with inertia pitching parameter for 22 British monoplanes (From reference 1).

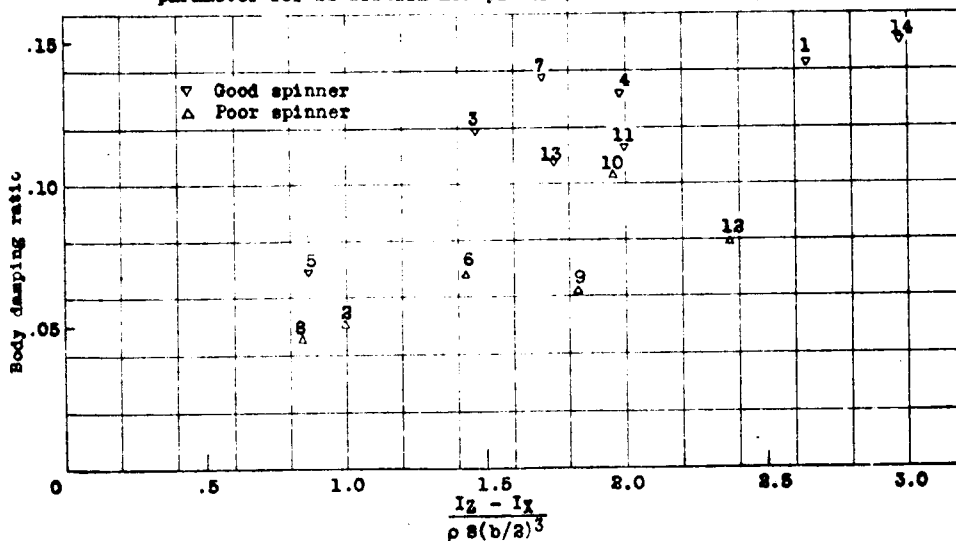


Figure 6.- Variation of body damping ratio with inertia pitching parameter for 14 American monoplanes.

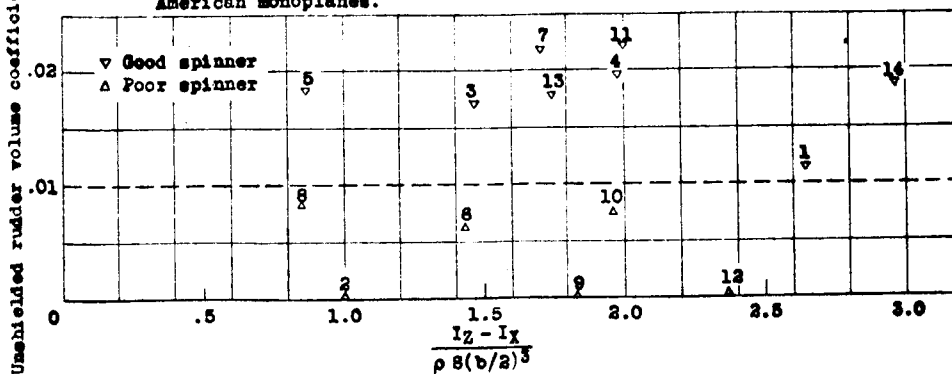


Figure 7.- Variation of unshielded rudder volume coefficient with inertia pitching parameter for 14 American monoplanes.

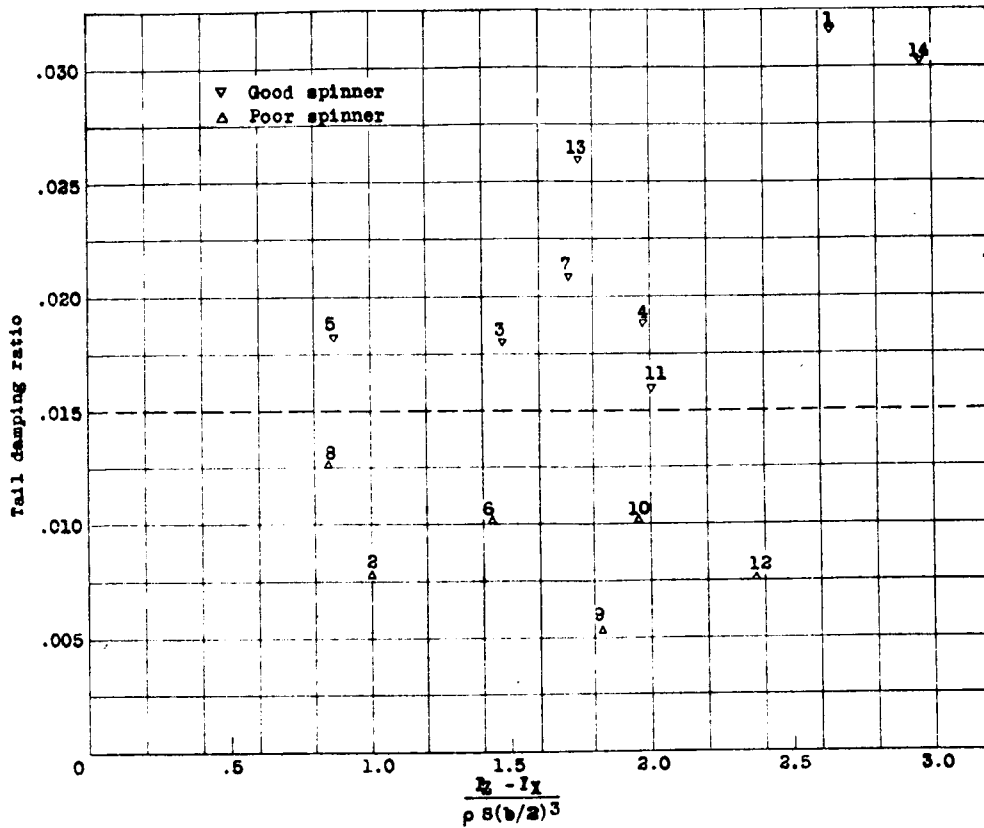


Figure 8.- Variation of tail damping ratio with inertia pitching parameter for 14 American monoplanes.

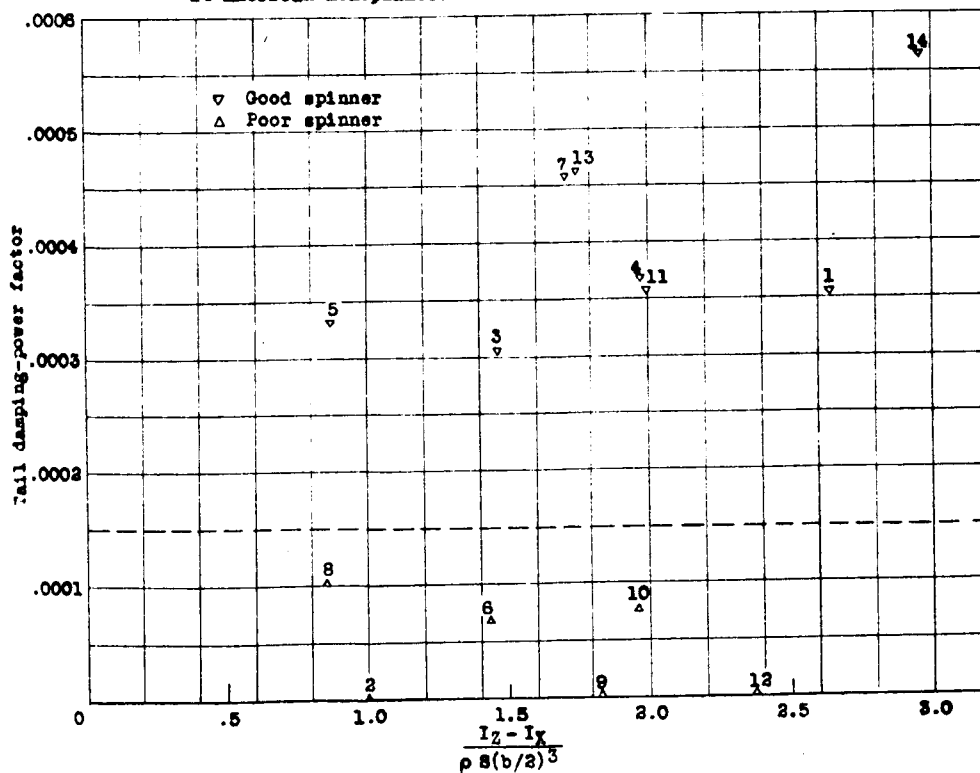


Figure 10.- Variation of tail damping-power factor with inertia pitching parameter for 14 American monoplanes.

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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CONFIDENTIAL MEMORANDUM REPORT

for the  
Materiel Division, Army Air Corps, Wright Field  
SPIN TESTS OF A 1/20-SCALE MODEL OF  
THE XP-39 AIRPLANE  
By CHARLES J. DONLAN

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March 15, 1939

# UNCLASSIFIED REPORT

for the

Material Division, Army Air Corps, Wright Field

SPIN TESTS OF A 1/20-SCALE MODEL OF

THE XP-39 AIRPLANE

By CHARLES J. DONLAN

## INTRODUCTION

The tests reported herein were performed at the request of the Army Air Corps, Material Division, to determine the spinning characteristics of a 1/20-scale model of the Bell XP-39 airplane. The investigation included the study of the effects of loading changes and of various control dispositions. Subsequent tests were performed to determine the effect of a change in wing dihedral.

Preliminary results have already been forwarded to the Material Division.

## APPARATUS AND MODEL

The tests were performed in the N. A. C. A. free-spinning wind tunnel, as described in reference 1.

The 1/20-scale model was furnished by the Material Division and was built from model drawing 4X005 of the Bell Aircraft Corporation. The dimensions of the model were not checked by the N. A. C. A.

The model was constructed principally of balsa wood. The fuselage was hollowed and cut-outs were made in the wings in order to reduce the weight. Lead ballast was placed at suitable locations in order to bring the weight, the moments of inertia, and the center-of-gravity position to the desired values. The ailerons and flaps were installed by the N. A. C. A. from model drawing 4X005 of the Bell Aircraft Corporation. The landing gear was independently ballasted so that the proper change in mass distribution would result when it was extended.

The weight, the moments of inertia, and the center-of-gravity position were taken from report 4Z010 of the Bell Aircraft Corporation. In the model these factors were held to the true scaled-down values within the following limits:

Weight - - - - -  $\pm 1$  percent

Center-of-gravity location - - - -  $\pm 1$  percent of root chord

Moments of inertia - - - - -  $\left\{ \begin{array}{l} \text{A, 0 percent to 6 percent} \\ \text{B, -1 percent to 5 percent} \\ \text{C, -6 percent to 0 percent} \end{array} \right.$

Photographs of the model are shown in figure 1.

#### TEST CONDITIONS

Maximum control displacements of  $\pm 30^\circ$  for the rudder,  $35^\circ$  up and  $15^\circ$  down for the elevators,  $45^\circ$  down for the flaps,

25° up and 10° down for the differential ailerons were taken from drawing 4X005 of the Bell Aircraft Corporation. When the flaps were lowered the ailerons drooped 20° down and the maximum deflections of 25° up and 10° down were measured from the drooped position. The original model as tested had 0° wing dihedral.

The normal model loading condition at an equivalent test altitude of 6000 feet ( $\rho = 0.001988$  slugs/ft.<sup>3</sup>) corresponded, within the limits of accuracy previously mentioned, to the following mass distribution of the full-scale airplane with cockpit covered, flaps up, and landing gear retracted:

|                   |                            |
|-------------------|----------------------------|
| Weight - - - - -  | 5,834 lb                   |
| $x/c_r$ - - - - - | 26.4 percent               |
| $z/c_r$ - - - - - | 10 percent                 |
| A - - - - -       | 2074 slug/ft. <sup>2</sup> |
| B - - - - -       | 4358 slug/ft. <sup>2</sup> |
| C - - - - -       | 6113 slug/ft. <sup>2</sup> |

Where  $x/c_r$  is the ratio of the distance of the center of gravity aft of the leading edge of the root chord to the root chord.

$z/c_r$  is the ratio of the distance of the center of gravity below the thrust line to the root chord.

A, B, and C are the moments of inertia about the body axes X, Y, and Z, respectively.

Tests were performed to determine the effect of variations in moments of inertia and of center-of-gravity positions upon the spinning characteristics of the model. To obtain these loading variations, provision was made for distributing weights along the wings (changing the moments of inertia A and C), for distributing weights along the fuselage (changing the moments of inertia B and C), and for moving a large fuselage weight fore and aft (changing the center-of-gravity position).

### RESULTS

The results of the investigation are presented in charts I - XII and table 1. The steady spin parameters presented in the charts and tables were determined by methods described in reference 1 and have been converted to corresponding full-scale values.  $\theta$  is the angle between the thrust axis and the horizontal (negative for the nose-down position) and determines the angle of attack, which is approximately equal to  $90^\circ + \theta$ .  $\phi$  is the angle between the span axis and the horizontal and is considered positive when the right wing is down. The angle of sideslip,  $\beta$ , is approximately equal to  $\phi - \sigma$ , where  $\sigma$  is the helix angle (angle between flight path and the vertical). (Inward sideslip is considered positive in a right spin, and negative in a left;  $\sigma$  is considered positive in a right spin and negative in a left.) For the XP-39 model spins, the numerical value of  $\sigma$  was about  $40^\circ$ .

Recoveries were measured by the number of turns the spinning model made from the time the controls were observed to move until the spinning motion ceased.

#### DISCUSSION

Comparison between model and airplane spin results (references 1, 2, and 3) indicates that because of scale and tunnel effects, lack of detail in the model and differences in operators' technique, the spin-tunnel results are not always in complete agreement with full-scale spinning data. In general, for a given loading condition and control setting, the model steady-spin results have shown a somewhat smaller angle of attack, a somewhat higher rate of descent, and, at a given angle of attack, from  $5^{\circ}$  to  $10^{\circ}$  more outward sideslip. The comparison made in reference 3 showed that 80 percent of model recovery tests predicted satisfactorily the corresponding full-scale recoveries, 10 percent overestimated the full-scale recoveries, and 10 percent underestimated the full-scale recoveries.

Because of fin-offset on the XP-39 model left spins were steeper than right spins, and since, for a given model, fastest recoveries are usually associated with the steepest spins, the majority of spins were made to the right. Unless otherwise specified the following discussion concerns right spins of the original model with  $0^{\circ}$  dihedral.



Steady spins. - Chart I shows the effect of control setting on the steady spin for the model in the normal loading condition. With the rudder set for the spin, ailerons neutral, and elevator up, the angle of attack was about  $49^{\circ}$ , the rate of descent about 171 feet per second, and  $\frac{\Omega b}{2V}$  was about 0.23. The angle of the wing to the horizontal was small. With the elevator neutral or down the angle of attack increased, the rate of descent increased, and  $\frac{\Omega b}{2V}$  increased. With the ailerons set against the spin (right aileron down  $10^{\circ}$  and left aileron up  $25^{\circ}$ ),  $\frac{\Omega b}{2V}$  increased somewhat with down-setting of the elevator but there was little change in angle of attack or rate of descent for any elevator setting.

With the rudder neutral the model would spin only with elevator down and ailerons against the spin. With this control disposition the spin was steeper and the rate of rotation slightly slower than the corresponding spin with the rudder set full with.

The model would not spin for any elevator-aileron combination with the rudder set against the spin.

Setting the ailerons against the spin resulted in flatter spins and decreased rates of descent, while setting the ailerons with the spin resulted in steeper spins and such high rates of descent that it was impossible to hold these spins in the tunnel.

Chart II indicates that opening the cockpit had very little effect on the steady-spin characteristics of the model. (The cockpit opening simulated on the model was in the form of a sliding canopy over the pilot's head.) The effect of a  $3^{\circ}$  increase in wing dihedral is also presented on chart II. The spins with  $3^{\circ}$  dihedral wandered considerably more than those with  $0^{\circ}$  dihedral, particularly with up-settings of the elevator. In general, increasing the dihedral decreased the angle of attack, increased the rate of descent and slightly increased the rate of rotation.

The effect of a forward movement of center-of-gravity position - a loading associated with the slowest recoveries and hereafter referred to as the "worst load" - is shown in chart III. In general, the effect of the worst load was to decrease the angle of attack somewhat, decrease the vertical velocity slightly, and increase the rate of rotation slightly.

The effect of flaps and drooped ailerons on the steady-spin characteristics is presented in chart IV. Drooping the ailerons  $20^{\circ}$  down in conjunction with lowering the flaps  $45^{\circ}$  resulted in lower rates of descent, and when the elevator was down the spin flattened somewhat. With the ailerons set against the spin, the spins were flatter for all elevator settings and the rates of descent materially reduced. The model would not spin in this condition when the ailerons were deflected with the spin.

Recoveries. - The results of the recovery tests for the normal loading condition are presented in chart I and table 1. Recoveries from steady spins with ailerons neutral, rudder with the spin, and elevator up were rapid by reversal of rudder alone or by simultaneous reversal of both rudder and elevator. Slower recoveries were obtained from the elevator neutral and elevator down spins, with the latter spins giving the slowest recoveries by reversal of rudder alone.

Steady spins were not obtained with the ailerons set for the spin and consequently no recovery tests were attempted. It would appear that recovery from normal spins might be improved by displacing the ailerons with the spin. This was not tried, however.

Aileron settings against the spin tended to retard recovery except with elevator-up settings, in which instances recoveries were slightly faster.

Although chart I indicates the model would not attain a steady spinning equilibrium with neutral rudder setting (with one exception), recoveries from rudder-with spins attempted by neutralizing the rudder alone were generally unsatisfactorily long (table 1).

The effects on recovery of various changes in loading (charts V, VI, and VII) were negligible with the exception of a forward movement of the center-of-gravity position -

about 6 inches full-scale - which tended to retard recovery for most elevator and aileron dispositions (chart VII ). The effect was greater with down-settings of the elevator and less with neutral and up-settings of the elevator.

The effect on recovery of various landing combinations is given in chart VIII. Opening the cockpit had only a slight adverse effect. Lowering the landing gear gave slight increases in turns for recovery. Lowering the flaps in conjunction with drooping the ailerons had a detrimental effect on recovery for certain aileron deflections. With the ailerons set for the spin the model would recover on its own accord when forced into the spin, whereas with ailerons set against the spin the model would not recover by reversal of rudder. This critical effect is attributable to the drooped ailerons, as can be seen from chart IV. With normal aileron setting against the spin (right aileron  $10^{\circ}$  down, left aileron  $25^{\circ}$  up), the effect on recovery of the lowered flap was slight but with drooped ailerons set against the spin (right aileron  $30^{\circ}$  down, left aileron  $5^{\circ}$  up), no recovery occurred when the rudder was reversed.

Charts IX, X, and XI reveal the effects of dihedral. Increasing the dihedral from  $0^{\circ}$  to  $5^{\circ}$  had a small beneficial effect on recovery.

A comparison of left and right spins is presented in chart XII. In general, left spins exhibited slightly faster

recoveries than the corresponding right spins.

### CONCLUSIONS

To effect the most rapid recovery from the spin the following procedure is recommended: During the steady spin hold the stick full back, the rudder full with the spin, and the ailerons full with the spin (right aileron up in a right spin). For recovery move the rudder hard against the spin and later, after an additional one-half turn has been made, briskly move the elevators full down, still keeping the ailerons full with the spin.

The model tests of the XP-39 also indicate the following spin characteristics:

1. In the high-speed condition, with normal load, the airplane will spin normally. Recovery by reversal of the rudder alone, or by reversal of the rudder followed by downward deflection of the elevator will be satisfactory. If the rudder is reversed after the elevator has been moved down, the recovery will be slower. Recovery by neutralizing the rudder will be unsatisfactory. Ailerons deflected against the spin will retard recovery; ailerons deflected with the spin will improve recovery.

2. In the landing condition (flaps and wheels down, and ailerons drooped), the aileron effect mentioned previously will be considerably increased,

and failure to recover may result if ailerons are deflected against the spin.

3. Reasonable changes in mass distribution will have only a minor effect on recoveries. Forward movement of the center of gravity will tend to retard recovery and aft movement to improve it.

4. Increasing the wing dihedral from  $0^\circ$  to  $3^\circ$  will produce a slight beneficial effect on recoveries.

Langley Memorial Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., March 15, 1939.

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EC

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TABLE

XP-39

RECOVERIES (RIGHT SPINS)

## EFFECT OF CONTROL SETTING

| MODEL CONDITION | Control Setting |             |                 |               |                 |               |        | Turns<br>for<br>Recovery |
|-----------------|-----------------|-------------|-----------------|---------------|-----------------|---------------|--------|--------------------------|
|                 | Ailerons        |             | Rudder          |               | Elevator        |               |        |                          |
|                 | Rt.<br>Deg.     | Lt.<br>Deg. | Initial<br>Deg. | Final<br>Deg. | Initial<br>Deg. | Final<br>Deg. |        |                          |
| Normal ✓        |                 |             |                 |               |                 |               |        | ✓                        |
|                 | 25U             | 10D         | 30W             | 30A           | 35U             | 15D           |        |                          |
| "               | 0               | 0           | 30W             | 30A           | 35U             | 15D           | 1½, 1½ |                          |
| "               | 10D             | 25U         | 30W             | 30A           | 35U             | 15D           | 1, 1   |                          |
| "               |                 |             |                 |               |                 |               |        |                          |
| "               | 0               | 0           | 30W             | 0             | 35U             | 35U           | 3¼, 3  |                          |
| "               | 10D             | 25U         | 30W             | 0             | 35U             | 35U           | 3¼     |                          |
| "               | 0               | 0           | 30W             | 0             | 0               | 0             | ∞, ⑦   | ✓                        |
| "               | 10D             | 25U         | 30W             | 0             | 0               | 0             | ⑥, ⑤   |                          |
| "               | 0               | 0           | 30W             | 0             | 15D             | 15D           | ⑧      |                          |
| "               | 10D             | 25U         | 30W             | 0             | 15D             | 15D           | ⑦      |                          |
| "               |                 |             |                 |               |                 |               |        |                          |
| "               |                 |             |                 |               |                 |               |        |                          |
| "               |                 |             |                 |               |                 |               |        |                          |
| "               |                 |             |                 |               |                 |               |        |                          |
| "               |                 |             |                 |               |                 |               |        |                          |

✓ In the normal loading condition, the landing gear was retracted and the cockpit covered

2/ Wanders too greatly to test.

3/ ∞ = No recovery ; ⑦ = No recovery in 7 turns.

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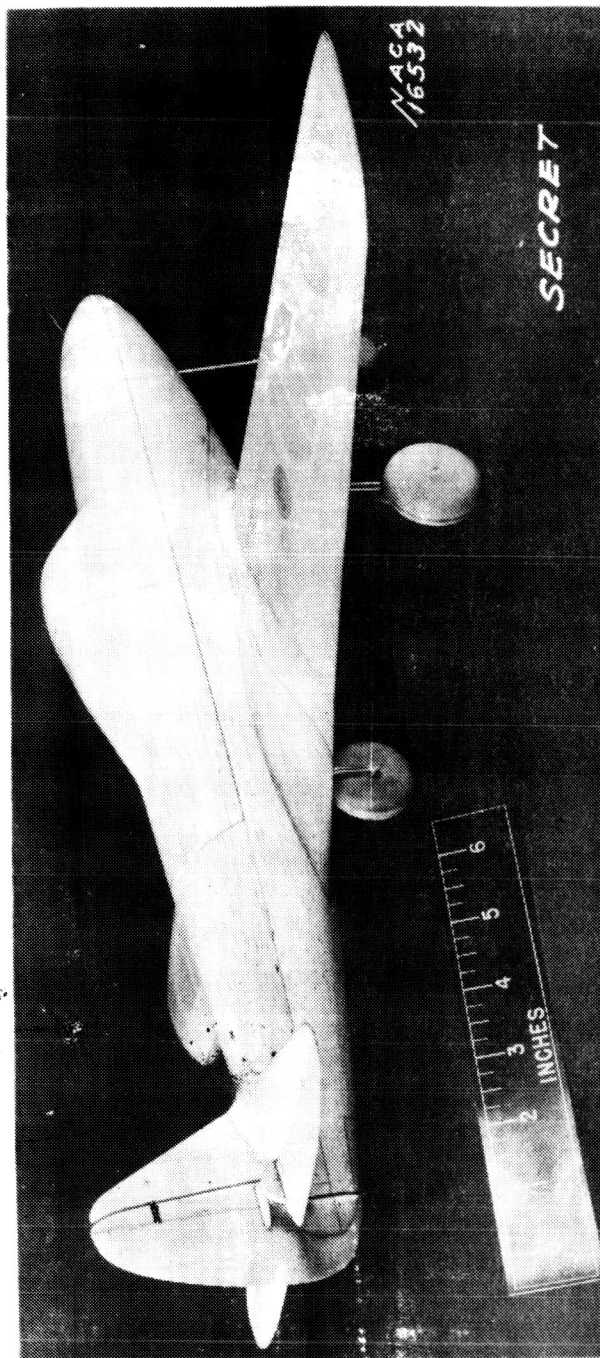
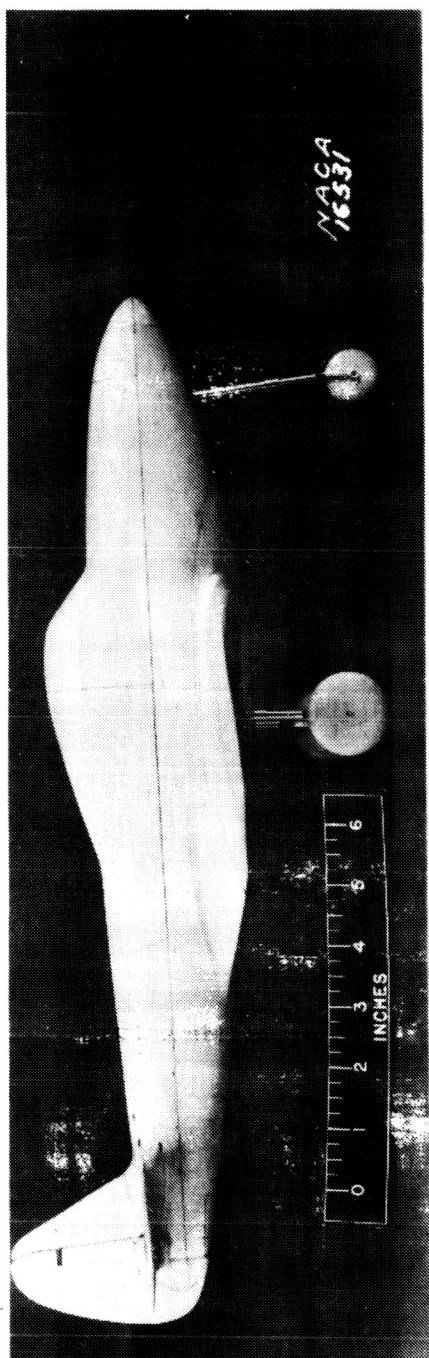


Figure 1. - 1/20-scale model of the XP-59 wing.

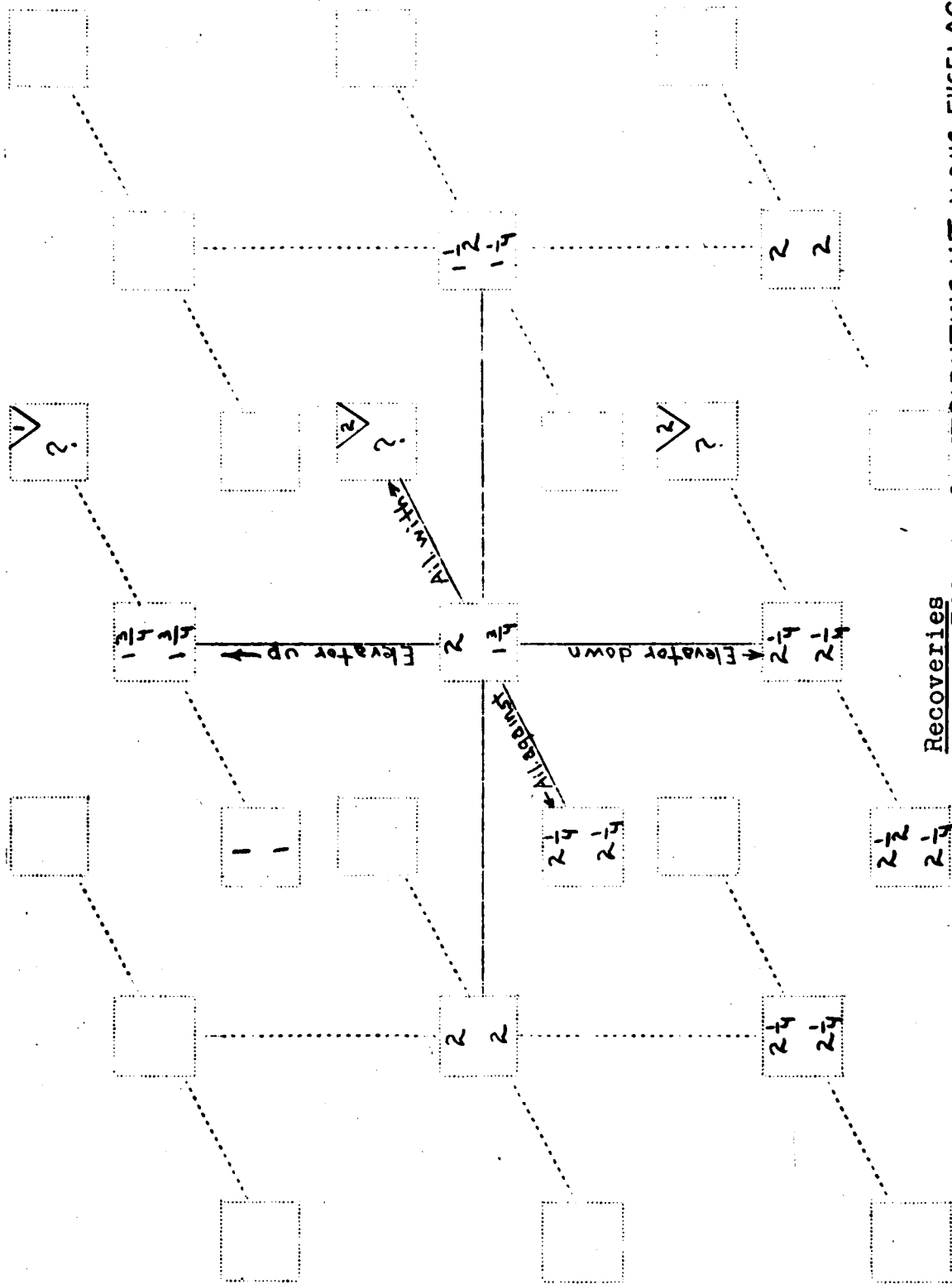
Note: Chart shows control disposition prior to recovery attempt. Figures in ~~circles~~ are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART V

Wt. extended along fuselage  
(B and C increased 20% of B)

Normal

Wt. retracted along fuselage  
(B and C decreased 20% of B)



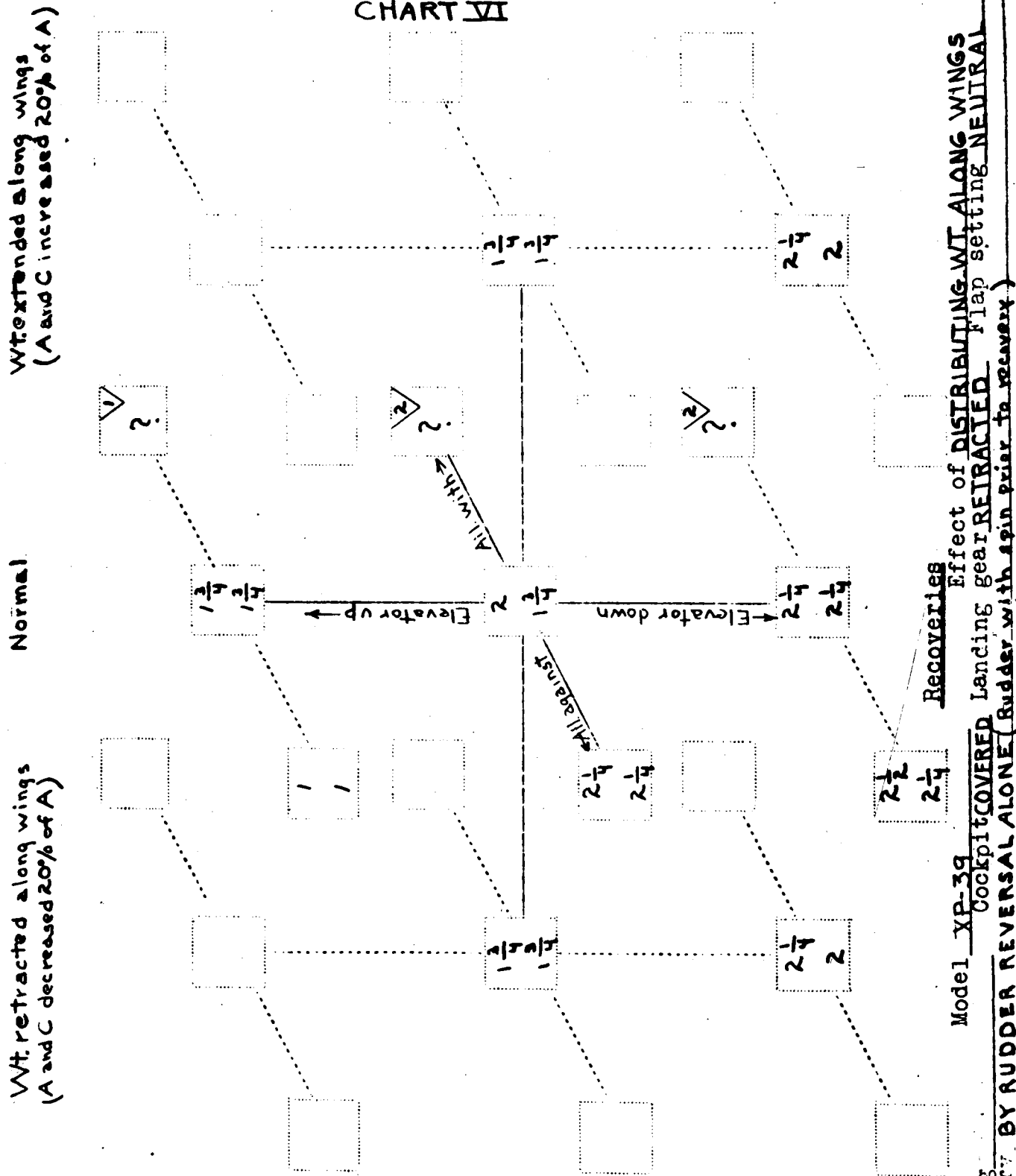
Model XP-39 Cockpit COVERED Effect of DISTRIBUTING WT. ALONG FUSELAGE  
Recovery BY RUDDER REVERSAL ALONE (Rudder with spin prior to recovery)  
Recovery BY RUDDER REVERSAL ALONE (Rudder with spin prior to recovery)  
Recovery BY RUDDER REVERSAL ALONE (Rudder with spin prior to recovery)

- 1/ Wanders too greatly to test.
- 2/ Vertical velocity too great for tunnel.

SECRET

Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART VI



Wanders too greatly to test.  
Vertical velocity too great for tunnel

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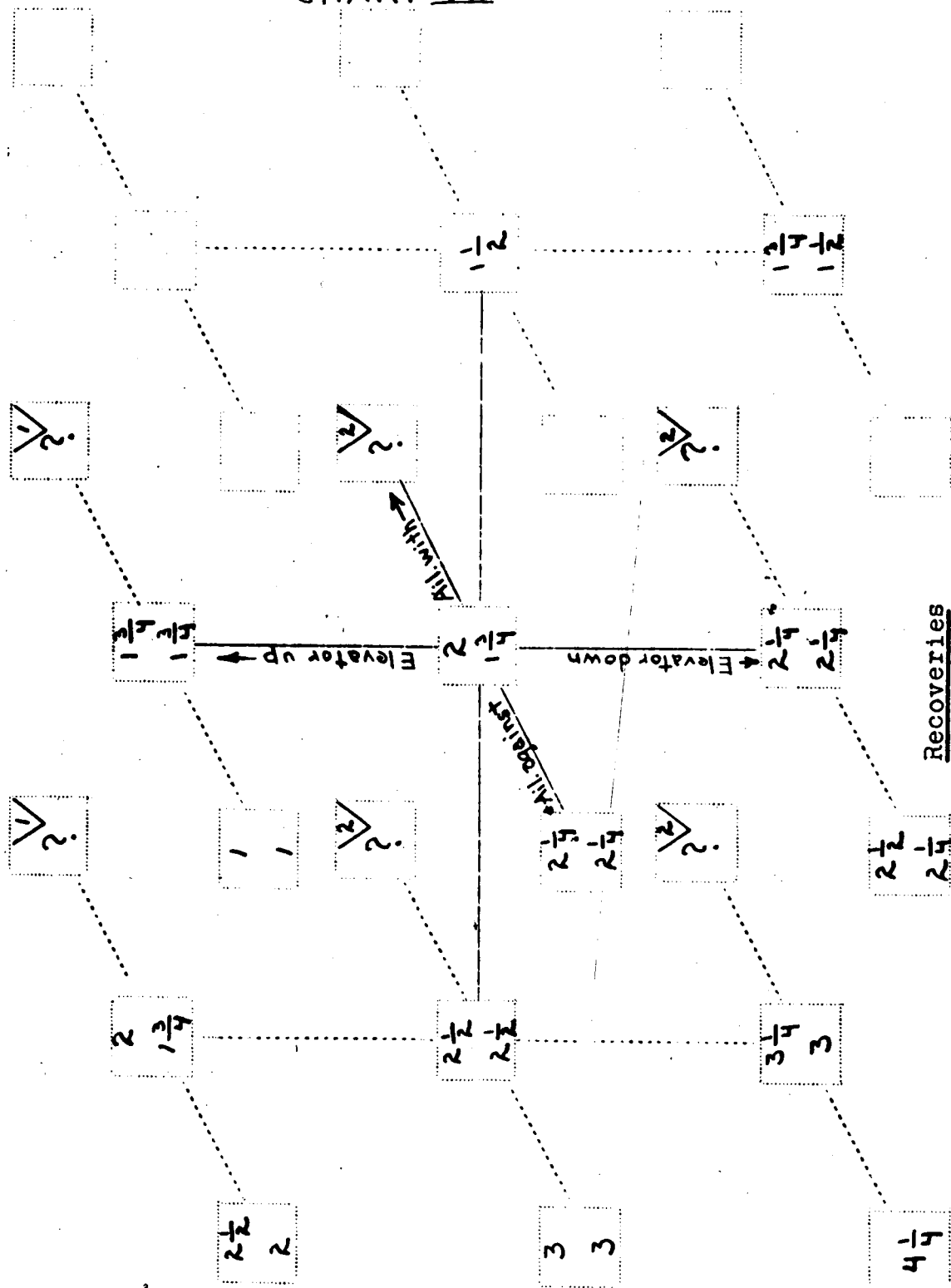
Note: Chart shows control disposition prior to recovery attempt. Figures in ~~circles~~ <sup>squares</sup> are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART VII

C.G. aft 7.38% Root Cord  
(7.38 inches full-scale)

Normal

C.G. forward 5.74% Root Cord  
(5.74 inches full-scale)



Recoveries

Model XP-39

Effect of C.G. POSITION  
Loading NORMAL  
Cockpit COVERED  
Gear RETRACTED  
Flap setting NEUTRAL  
Recovery BY RUDDER REVERSAL ALONE (Rudder with spin prior to recovery)

✓ Wanders too greatly to test.  
✗ Vertical velocity too great for tunnel.

SECRET

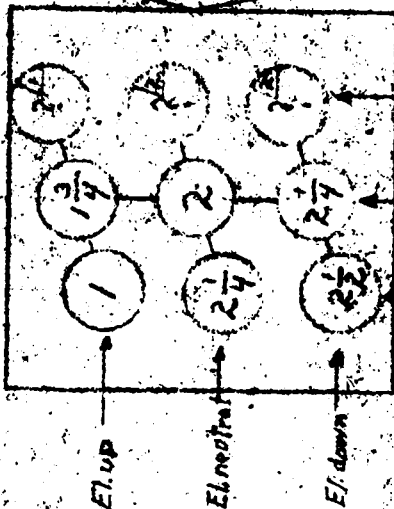
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# XP-37 - NORMAL LOAD LANDING COMBINATIONS

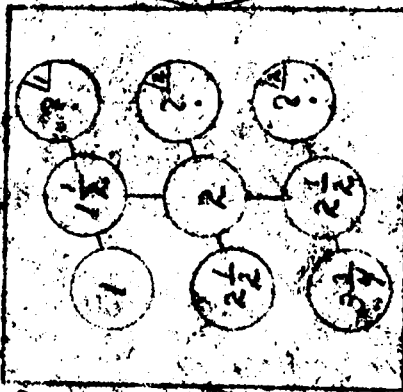
Chart shows control disposition prior to recovery attempt. Figures in circles are turns for recovery. No means motor would not spin.

NOTE: When flaps are low, flaps up - wheels up. Deflections are from this attitude position.

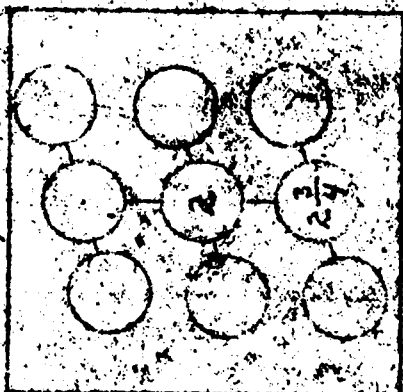
COCKPIT COVERED  
FLAPS UP - WHEELS UP



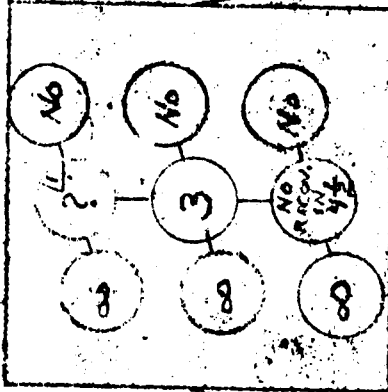
COCKPIT OPEN  
FLAPS UP - WHEELS UP



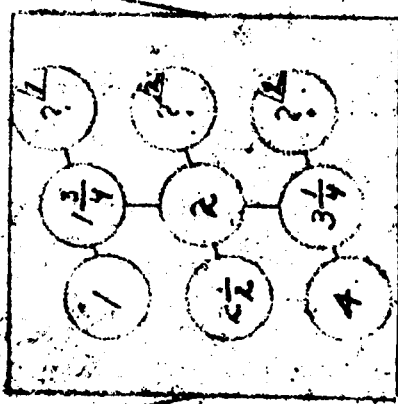
COCKPIT COVERED  
FLAPS UP - WHEELS DOWN



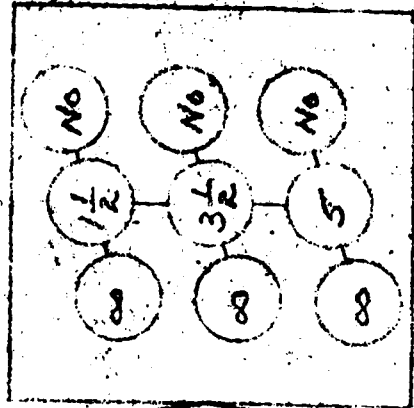
COCKPIT OPEN  
FLAPS DOWN - WHEELS UP



COCKPIT OPEN  
FLAPS UP - WHEELS DOWN



COCKPIT OPEN  
FLAPS DOWN - WHEELS DOWN

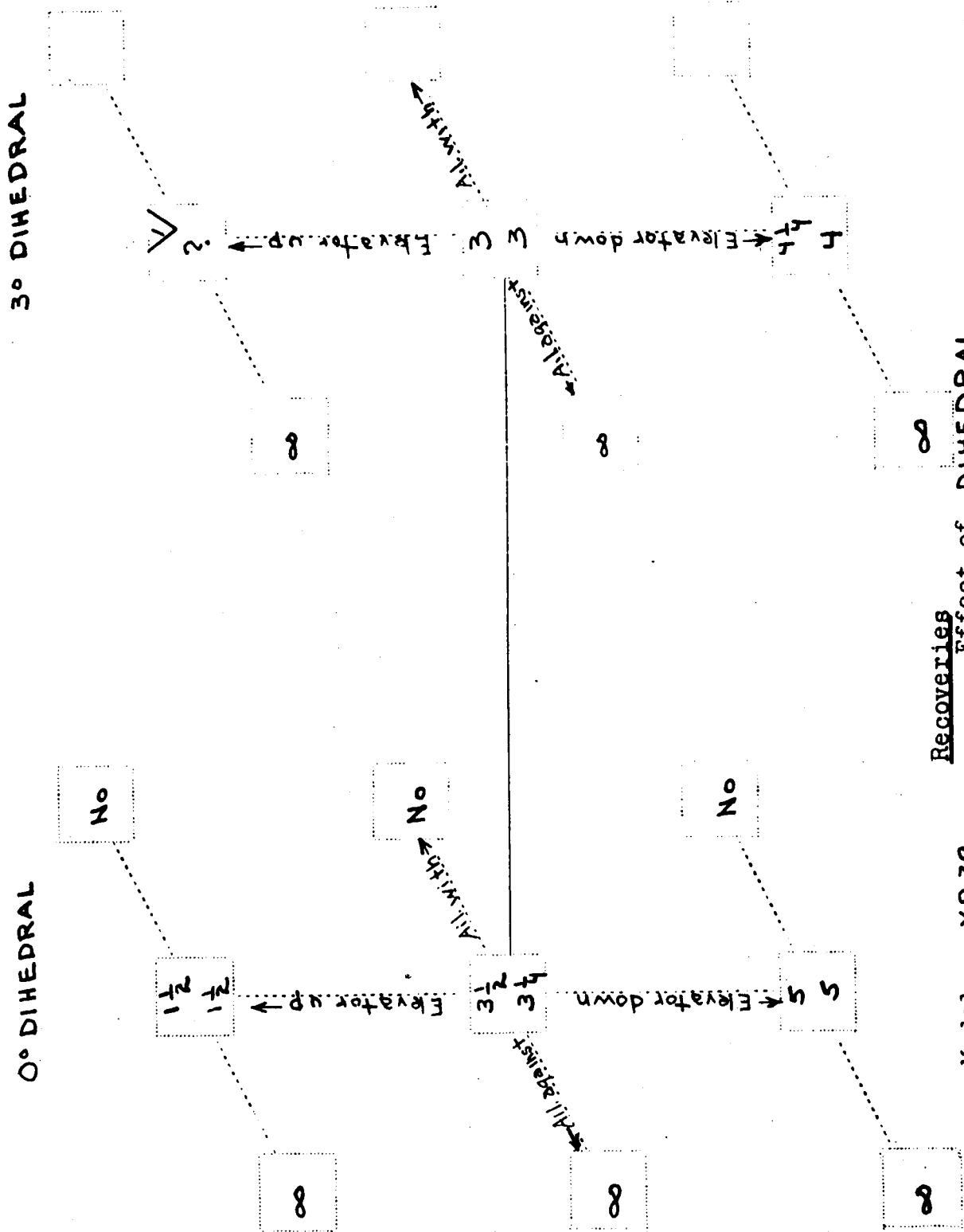


1. Wonders too greatly too fast  
2. Verticed/velocity too great for turns

{ Rudder with spin prior to recovery  
Recovery by rudder reversal alone

Note: Chart shows control disposition prior to recovery attempt.  
 Figures in ~~circles~~ <sup>squares</sup> are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

# CHART IX



## Recoveries

Model XP-39 Effect of DIHEDRAL Flap setting 45° DOWN (Dropped AI-20°)  
 Loading NORMAL Cockpit UNCOVERED Landing gear EXTENDED  
 Recovery BY RUDDER REVERSAL ALONE (Rudder full with prior to recovery)

Wanders too greatly to test.

SECRET

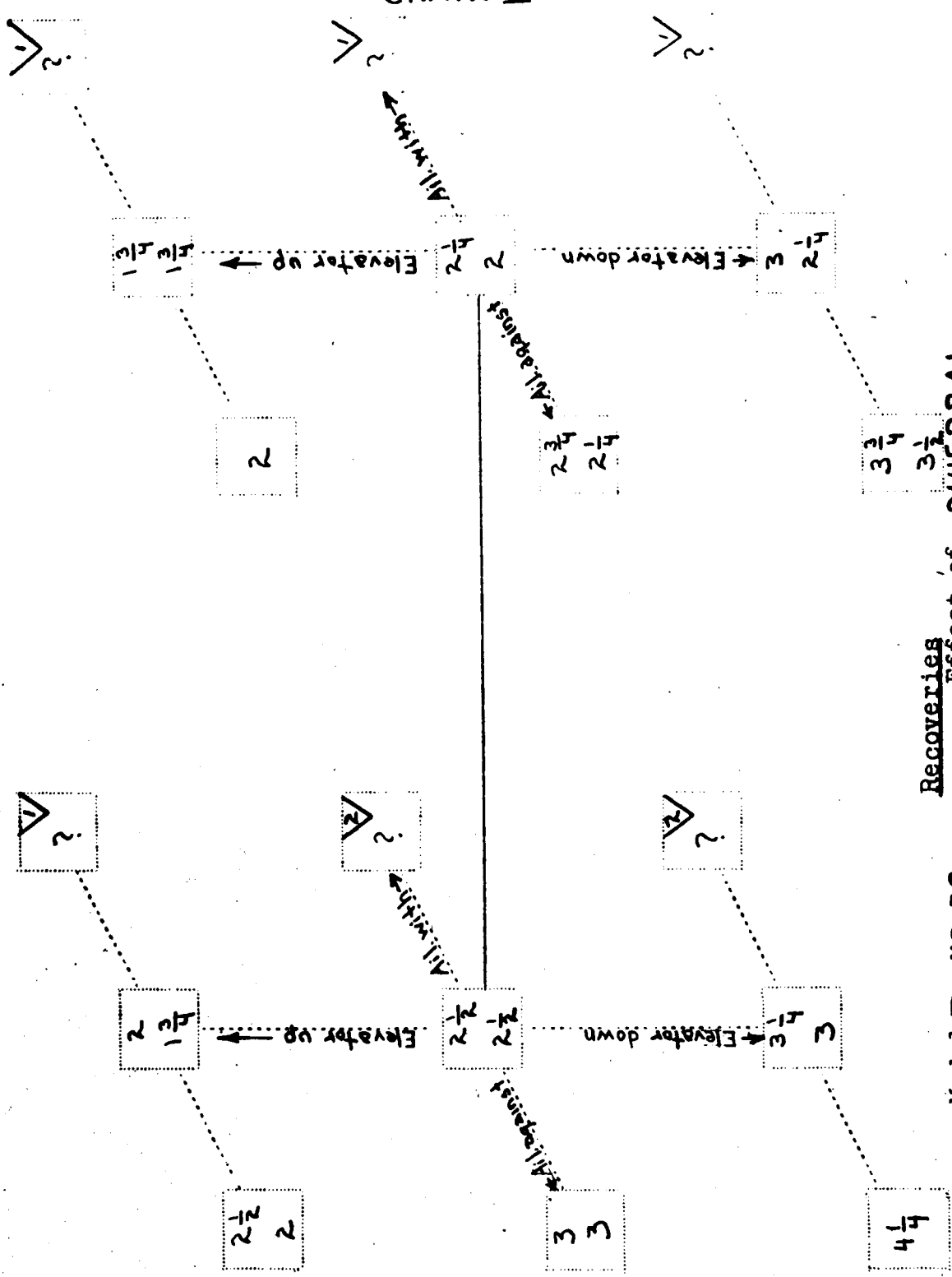
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Note: Chart shows control disposition prior to recovery attempt.  
 Figures in ~~circles~~ <sup>squares</sup> are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART X

3° DIHEDRAL

0° DIHEDRAL



Recoveries Effect of DIHEDRAL  
 Model XP-39  
 Loading CG 6" forward of cockpit **COVERED**  
 Recovery BY RUDDER REVERSAL ALONE (Rudder full with prior to recovery)  
 Landing gear **RETRACTED** Flap setting **NEUTRAL**

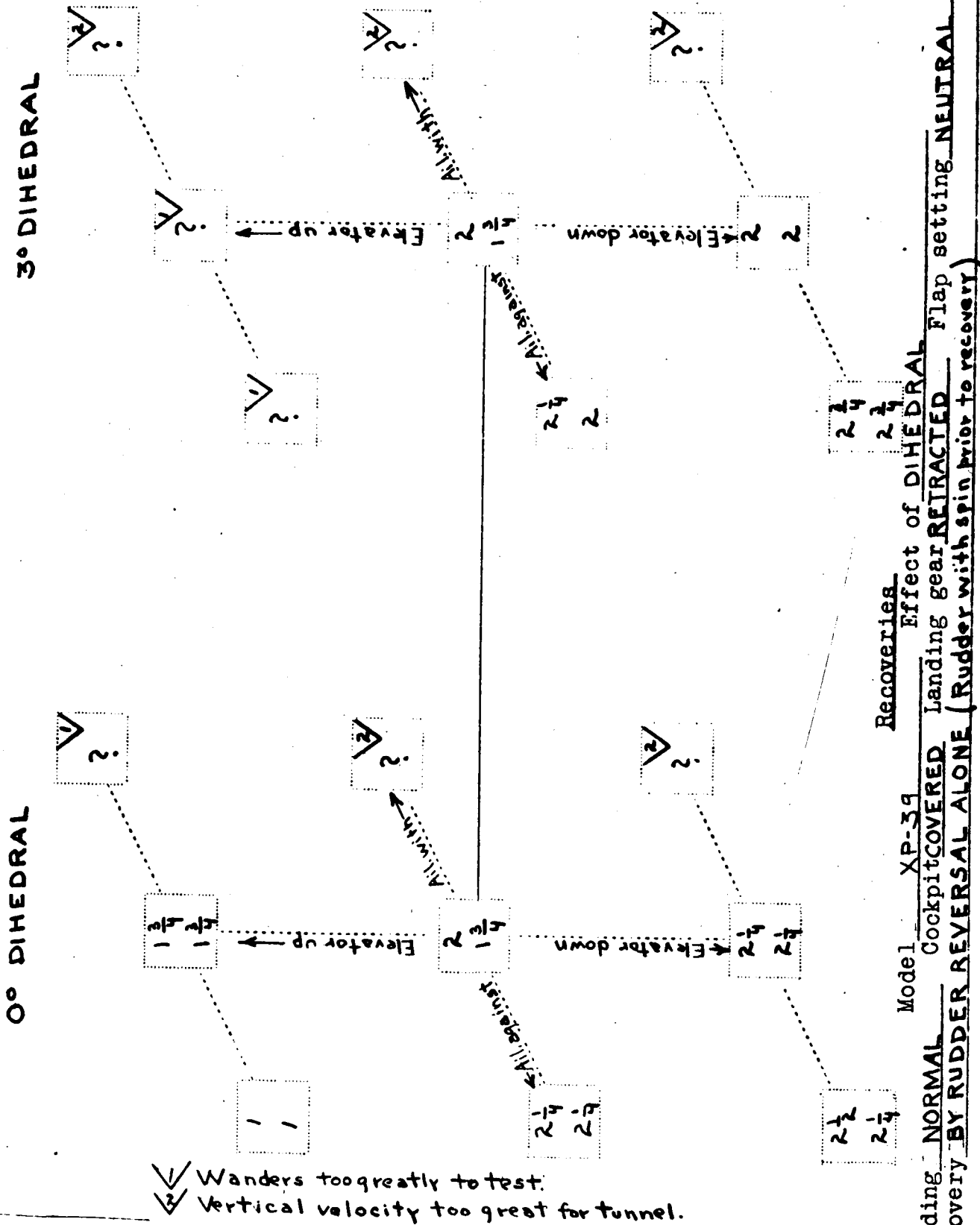
Wanders too greatly to test  
 Vertical velocity too great for tunnel.

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~~SECRET~~

## CHART XI



**SECRET**

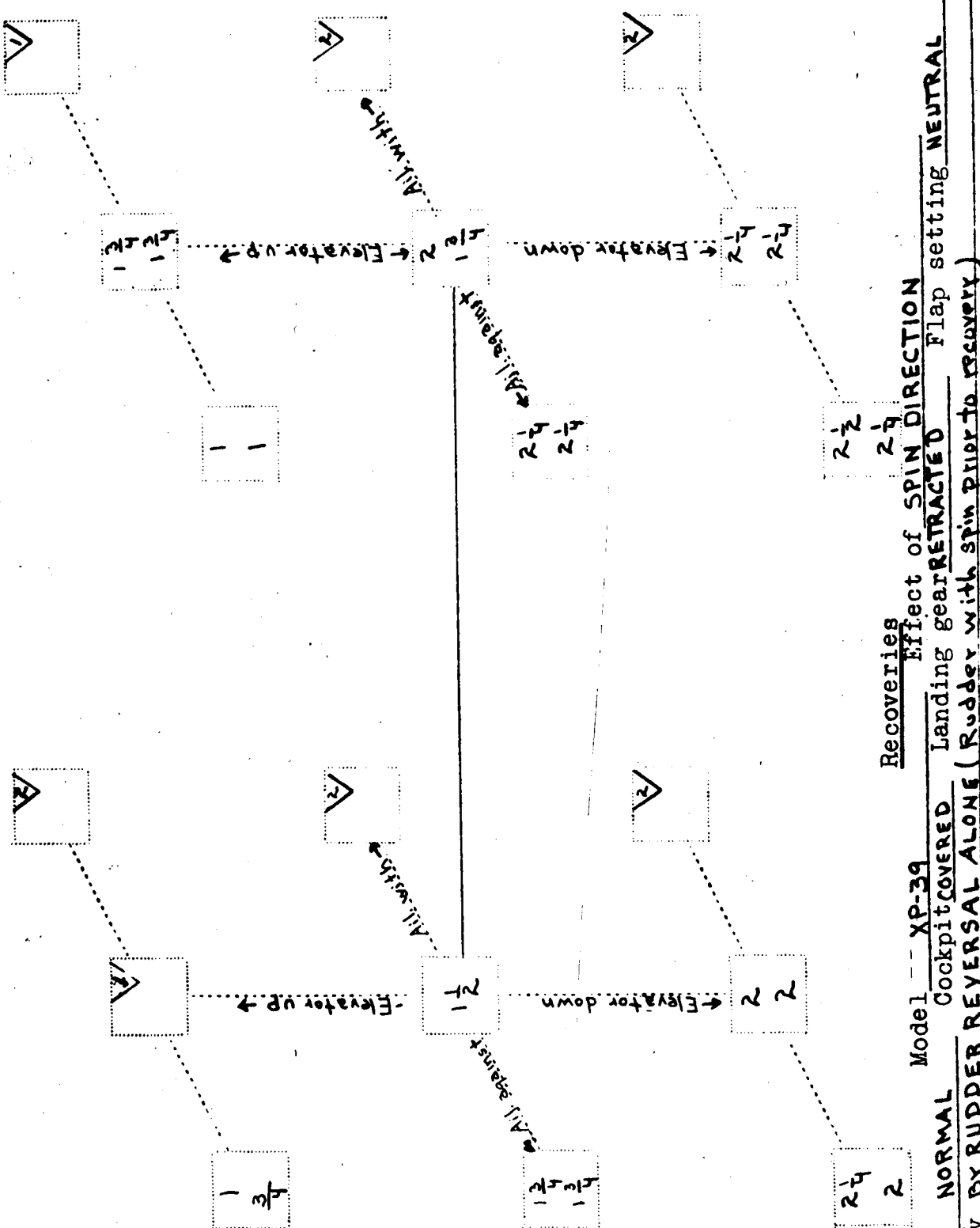


Note: Chart shows control disposition prior to recovery attempt.  
 Figures in circles are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

# CHART XII

RIGHT SPINS

LEFT SPINS



Wanders too greatly to test.  
 Vertical velocity too great for tunnel.

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Model -- XP-39  
 Cockpit COVERED  
 Landing gear RETRACTED  
 Flap setting NEUTRAL  
 Recovery BY RUDDER REVERSAL ALONE (Rudder with spin prior to recovery)

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

**CONFIDENTIAL**

CONFIDENTIAL MEMORANDUM REPORT

for

Bureau of Aeronautics, Navy Department

SPIN TESTS OF A 1/20-SCALE MODEL OF

THE XF4U-1 AIRPLANE

By CHARLES J. DONLAN

CLASSIFICATION CHANGED

UNCLASSIFIED

To \_\_\_\_\_

By authority of \_\_\_\_\_

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July 12, 1939

CONFIDENTIAL MEMORANDUM REPORT

for

Bureau of Aeronautics, Navy Department

SPIN TESTS OF A 1/20-SCALE MODEL OF

THE XF4U-1 AIRPLANE

By CHARLES J. DONLAN

- Reference: (a) Bureau of Aeronautics Let. to NACA  
dated Aug. 31, 1938, Aer-E-231-FAM  
VF4U-1/F1-1.  
(b) Prel. Memo. Apr. 18, 1939, H. A. Soulé.  
(c) Prel. Memo. June 3, 1939, C. J. Donlan.

INTRODUCTION

These tests were performed at the request of the Bureau of Aeronautics, Navy Department, to determine the spinning characteristics of the 1/20-scale model of the XF4U-1 airplane. The investigation included the study of the effects of loading changes and various control dispositions using both the original and modified vertical tail surfaces. Subsequent tests were made to determine the effects of additional tail modifications.

Preliminary results have already been forwarded to the Bureau of Aeronautics (references (b) and (c)).

#### APPARATUS AND MODEL

The tests were performed in the N. A. C. A. free-spinning wind tunnel, as described in reference 1.

The 1/20-scale model was furnished by the Bureau of Aeronautics, Navy Department and was built from model drawings CV-18909, CV-18910, CV-18911, and CV-S-5593 of the Chance Vought Aircraft Company. The dimensions of the model were not checked by the N. A. C. A.

The model was constructed principally of balsa wood. The fuselage was hollowed and cut-outs were made in the wings to reduce weight. Lead ballast, placed at suitable locations, served to bring the weight, the moments of inertia, and the center-of-gravity position to the desired values. The ailerons and flaps were installed by the N. A. C. A. in accordance with model drawings CV-18909 and CV-18911 of the Chance Vought Aircraft Company. The landing gear was independently ballasted so that the proper change in mass distribution would result when it was extended.

The weight, the moments of inertia, and the center-of-gravity location were taken from Report No. 4589 of the Chance Vought Aircraft Company. In the model these factors were held to the true scaled-down values within the following limits:

Height - - - - -  $\pm 1$  percent  
Center-of-gravity location - - -  $\pm 1$  percent of mean  
aerodynamic chord  
Moments of inertia - -  $\left\{ \begin{array}{l} I_x, \text{ 0 percent to 6 percent} \\ I_y, \text{ 0 percent to 6 percent} \\ I_z, \text{ 0 percent to 6 percent} \end{array} \right.$

A photograph of the model with the original vertical tail is shown in figure 1.

#### TEST CONDITIONS

The maximum control displacements used were as follows: (1)  $\pm 25^\circ$  for the rudder, (2)  $30^\circ$  up and  $20^\circ$  down for the elevators, (3)  $40^\circ$  down and  $60^\circ$  down for the flaps, and (4)  $20^\circ$  up and  $10^\circ$  down for the ailerons. (1) and (2) were taken from drawing CV-S-18001; (3), from drawing CV-S-18911; and (4), from the stick-aileron chart supplied by the Chance Vought Aircraft Company. When the flaps were lowered the ailerons drooped  $10^\circ$  down, and the maximum aileron deflections in the drooped condition were  $15^\circ$  up and  $25^\circ$  down from neutral.

The normal model loading condition at an equivalent test altitude of 8,000 feet ( $\rho = 0.001869 \frac{\text{slugs}}{\text{ft}^3}$ ) corresponded, within the limits of accuracy previously mentioned, to the following mass distribution of the

full-scale airplane with cockpit covered, flaps up, and the landing gear retracted.

|        |           |                               |
|--------|-----------|-------------------------------|
| Weight | - - - - - | 8,598 pounds                  |
| $x/c$  | - - - - - | 0.249                         |
| $z/c$  | - - - - - | 0.0785                        |
| $I_x$  | - - - - - | 7,400 slug-feet <sup>2</sup>  |
| $I_y$  | - - - - - | 8,072 slug-feet <sup>2</sup>  |
| $I_z$  | - - - - - | 14,421 slug-feet <sup>2</sup> |

where  $x/c$  is the ratio of the distance of the center of gravity aft of the leading edge of the mean aerodynamic chord to the mean aerodynamic chord.

$z/c$  is the ratio of the distance of the center of gravity below the thrust line to the mean aerodynamic chord.

$I_x$ ,  $I_y$ , and  $I_z$  are the moments of inertia about the body axes  $X$ ,  $Y$ , and  $Z$ , respectively.

Tests were performed to determine the effect of variations in moments of inertia and of center-of-gravity position upon the spinning characteristics of the model. To procure these loading variations, provision was made for distributing weights along the wings (changing the moments of inertia  $I_x$  and  $I_z$ ), for distributing

weights along the fuselage (changing the moments of inertia  $I_y$  and  $I_z$ ), and for moving fuselage weights fore and aft (changing the center-of-gravity position).

As is customary in routine spin-tunnel tests, the majority of tests were performed with the tail wheel removed, as this tends toward more conservative results. The few tests performed with the tail wheel in place are specifically noted in the text.

## RESULTS

The results of the investigation are presented in charts I to XX and tables 1 to 4. The steady spin parameters presented in the charts and tables were determined by methods described in reference 1, and have been converted to the corresponding full-scale values. The angle of attack,  $\alpha$ , is the angle between the thrust axis and the vertical. The angle between the span axis and the horizontal is  $\phi$  and is considered positive when the right wing is down. The angle of sideslip,  $\beta$ , is approximately equal to  $\phi - \sigma$ , where  $\sigma$  is the helix angle (angle between flight path and the vertical). (Inward sideslip is considered positive in a right spin, and negative in a left;  $\sigma$  is considered positive in a right spin

and negative in a left.) For the XF4U-1 model spins, the numerical value of  $\sigma$  averaged about  $3^\circ$ .

Recoveries were measured by the number of turns the spinning model made from the instant the controls were observed to move until the spinning motion ceased.

### DISCUSSION

Comparison between model and airplane spin results (references 1 and 2) indicates that because of scale and tunnel effect, absence of detail in the model, and differences in operators' technique, the spin-tunnel results are not always in complete agreement with full-scale spinning data. In general, for a given loading condition and control setting, the model steady-spin results have shown a somewhat smaller angle of attack, a somewhat higher rate of descent, and, at a given angle of attack, from  $5^\circ$  to  $10^\circ$ , more outward sideslip. The comparison made in reference 2 showed that 80 percent of the model recovery tests predicted satisfactorily the corresponding full-scale recoveries, 10 percent overestimated the full-scale recoveries, and 10 percent underestimated the full-scale recoveries.

The majority of test spins were made to the right although a number of check spins were made to the left.



Unless otherwise specified, the following discussion concerns right spins of the model with the original (larger) vertical tail surface.

Chart I and table 1 show the effect of control settings on the steady spin and recovery characteristics of the XF4U-1 model in the normal loading condition. In the normal condition the model would spin only with the rudder held for the spin. The flattest spins occurred with up settings of the elevator and, in general, there was a progressive steepening of the spin as the elevator was lowered to its normal maximum downward deflection of  $20^{\circ}$ . Further downward deflection of the elevator tended to flatten the spin slightly. Setting the ailerons against the spin tended to decrease the angle of attack and increase the rate of descent for neutral and down settings of the elevator but had little effect for up settings of the elevator. Setting the ailerons with the spin had little effect on the steady spin for any elevator setting. The slowest recoveries were effected from spins with up settings of the elevator; the fastest, from spins with down settings of the elevator. In general, setting the ailerons with the spin retarded recovery, and setting the ailerons against the spin improved recovery.

Recoveries effected by simultaneous reversal of rudder and elevator were faster than the recoveries effected by reversal of rudder alone. Recoveries attempted by neutralizing the rudder were slow and unsatisfactory.

The effect of opening the cockpit is presented in table 2 (steady spins) and chart II (recoveries). The effect on both the steady spin and recovery characteristics of the model was small and inconsistent.

The effect of lowering the flaps (and drooping the ailerons) is shown in chart III. With the flap setting  $60^{\circ}$ , the landing gear retracted, the elevators full up, and with the ailerons set with the spin (right aileron up  $15^{\circ}$ , left aileron down  $25^{\circ}$ ), the model appeared to recover of its own accord when launched in a spin. With the ailerons set against the spin, however, the model would continue to spin even after the rudder was reversed in an attempt at recovery.

Neutral and down settings of the elevator with aileron settings neutral or against the spin produced fast, flat spins from which recovery was either uncertain or unsatisfactory. Setting the ailerons with the spin produced steeper spins from which recovery was effected more rapidly. In general, the effect of lowering the landing gear was small except when the ailerons were set against the spin. With the ailerons set against

the spin, the spins with the landing gear down were steeper than the corresponding spins with the landing gear up, and recovery was effected more readily. With the flaps down only  $40^{\circ}$  (using the drooped aileron settings corresponding to  $60^{\circ}$  flap deflection), the spins were generally steeper and the recoveries faster than the corresponding spins with  $60^{\circ}$  flaps, for the cases with ailerons against the spin.

In many instances where flap deflections are accompanied by a drooping of the ailerons, the detrimental effect of the combination on the recovery characteristics of the model can be attributed largely to the drooping ailerons. This is not true for the XF4U-1 model, however, as may be seen from chart IV, which clearly illustrates the adverse effect of the flap alone.

To afford a ready comparison, the effect of the flaps and other landing combinations on the recovery characteristics of the model have been summarized in chart II. It will be observed that with the flaps up the extension of the landing gear had little effect on recovery except with the ailerons set with the spin, in which disposition the landing gear had an adverse effect. In the normal landing condition, recovery was generally unsatisfactory.

The effect on recovery of various changes in loading is given in table 3. The loading associated with the slowest recoveries (and hereafter referred to as the "worst load") was obtained by extending weights along the fuselage (increasing  $I_y$  and  $I_z$  by 30 percent). This loading was investigated more thoroughly than the others and the results of the investigations are presented in chart V.

In general, the effect of the worst load was to flatten the spin, decrease the rate of descent, decrease the angular velocity, and to retard recovery. The effects were greatest with down settings of the elevator and with the ailerons set against the spin, although satisfactory recoveries were not obtained for any control disposition. The model in this loading condition would spin with the rudder neutral unless the elevator was held up. Recoveries were slow and unsatisfactory.

A comparison of left and right spins may be obtained by comparing chart VI with chart I. In general, there was little difference in either steady spins or recoveries, except with the ailerons held against the spin, with which control disposition the left spins were definitely steeper, the air speed being so high that test results could not be obtained. The difference may be attributed, qualitatively, to the effect of the fin offset.

Attempts were made to obtain inverted spins of the model in the normal loading condition but it would not spin inverted for any control disposition.

Since unsatisfactory recovery was obtained for many control settings, it was apparent that a structural modification of the airplane would be necessary. In the initial attempts to improve the recovery characteristics of the model, the following tail modifications were tried: (1) additional area in front of the vertical fin; (2) additional area on the rear upper portion of the rudder; and, (3) additional area below the horizontal tail surfaces. The optimum location for additional area appeared to be below the horizontal tail: Figure 2 shows the size and location of one of the most effective additional fins tried. Its effect on the recovery characteristics of the model is tabulated in table 4. The addition of this area (in conjunction with the tail wheel) effected satisfactory recoveries from spins from which recovery was previously uncertain or unsatisfactory.

After the completion of these tests the additional fin was removed and the original (larger) vertical tail was replaced by the modified vertical tail shown on Chance Vought drawing CV-S-5593. A comparison of the results for the original and modified vertical tail surfaces is given in charts VII, VIII, and IX. In the

normal load, the modified (smaller) vertical tail gave only slightly slower recoveries than the original (larger) tail. In the worst loading condition and in the normal loading condition with flaps down, neither tail was very satisfactory although on the whole the original tail appeared to be the better. Increasing the rudder deflection from  $\pm 25^\circ$  to  $\pm 30^\circ$  was tried on the modified tail but the effect was small.

The effect of adding the additional area shown in figure 2 to the modified tail is shown in charts X and XI. With the auxiliary fin, satisfactory recoveries were obtained in both the normal and the worst loading condition.

Although the additional fin aft of the tail wheel gave marked improvement, this modification was not considered to be practicable on the airplane and other, more extensive, changes were investigated. These tests were made with the modified (smaller) tail.

Attempts were made to improve the recovery characteristics by changing the location of the horizontal and vertical tail surfaces. The first of these latter modifications consisted of a 10-inch (full-scale) elevation of the horizontal tail surfaces and the results of this investigation are presented in charts XII, XIII, and XIV.

When the elevator was full up it interfered with the rudder movement, so at this time no information was obtained for the elevator up spins. For neutral and down elevator settings, however, the raised horizontal tail effected a marked improvement for all the loading conditions and aileron dispositions tested. The slowest recoveries were obtained in the worst loading condition with the ailerons neutral and the elevators down (chart XII). (chart XIII) and in the normal loading condition. In the normal loading condition/with flaps deflected  $60^{\circ}$  (chart XIV), the model either recovered of its own accord when launched in a spin or spun so steeply that its rate of descent was too high for the tunnel. At this stage the model suffered considerable damage (fig. 2) in a crack-up; and, in the course of rebuilding, the usual constructional tolerances were somewhat exceeded.

The results obtained after the rebuilding of the model were not, in general, as consistent as those previously obtained.

Following this, the stabilizer was restored to its normal location and the original (larger) vertical tail successively moved 10 inches (full-scale) and 20 inches (full-scale) forward of the normal location. The results of these two vertical tail modifications are presented in charts XV, XVI, and XVII. While both the 10-inch and the 20-inch movements were superior to the

normal location, neither modification resulted in satisfactory recoveries from any but the normal loading condition. These results discouraged further research with forward movements of the vertical tail alone and recourse was again had to raising the stabilizer.

Charts XVIII, XIX, and XX show the effect of raising the stabilizer 5 inches with and without an accompanying 15-inch forward movement of the vertical tail. The modified (smaller) tail was used for these tests. The 5-inch stabilizer movement by itself resulted in only a minor improvement, but when used in conjunction with a 15-inch forward movement of the vertical tail a more noticeable improvement resulted. It should be noted, however, that with the flaps down  $60^{\circ}$ , the elevator full up, and the ailerons neutral, recovery by rudder reversal still was not effected. Placing the ailerons with the spin, with the elevator neutral, steepened the spins so greatly that the rate of descent was too high for the tunnel.

It was thought advisable to repeat and extend the tests with the stabilizer raised 10 inches in an attempt to secure results for the elevator up spins which were not obtained originally. This was done and these results are also presented in charts XVIII, XIX, and XX. Here a peculiar situation has presented itself, particularly as regards the flap results. When the



model was in the original condition, recovery was very unsatisfactory with the flaps down (chart III). Then, in the first series of tests with the stabilizer raised 10 inches (chart XIV), it was found that when the model was launched in a spin it gradually steepened and eventually recovered without movement of the controls. When these latter tests were repeated, however, after the damage to the model, it was found that the spins would persist and that recoveries were again unsatisfactory (chart XX). The steady spin appeared to be extremely critical so that slight alterations in the model (such as those incurred through reconstruction, etc.) exerted considerable influence on the type of spin established. It appears that, despite the relatively large unshielded rudder area, the rudder itself remains ineffective - possibly because of the turbulent flap wake, which is lower than usual on this model due to the wing arrangement.

Although the results shown in charts XVIII to XX are probably not as accurate as those previously obtained (charts XII to XIV), they again indicate that significant improvement is obtained by raising the horizontal tail surfaces 10 inches. The improvement thus obtained is more marked than the improvement due to raising the horizontal surfaces 5 inches and at the same time moving the vertical surfaces forward 15 inches.

### CONCLUSIONS

The model spin tests of the XF4U-1 indicate the following spinning characteristics:

1. In the high-speed condition with normal load, the airplane will spin normally. Recovery by rudder reversal alone will probably be satisfactory but not rapid. Rudder reversal followed by downward deflection of the elevator will produce faster recovery. Up settings of the elevator and aileron settings with the spin will retard recovery; down settings of the elevator and aileron settings against the spin will improve recovery.
2. In the landing condition (flaps and wheels down, ailerons drooped, and cockpit open) satisfactory recoveries by rudder reversal will be effected only if the elevator is held full up and the ailerons are held full with the spin prior to the recovery attempt.
3. Recovery from inverted spins will be rapid.
4. An increase of weight along the fuselage may retard recovery considerably. Other reasonable mass changes will have a lesser adverse effect.
5. The optimum location for additional fin area is directly below the horizontal tail surfaces.
6. A 15-inch (full-scale) forward movement of the vertical tail surfaces accompanied by a 5-inch (full-scale)

elevation of the horizontal tail surfaces will improve the recovery characteristics somewhat.

7. Considerable improvement in recovery characteristics will result if the horizontal tail surfaces are raised 10 inches (full-scale).

8. The modified (smaller) vertical tail surfaces will not be as effective as the original (larger) tail but should prove satisfactory if additional fin area is added below the horizontal tail and/or the horizontal tail surfaces are raised.

Langley Memorial Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., July 12, 1939.

  
Charles J. Donlan,  
Junior Aeronautical Engineer.

Approved:

  
Elton W. Miller,  
Principal Mechanical Engineer.

VB

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2. Seidman, Oscar, and Neihouse, A. I.: Comparison  
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## XF4U-1 RECOVERIES (RIGHT SPINS)

## EFFECT OF CONTROL SETTINGS

| MODEL CONDITION | Control Setting |             |             |                 |               |                 |               | Turns<br>for<br>Recovery |
|-----------------|-----------------|-------------|-------------|-----------------|---------------|-----------------|---------------|--------------------------|
|                 | FLAPS           | Ailerons    |             | Rudder          |               | Elevator        |               |                          |
|                 |                 | Rt.<br>Deg. | Lt.<br>Deg. | Initial<br>Deg. | Final<br>Deg. | Initial<br>Deg. | Final<br>Deg. |                          |
| NORMAL          | 0               | 20U         | 10D         | 25W             | 0             | 30U             | 30U           | No recovery in<br>6½     |
| "               | "               | 0           | 0           | 25W             | 0             | 30U             | 30U           | No recovery in<br>5¼, 5  |
| "               | "               | 10D         | 20U         | 25W             | 0             | 30U             | 30U           | 4, 3½                    |
| "               | "               | 20U         | 10D         | 25W             | 0             | 0               | 0             | No recovery<br>in 9      |
| "               | "               | 0           | 0           | 25W             | 0             | 0               | 0             | No recovery in<br>8½, 6¾ |
| "               | "               | 10D         | 20U         | 25W             | 0             | 0               | 0             | 4½, 2½                   |
| "               | "               | 20U         | 10D         | 25W             | 0             | 20D             | 20D           | No recovery<br>in 6      |
| "               | "               | 0           | 0           | 25W             | 0             | 20D             | 20D           | 4¾                       |
| "               | "               | 10D         | 20U         | 25W             | 0             | 20D             | 20D           | 2¼, 2¼                   |
| "               | "               | 20U         | 10D         | 25W             | 25A           | 30U             | 20D           | 2¼                       |
| "               | "               | 0           | 0           | 25W             | 25A           | 30U             | 20D           | 1½, 1                    |
| "               | "               | 10D         | 20U         | 25W             | 25A           | 30U             | 20D           | 1¼, ¾                    |
|                 |                 |             |             |                 |               |                 |               |                          |
|                 |                 |             |             |                 |               |                 |               |                          |

✓ In the Normal Loading Condition the landing gear was retracted, the tail wheel off, and the cockpit closed.

W=With; A=Against; U=Up; D=Down

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TABLE 2

VF-40-1 STEADY SPINS (LIGHT SPIN)

## EFFECT OF COCKPIT OPENING

| MODEL CONDITION | Control Setting |             |      |       | $\alpha$<br>Deg. | $\phi$<br>Deg. | V<br>ft./sec. | $\frac{C_b}{2V}$ |     |
|-----------------|-----------------|-------------|------|-------|------------------|----------------|---------------|------------------|-----|
|                 | Ailerons        |             | Rud. | Elev. |                  |                |               |                  |     |
|                 | Rt.<br>Deg.     | Lt.<br>Deg. | Deg. | Deg.  |                  |                |               |                  |     |
| NORMAL          |                 | 20U         | 10D  | 25W   | 20U              | 49             | +1            | 173              | .32 |
| "               | *               | 20U         | 10D  | 25W   | 20U              |                |               | ? $\checkmark$   |     |
| "               |                 | 0           | 0    | 25W   | 20U              | 51             | -1            | 173              | .35 |
| "               | *               | 0           | 0    | 25W   | 20U              | 50             | -1            | 177              | .33 |
| "               |                 | 20U         | 10D  | 25W   | 0                | 46             | 0             | 165              | .43 |
| "               | *               | 20U         | 10D  | 25W   | 0                | 45             | -1            | 175              | .40 |
| "               |                 | 0           | 0    | 25W   | 0                | 47             | -2            | 165              | .44 |
| "               | *               | 0           | 0    | 25W   | 0                | 47             | -2            | 173              | .42 |
| "               |                 | 20U         | 10D  | 25W   | 20D              | 41             | 0             | 167              | .46 |
| "               | *               | 20U         | 10D  | 25W   | 20D              | 44             | 0             | 167              | .45 |
| "               |                 | 0           | 0    | 25W   | 20D              | 43             | -2            | 161              | .47 |
| "               | *               | 0           | 0    | 25W   | 20D              | 47             | -2            | 169              | .43 |
| "               |                 | 10D         | 20U  | 25W   | 20D              | 40             | -2            | 169              | .47 |
| "               | *               | 10D         | 20U  | 25W   | 20D              |                |               | ? $\checkmark$   |     |
|                 |                 |             |      |       |                  |                |               |                  |     |

✓ In the Normal Loading Condition the landing gear was retracted, the tail wheel off, and the cockpit closed.

\* Cockpit open

✓ Vertical velocity too high to test.

W= With ; A= Against ; U= Up ; D= Down

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TABLE

XF4U-1 RECOVERIES (RIGHT SPIN)

## EFFECT OF CHANGES IN MASS DISTRIBUTION

| MODEL CONDITION                                                            | Control Settings |          |              |            |              |            | Turns for Recovery                |
|----------------------------------------------------------------------------|------------------|----------|--------------|------------|--------------|------------|-----------------------------------|
|                                                                            | Ailerons         | Rudder   | Elevator     | Initial    | Final        | Initial    | Final                             |
|                                                                            | RT. Deg.         | LT. Deg. | Initial Deg. | Final Deg. | Initial Deg. | Final Deg. |                                   |
| Normal *                                                                   | 0                | 0        | 25W          | 25A        | 0            | 0          | 2 $\frac{1}{2}$ , 2               |
| "                                                                          | 0                | 0        | 25W          | 25A        | 20D          | 20D        | 1 $\frac{1}{4}$ , 1 $\frac{1}{4}$ |
| Wt. extended a/c. fuselage<br>I <sub>x</sub> and I <sub>y</sub> increased  | 0                | 0        | 25W          | 25A        | 0            | 0          | 3 $\frac{3}{4}$                   |
| "                                                                          | 0                | 0        | 25W          | 25A        | 0            | 0          | 3 $\frac{3}{4}$                   |
| Wt. extended a/c. fuselage<br>I <sub>x</sub> and I <sub>y</sub> increased  | 0                | 0        | 25W          | 25A        | 0            | 0          | 4 $\frac{3}{4}$ , 4               |
| "                                                                          | 0                | 0        | 25W          | 25A        | 0            | 0          | 5, 5                              |
| Wt. extended a/c. fuselage<br>I <sub>x</sub> and I <sub>y</sub> increased  | 0                | 0        | 25W          | 25A        | 0            | 0          | 2 $\frac{1}{2}$ , 2 $\frac{1}{2}$ |
| "                                                                          | 0                | 0        | 25W          | 25A        | 0            | 0          | 2 $\frac{1}{2}$                   |
| Wt. retracted a/c. fuselage<br>I <sub>x</sub> and I <sub>y</sub> decreased | 0                | 0        | 25W          | 25A        | 0            | 0          | 3, 2 $\frac{3}{4}$                |
| "                                                                          | 0                | 0        | 25W          | 25A        | 20D          | 20D        | 3 $\frac{1}{4}$ , 3               |
| C.G. 2" 4" 1/2 A.A.C.                                                      | 0                | 0        | 25W          | 25A        | 0            | 0          | 3 $\frac{1}{4}$ , 2 $\frac{3}{4}$ |
| "                                                                          | 0                | 0        | 25W          | 25A        | 20D          | 20D        | 3                                 |
| Wt. -6W and 4" A.A.C.                                                      | 0                | 0        | 25W          | 25A        | 0            | 0          | 2 $\frac{3}{4}$ , 2 $\frac{3}{4}$ |
| "                                                                          | 0                | 0        | 25W          | 25A        | 0            | 0          | 2 $\frac{3}{4}$ , 2 $\frac{3}{4}$ |

\* In the Normal loading condition, the tail wheel was retracted, the tail wheel was not deflated and the tail wheel was not used.

V Worst Loading Condition

W=With, A=Against, D=Down

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TABLE

4

XF4U-1

RECOVERIES (RIGHT SPINS)

## EFFECT OF ADDED FIN BELOW HORIZONTAL TAIL SURFACES

| MODEL CONDITION<br>(ORIGINAL VERTICAL TAIL) | Control Setting |             |             |                 |               |                 |               | Turns<br>for<br>Recovery                      |
|---------------------------------------------|-----------------|-------------|-------------|-----------------|---------------|-----------------|---------------|-----------------------------------------------|
|                                             | FLAPS           | Ailerons    |             | Rudder          |               | Elevator        |               |                                               |
|                                             |                 | Rt.<br>Deg. | Lt.<br>Deg. | Initial<br>Deg. | Final<br>Deg. | Initial<br>Deg. | Final<br>Deg. |                                               |
| NORMAL                                      | 0               | 0           | 0           | 25W             | 25A           | 20D             | 20D           | 1 <sup>1</sup> / <sub>4</sub>                 |
| NORMAL *                                    | 0               | 0           | 0           | 25W             | 25A           | 20D             | 20D           | Would not spin                                |
| NORMAL                                      | 0               | 0           | 0           | 25W             | 25A           | 0               | 0             | 2 <sup>1</sup> / <sub>2</sub>                 |
| NORMAL *                                    | 0               | 0           | 0           | 25W             | 25A           | 0               | 0             | would not spin                                |
| NORMAL                                      | 60°             | 10D         | 10D         | 25W             | 25A           | 20D             | 20D           | No recovery in 10 <sup>1</sup> / <sub>2</sub> |
| NORMAL *                                    | 60°             | 10D         | 10D         | 25W             | 25A           | 20D             | 20D           | 2 <sup>1</sup> / <sub>4</sub>                 |
| WORST LOAD                                  | 0               | 20U         | 10D         | 25W             | 25A           | 0               | 0             | 4 <sup>1</sup> / <sub>4</sub>                 |
| WORST LOAD *                                | 0               | 20U         | 10D         | 25W             | 25A           | 0               | 0             | <sup>3</sup> / <sub>4</sub>                   |
| WORST LOAD                                  | 0               | 0           | 0           | 25W             | 25A           | 0               | 0             | 4 <sup>3</sup> / <sub>4</sub>                 |
| WORST LOAD *                                | 0               | 0           | 0           | 25W             | 25A           | 0               | 0             | 2 <sup>1</sup> / <sub>4</sub>                 |
| WORST LOAD                                  | 0               | 10D         | 20U         | 25W             | 25A           | 0               | 0             | 5 <sup>1</sup> / <sub>4</sub>                 |
| WORST LOAD *                                | 0               | 10D         | 20U         | 25W             | 25A           | 0               | 0             | vertical velocity too high                    |
|                                             |                 |             |             |                 |               |                 |               |                                               |
|                                             |                 |             |             |                 |               |                 |               |                                               |
|                                             |                 |             |             |                 |               |                 |               |                                               |

$\checkmark$  In the Normal Loading Condition the landing gear was retracted, the tail wheel off, and the cockpit closed.

\* Area added below horizontal tail and tail wheel in position

$\checkmark$  Ailerons droop 10° when flaps are lowered.  $\checkmark$  Cockpit open

$\checkmark$  Wts. extended along fuselage (I<sub>r</sub> and I<sub>z</sub> increased 30° of I<sub>r</sub>)

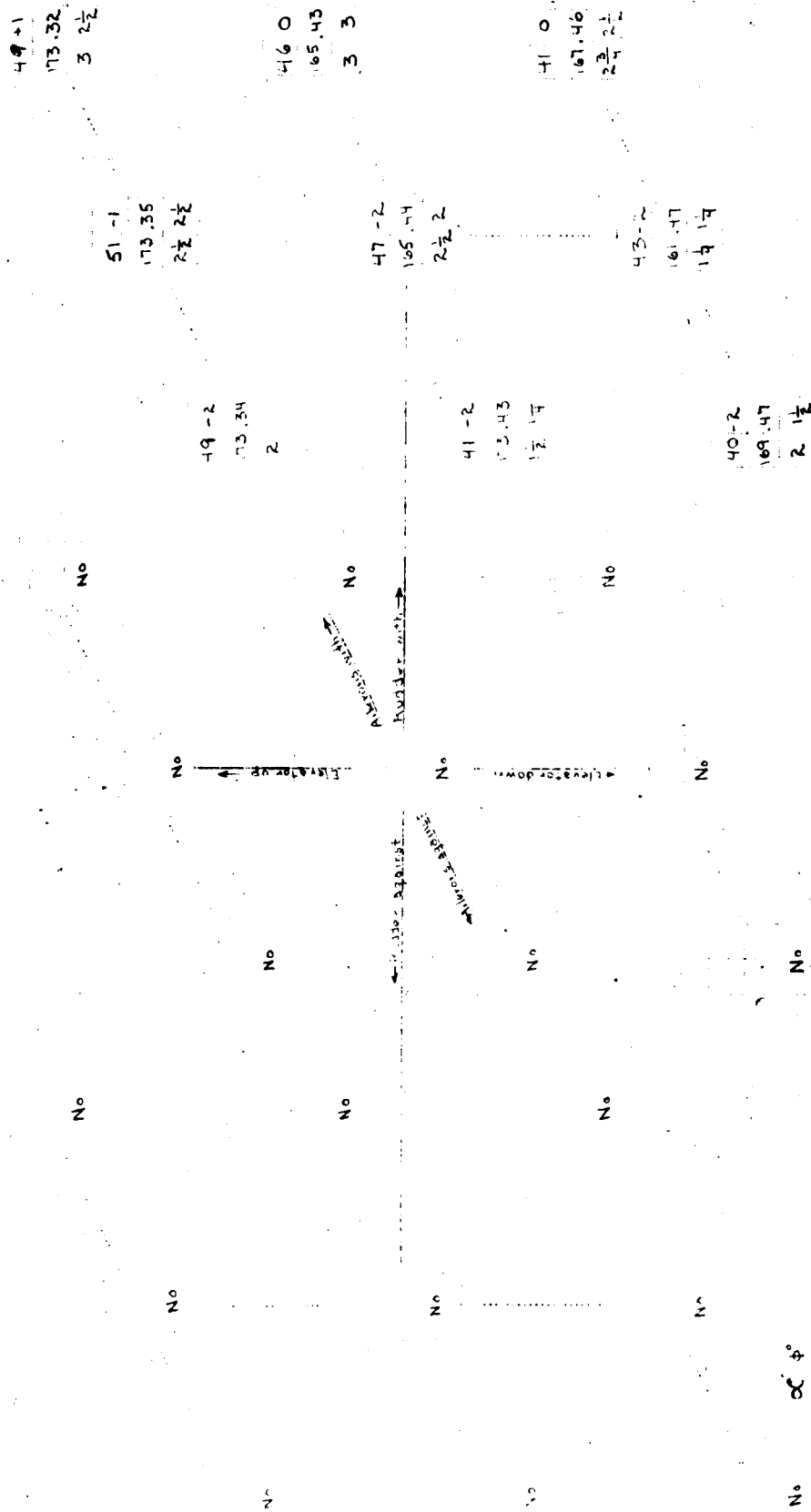
W=With; A=Against; U=Up; D=Down

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MODEL XF4U-1 CONTROL EFFECT OF SETTING \_\_\_\_\_ LOADING \_\_\_\_\_ NORMAL \_\_\_\_\_ SPIN CHARACTERISTICS (CHART I) \_\_\_\_\_  
RECOVERY BY RUDDER REVERSAL ALONE (From 10 to 20 degrees displacement) \_\_\_\_\_ COCKPIT CLOSED \_\_\_\_\_ LANDING GEAR \_\_\_\_\_ RETRACTED \_\_\_\_\_ FLAP SETTING \_\_\_\_\_ NEUTRAL \_\_\_\_\_



"NO" MEANS MODEL WOULD NOT SPIN  
"Y" MEANS MODEL WOULD SPIN BUT

MEANS MODH WOULD SPIN BUT NO DATA OBTAINED FOR REASONS NOTED

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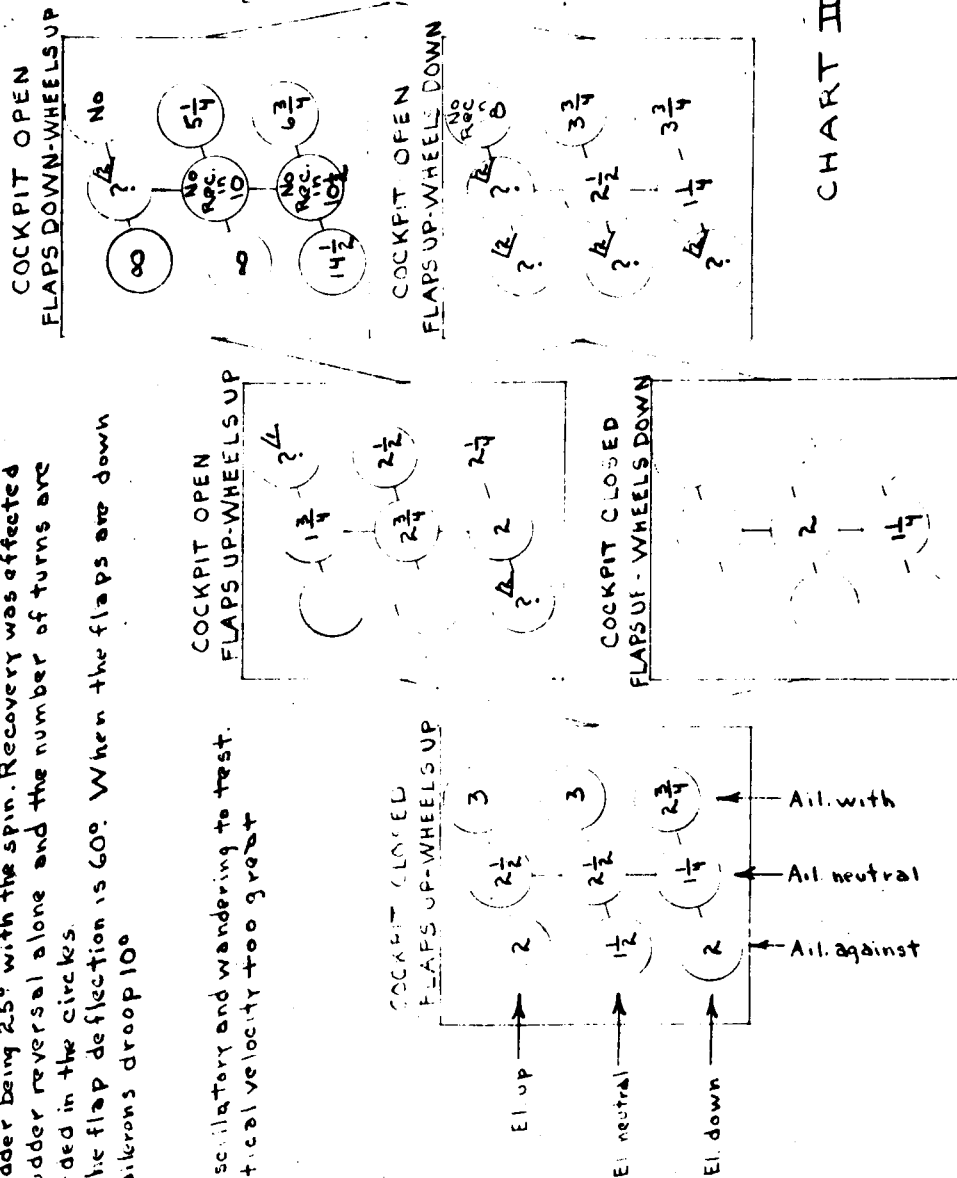
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# XF4U-1 MODEL - ORIGINAL (LARGER) TAIL LANDING COMBINATIONS NORMAL LOAD

Chart shows control disposition prior to recovery attempt, the rudder being 25° with the spin. Recovery was affected by rudder reversal alone and the number of turns are recorded in the circles.

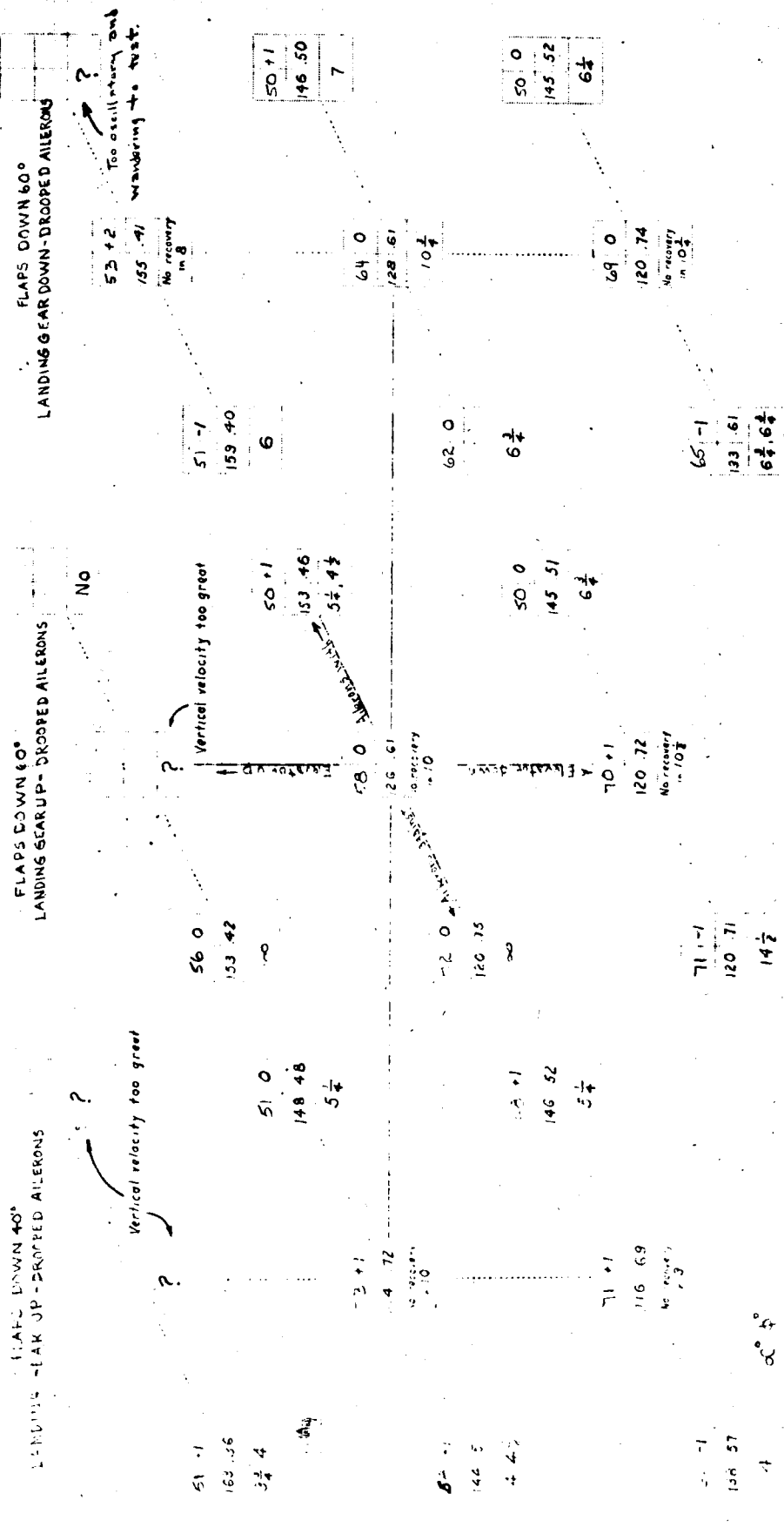
The flap deflection is 60°. When the flaps are down the ailerons droop 10°

- ∠ Too oscillatory and wandering to test.
- ∠ Vertical velocity too great



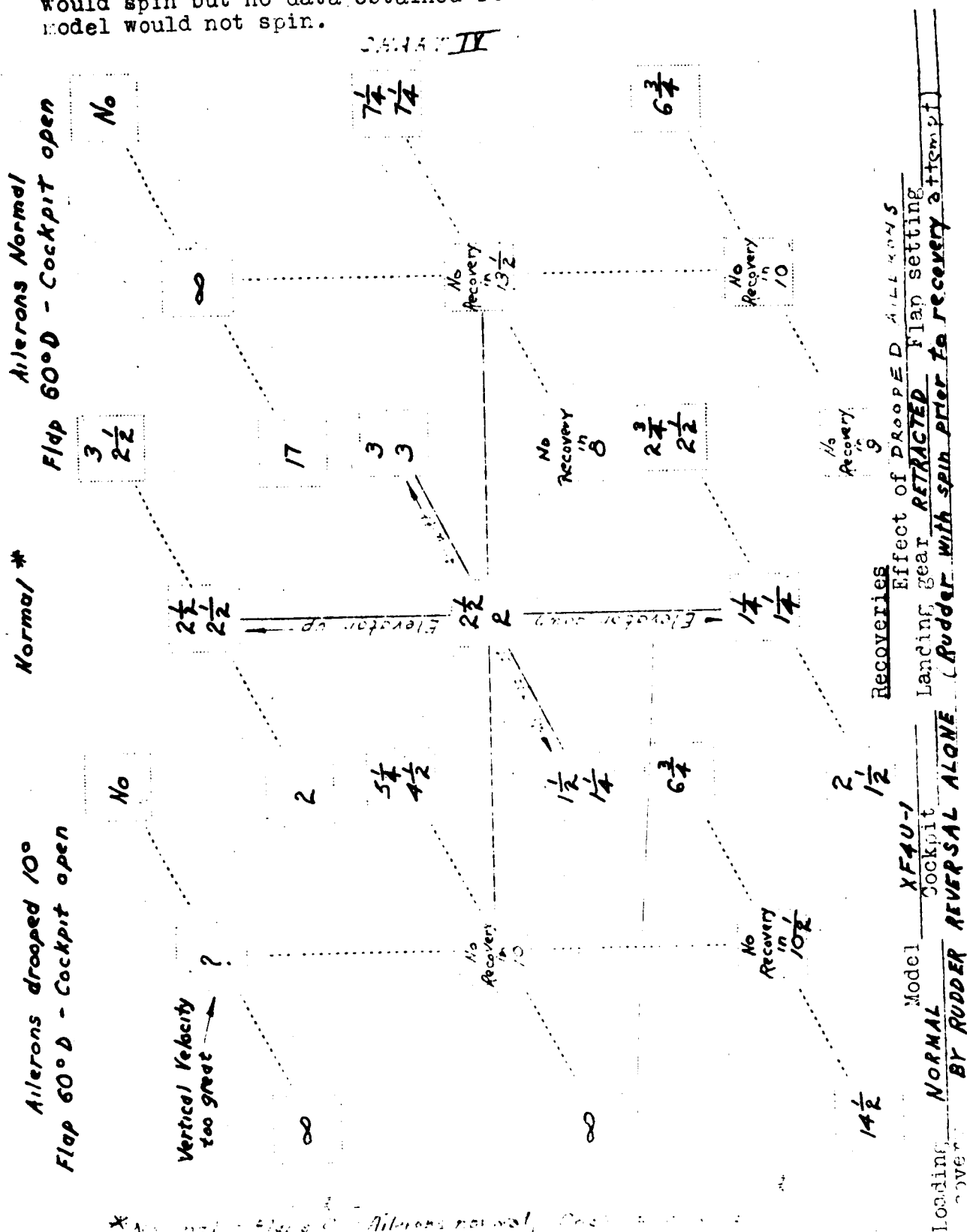
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SPIN CHARACTERISTICS CHART III  
 COCKPIT OPEN LANDING GEAR FLAP SETTING 40° AND 60° DOWN



NOT MEANS MODEL WOULD NOT SPIN  
 MEANS MODEL WOULD SPIN BUT NO DATA OBTAINED FOR REASONS NOTED

Note: Chart shows control disposition prior to recovery attempt.  
 Figures in squares are turns for recovery. "?" means model  
 would spin but no data obtained for reasons noted. "No" means  
 model would not spin.



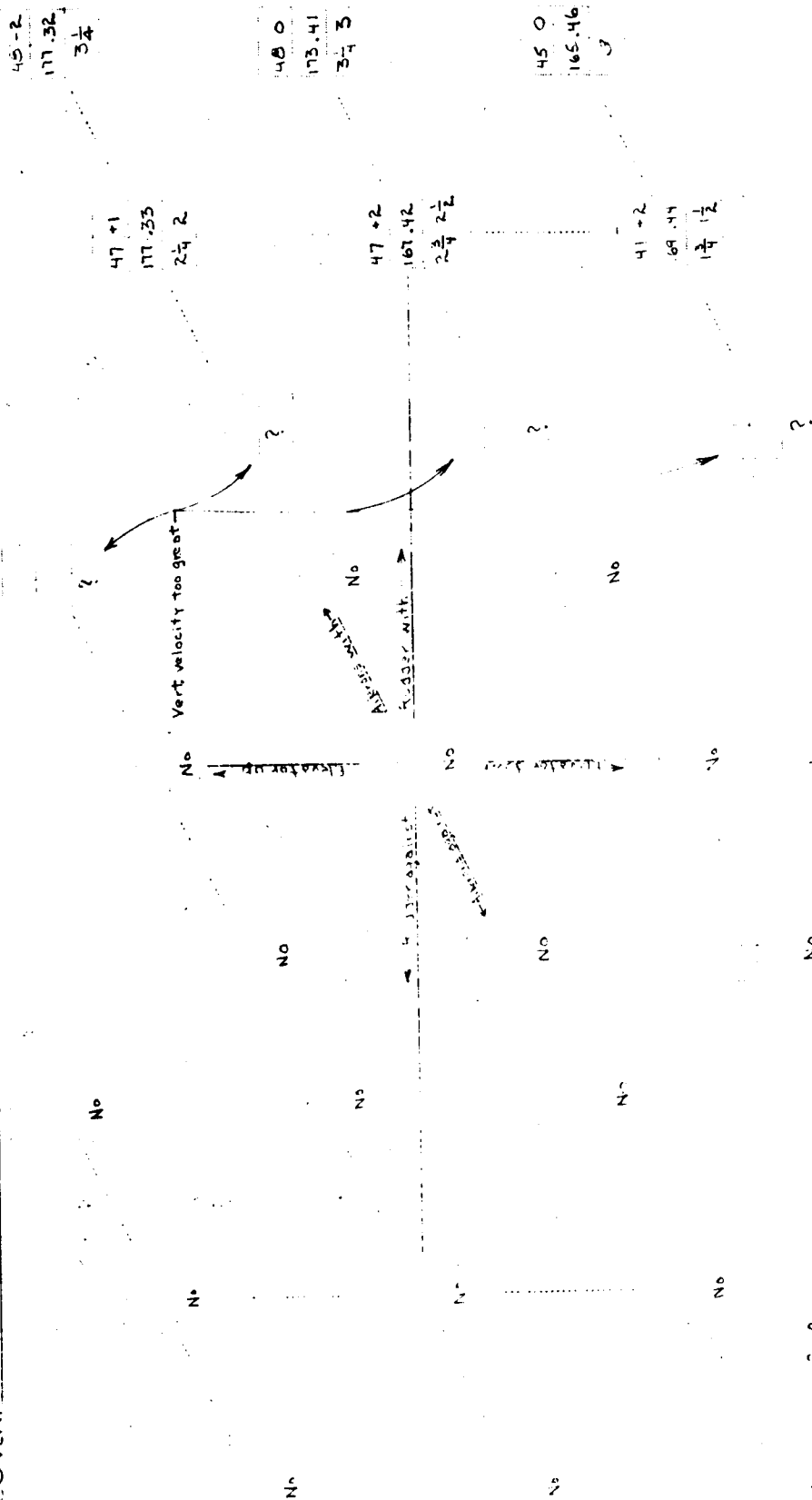
\* No spin - Ailerons normal, Cockpit open



# SPIN CHARACTERISTICS CHART VII

MODEL FAULT EFFECT OF LEFT SPINS LOADING NORMAL COCKPIT CLOSED LANDING GEAR RETRACTED FLAP SETTING NEUTRAL

RECOVERY BY RUDDER REVERSAL ALONE (From 117.3 control distance as indicated)



NO MEANS MODEL WOULD NOT SPIN  
MEANS MODEL WOULD SPIN BUT NO DATA OBTAINED FOR REASONS NOTED

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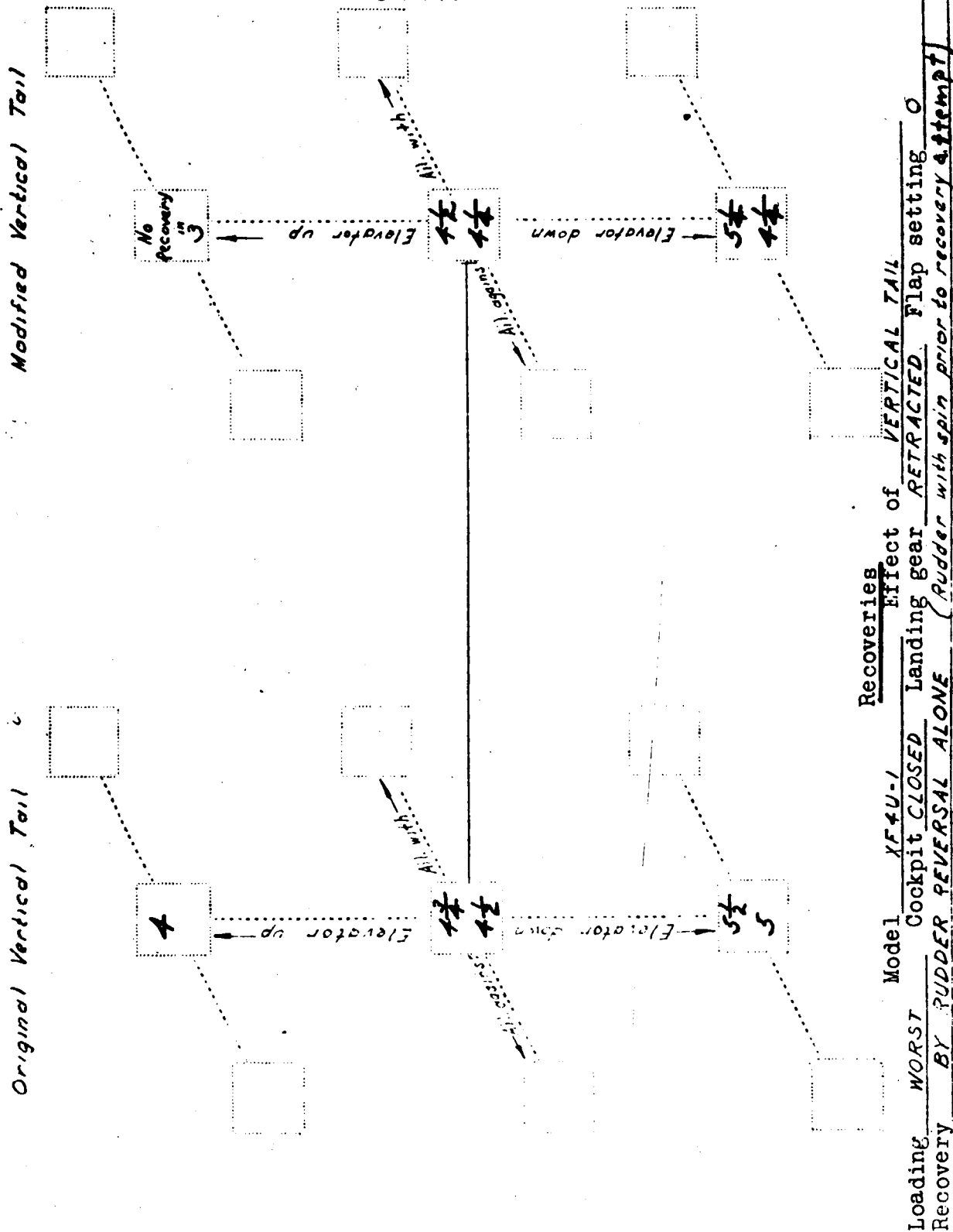
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## CHART. VII



Note: Chart shows control disposition prior to recovery attempt.  
 Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART VIII

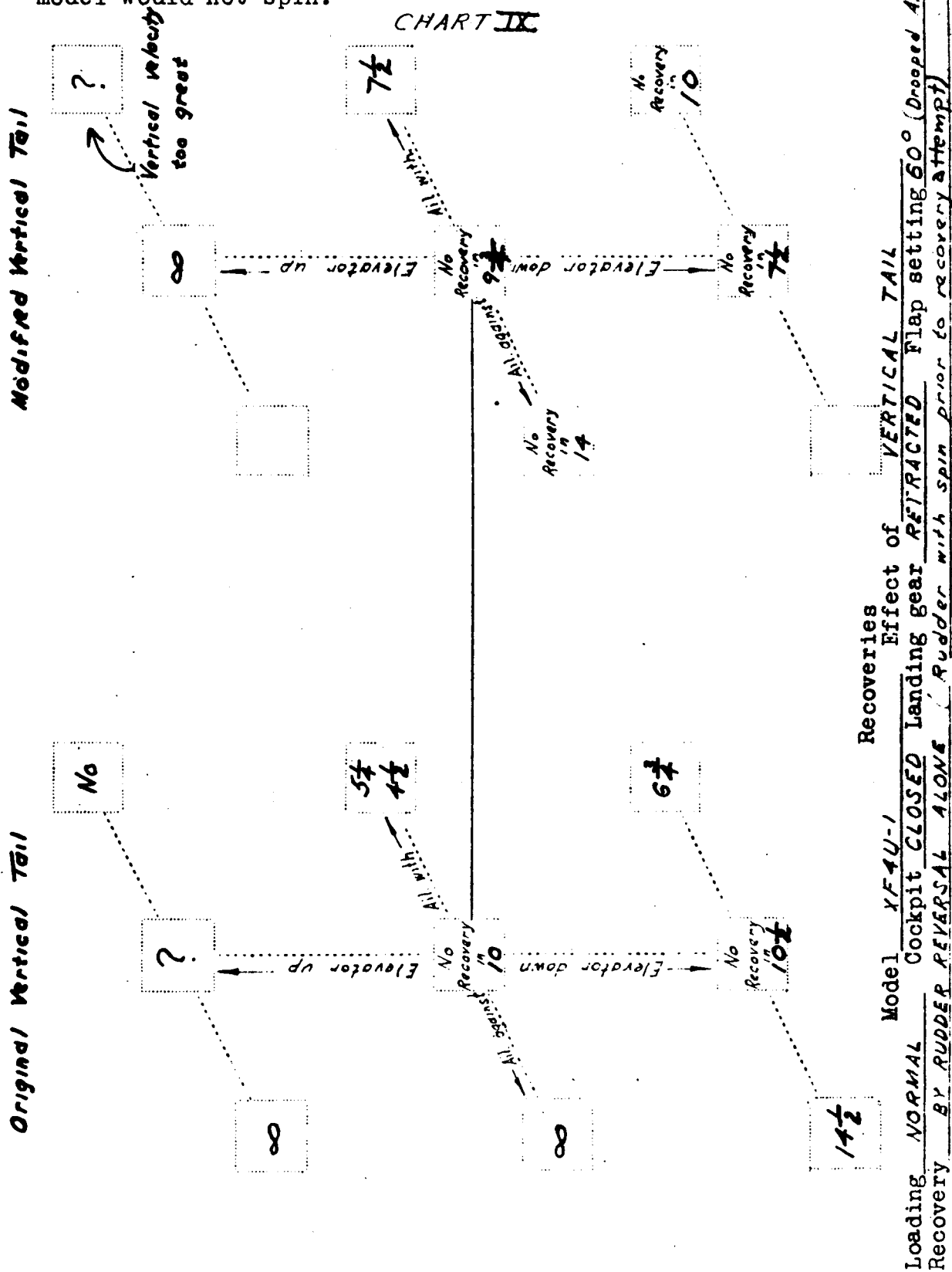


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Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART IX

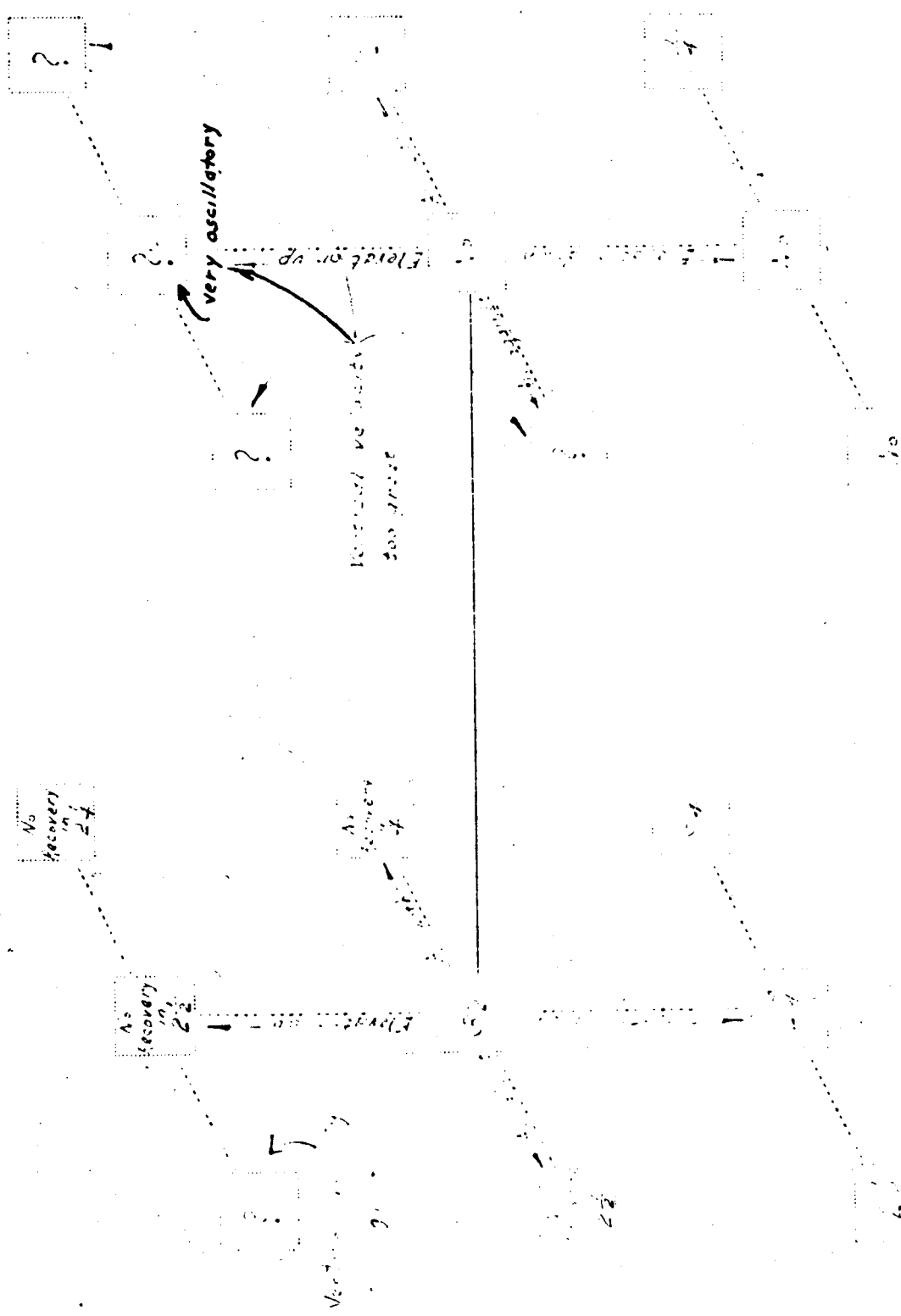


Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART X

Modified Vertical Tail with added fin area below horizontal tail

Modified Vertical Tail



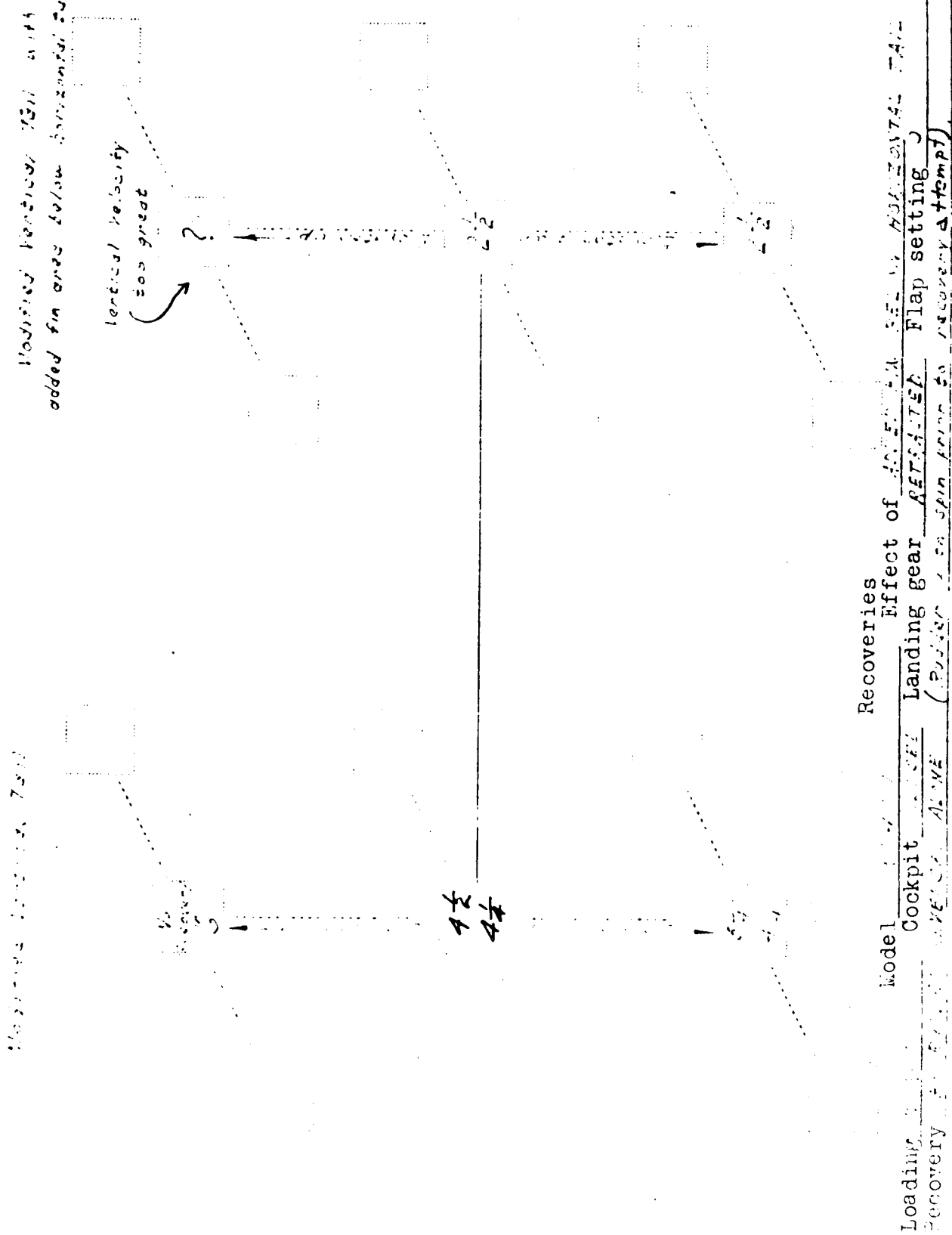
| Recoveries    |                                               |
|---------------|-----------------------------------------------|
| Model         | Effect of ADDITION BELOW HORIZONTAL TAIL      |
| Cockpit CHASE | Landing gear RETRACTED                        |
| Recovery      | Flap setting                                  |
|               | Recovery after spin prior to recovery attempt |

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Note: Chart shows control disposition prior to recovery attempt.  
 Figures in squares are turns for recovery. "?" means model  
 would spin but no data obtained for reasons noted. "No" means  
 model would not spin.

CHART XI

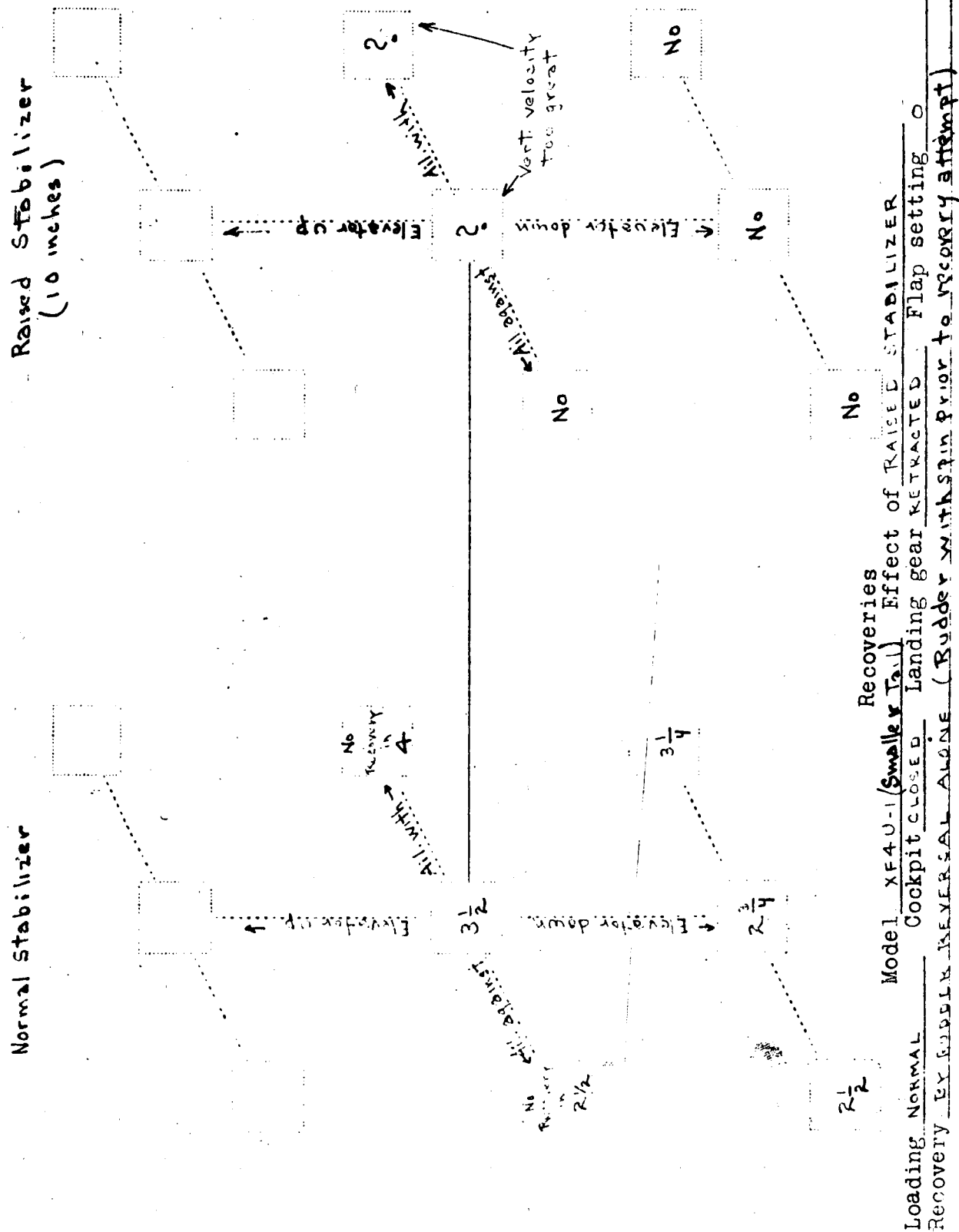


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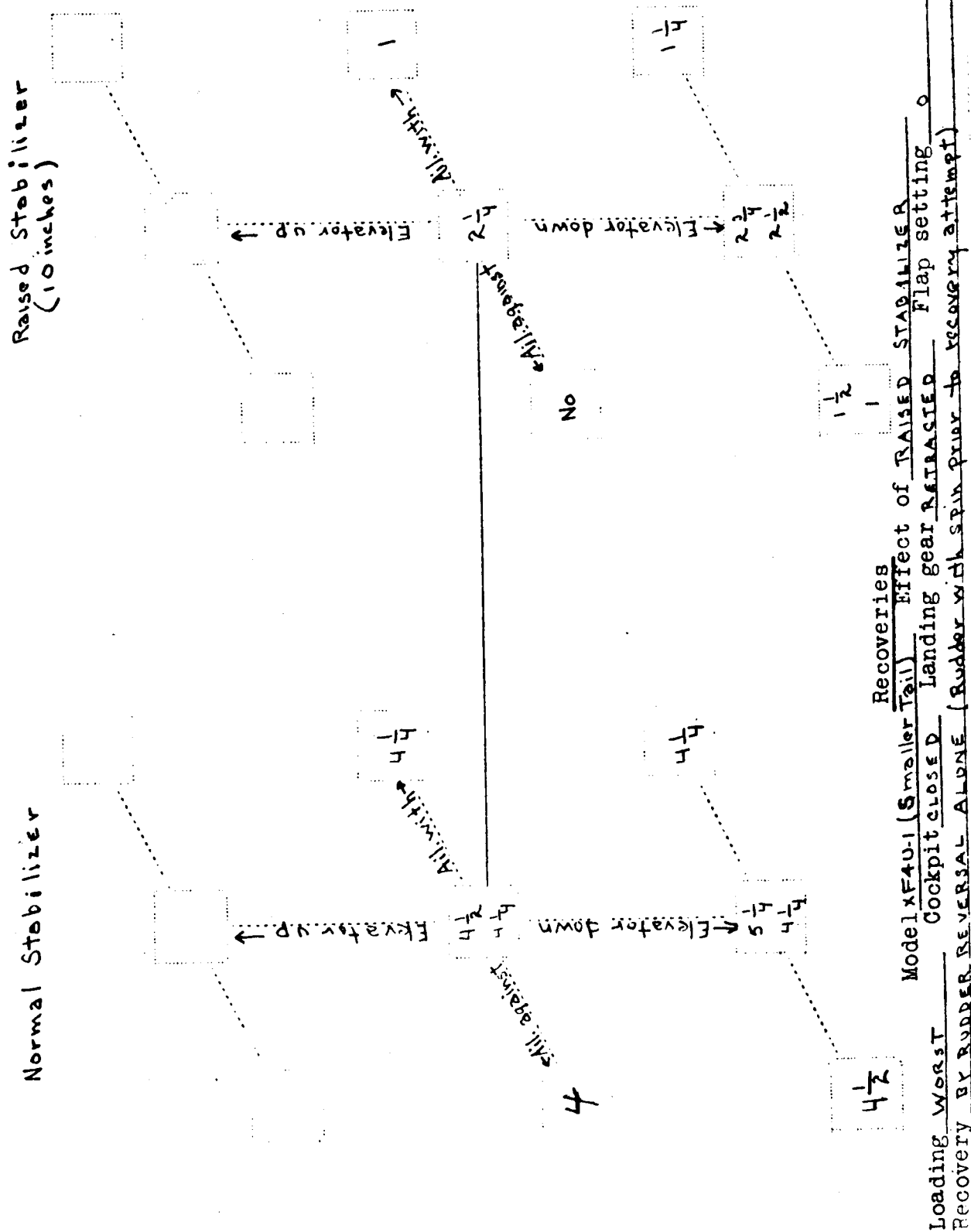
Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART ~~III~~



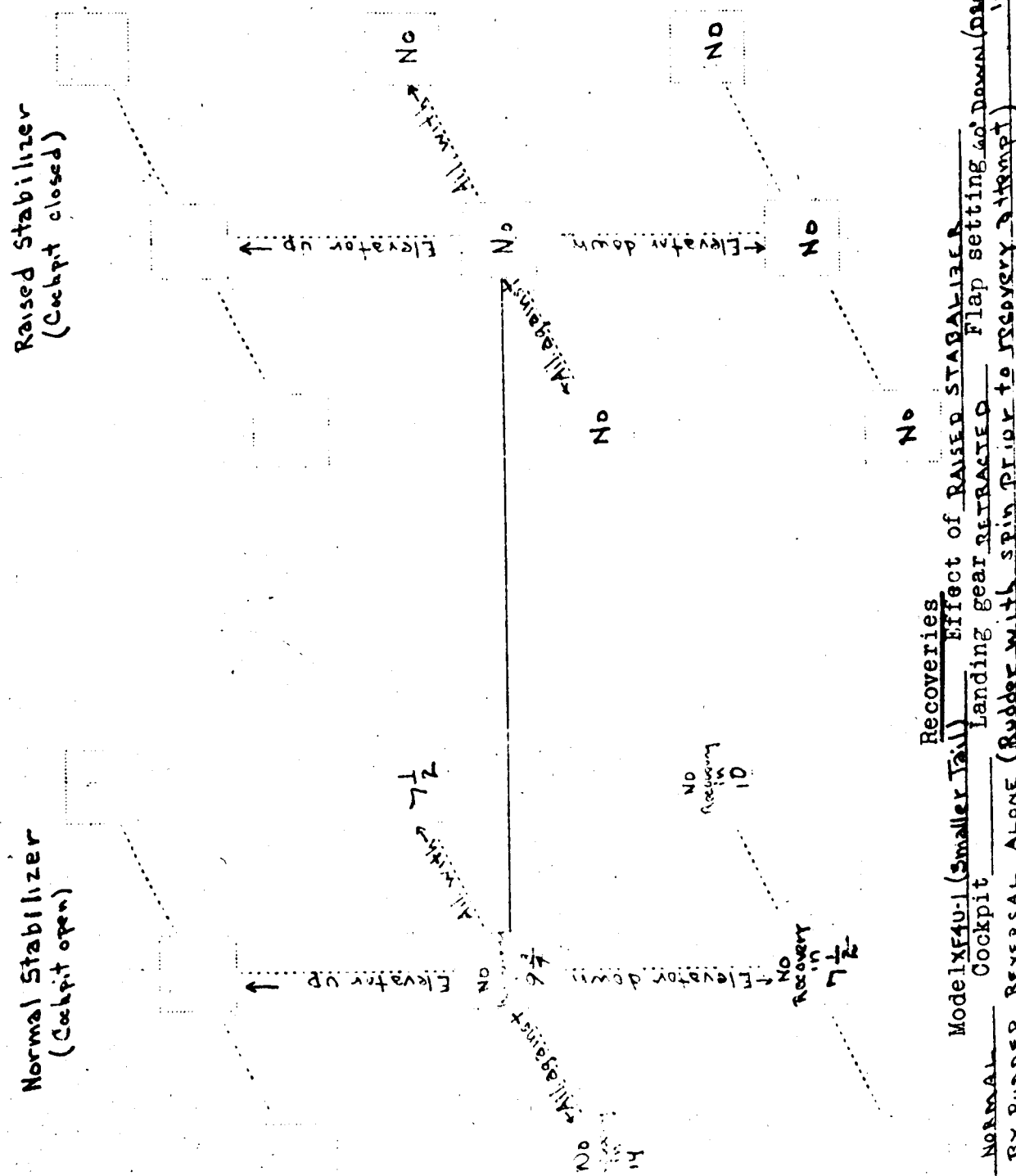
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Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin. CHART XIII



Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART XIV



Recoveries

Model XF4U-1 (Smaller Tail) Effect of RAISED STABILIZER  
 Loading NORMAL Cockpit Landing gear RETRACTED Flap setting 40° DOWN (DRAPED ALLENS) 10 1/2  
 Recovery BY RUDDER REVERSAL ALONE (Rudder with spin prior to recovery attempt) 10 1/2

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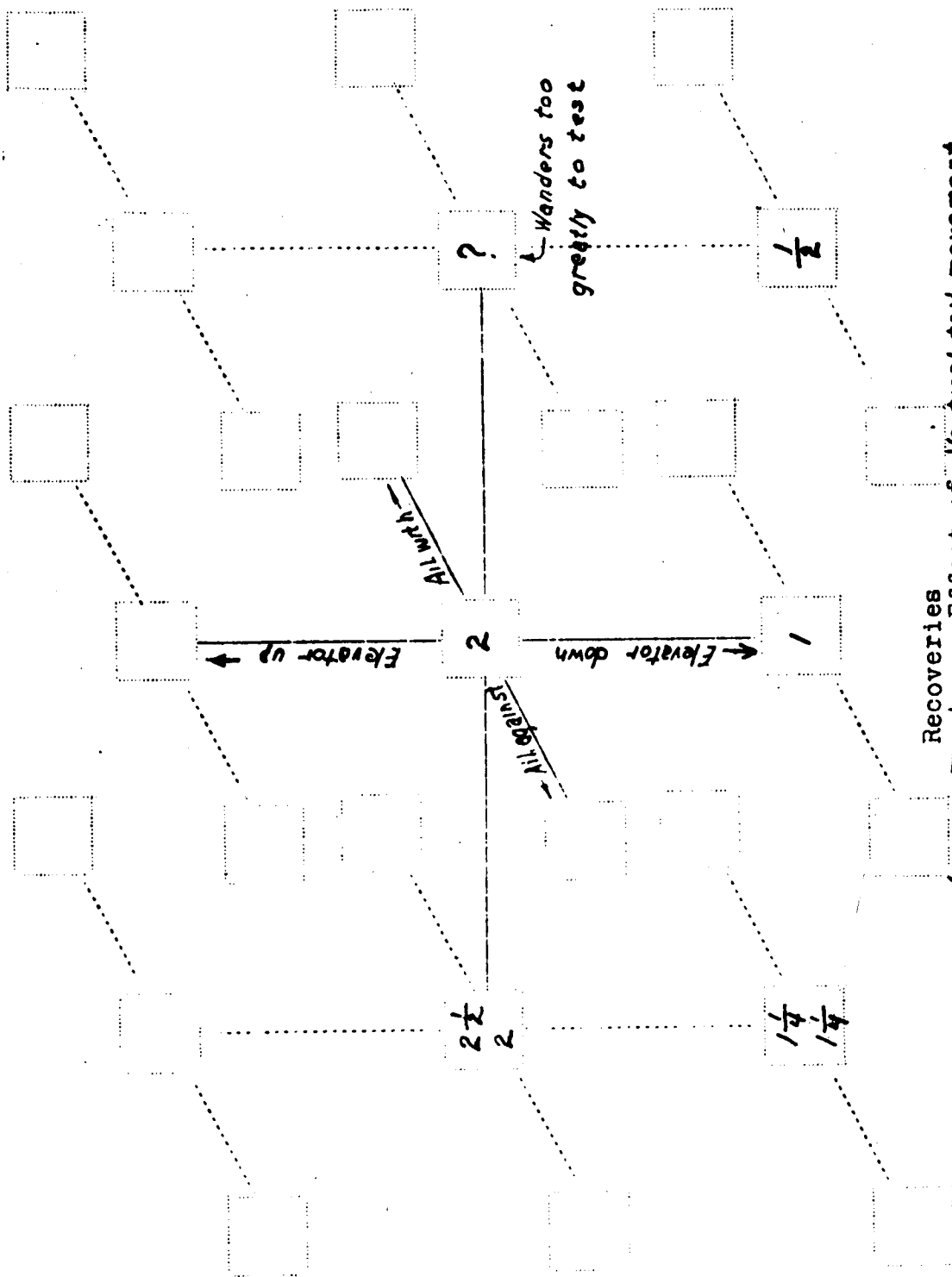
Note: Chart shows control disposition prior to recovery attempt.  
 Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART ~~VI~~

Vertical Tail  
20 inches forward

Vertical Tail  
10 inches forward

Normal location



Recoveries

Model XF4U-1 (Larger Tail) Effect of Vertical tail movement

Cockpit closed Landing gear Retracted Flap setting 0

Loading Normal

Recovery By rudder reversal alone. (Rudder with spin prior to recovery attempt)

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CONFIDENTIAL - Contains Data for Recovery

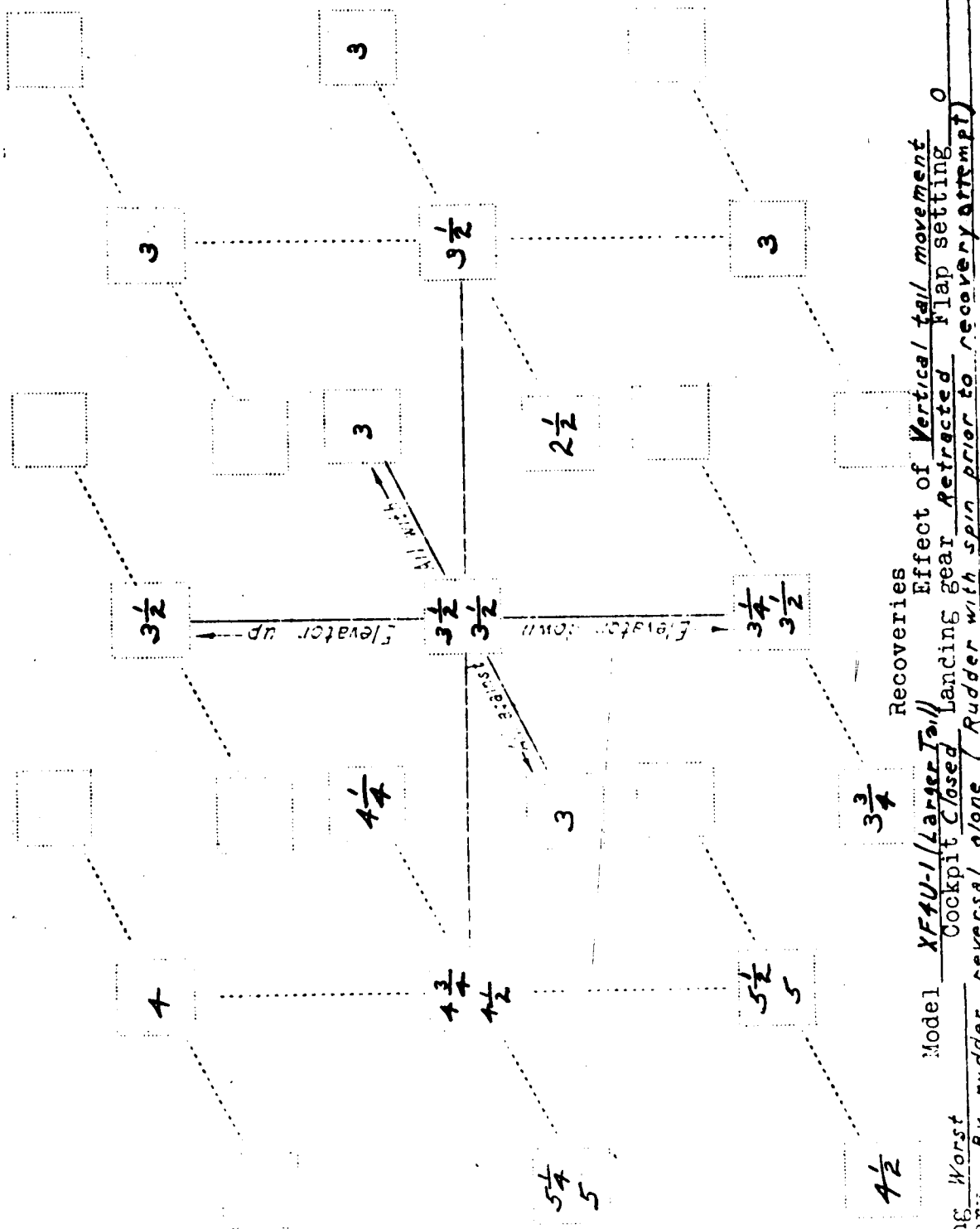
Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART XVI

Normal location

10 inches forward

20 inches forward



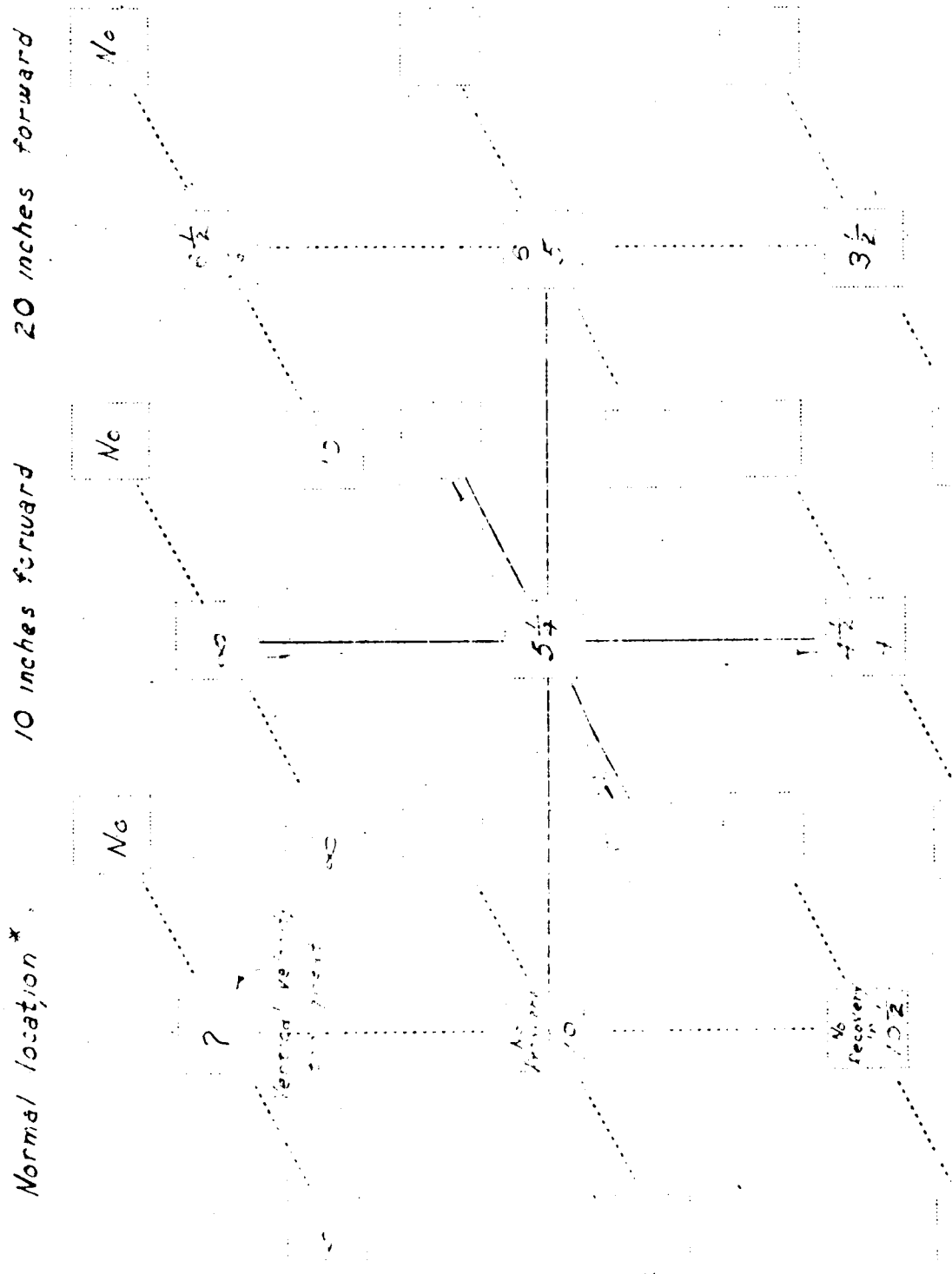
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Note: Chart shows control disposition prior to recovery attempt. Figures in **squares** are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

XVII



Recoveries

Model XF4U-1/Landing Gear Effect of Vertical tail movement  
 Loading Normal Cockpit Closed Landing gear Retracted Flap setting 60° Down (ail. dropped 10°)  
 Recovery: By rudder reversal alone Rudder With spin prior to recovery attempt

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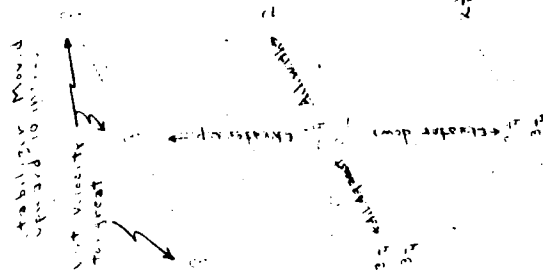
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## CHART XVIII



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CHART - 5

[illegible]

10

[illegible]

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## CHAPTER XX



Mode 1 (F4) (smaller Tail)  
cockpit closed Landing near Retracted Flap setting 60 (Aikens  
Recovered Effect of Tail Modification Flap setting 60 (Aikens  
Retracted)

cockpit closed. Landing gear retracted. Flap securing on jacking.  
 der reversal. Alone. (Kilber with main pump to recovery attempt).

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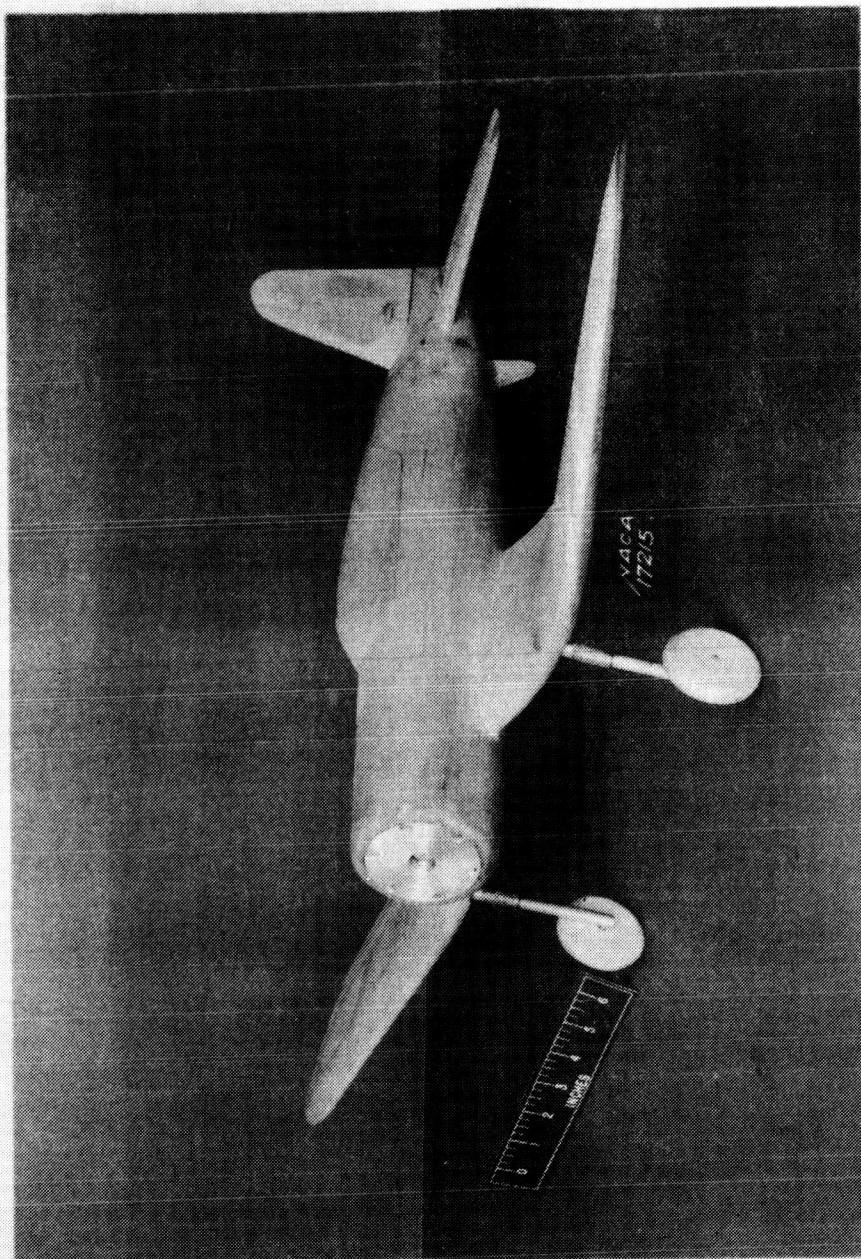
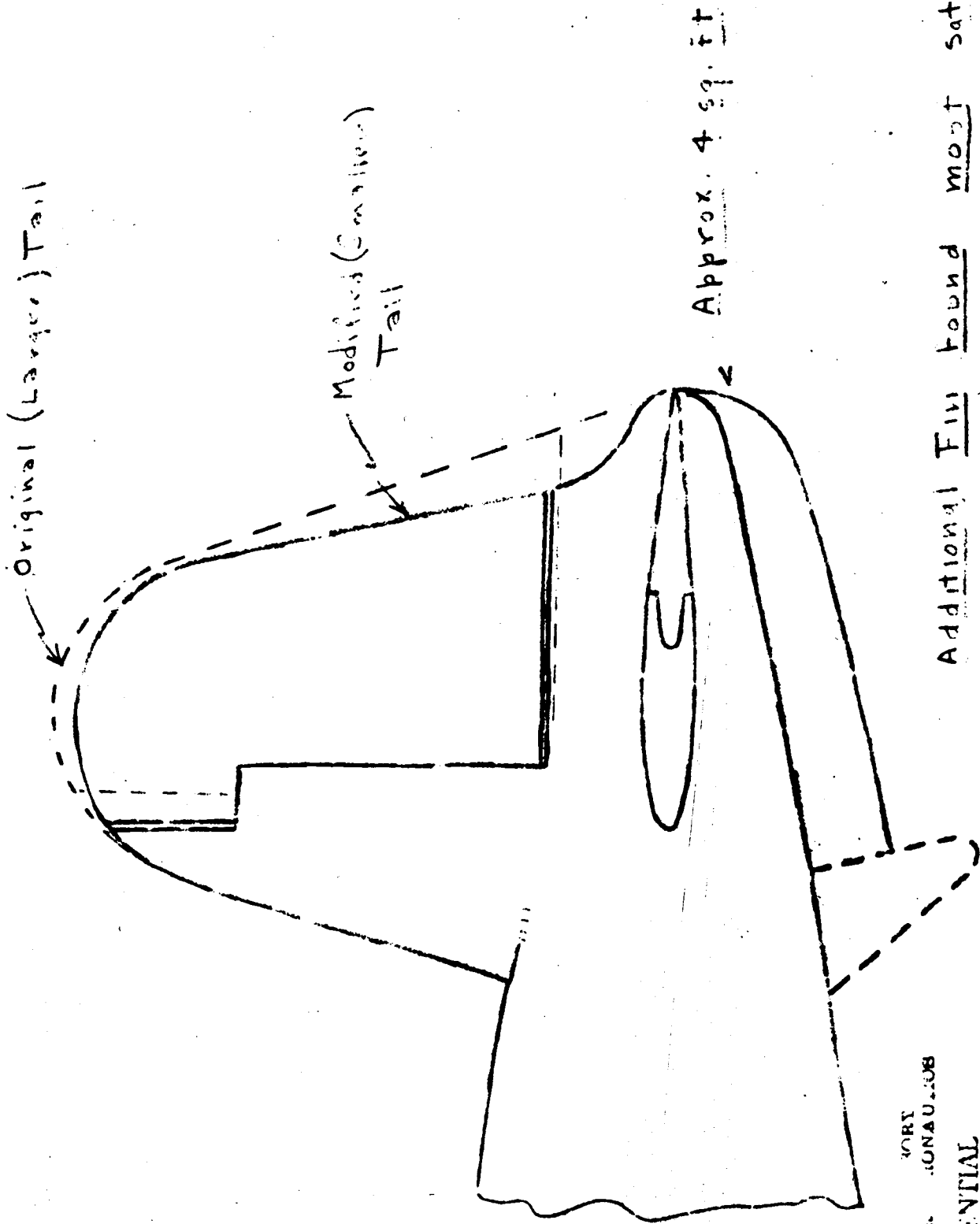


Figure 1. - 1/20-scale model of the XF4U-1 airplane.

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FIGURE 2



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AND SPACE ADMINISTRATION

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Additional Fin found most satisfactory  
for XF4D-1 Model



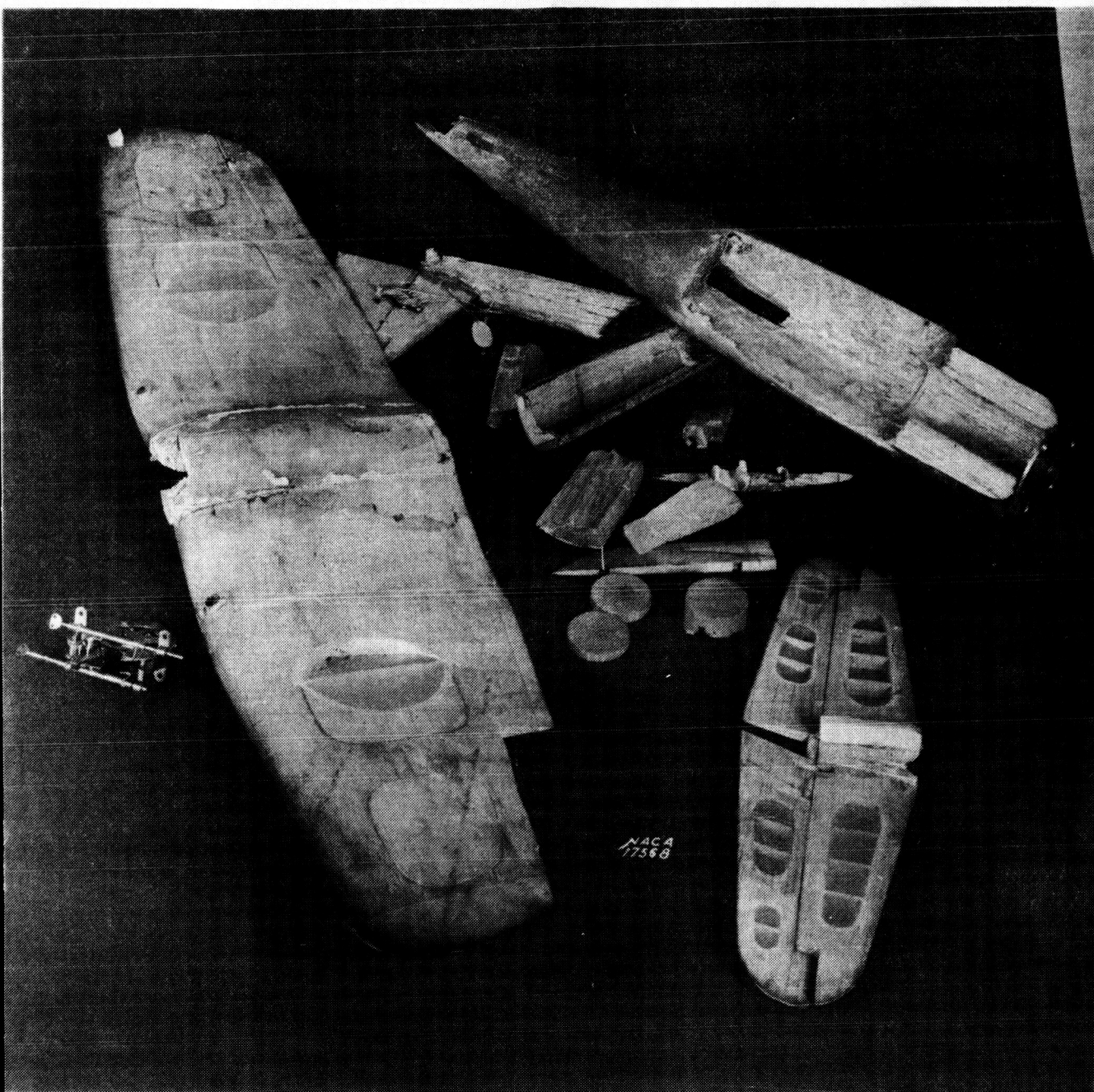


Figure 3. - Showing damage to XF4U-1 model after crack-up.

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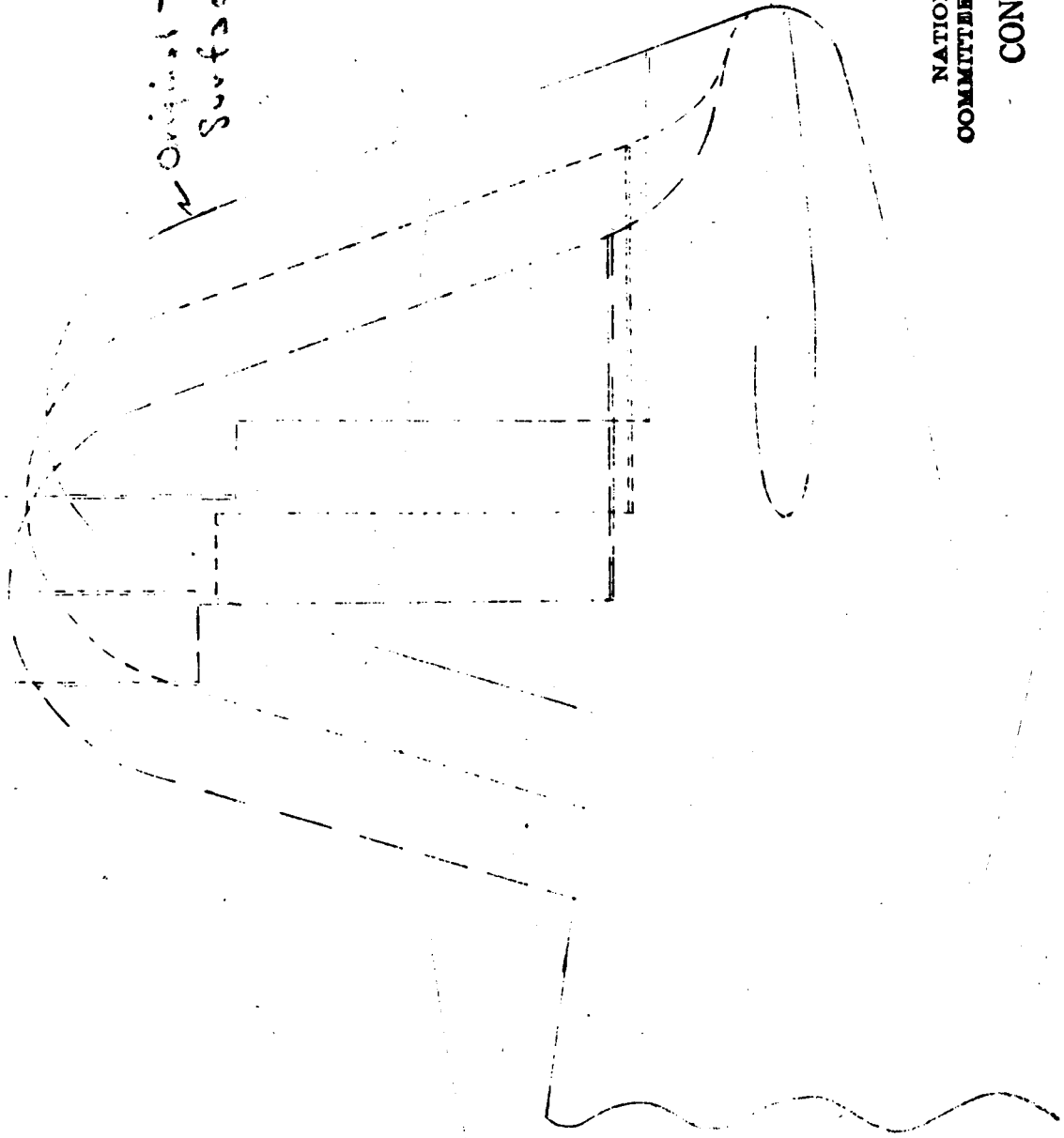
Figure 4

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VERTICAL TAIL MOVEMENTS

LOCATION

Original Tail  
Surfaces

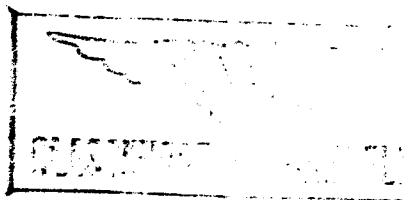




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CONFIDENTIAL MEMORANDUM REPORT

for

Bureau of Aeronautics,  
Navy Department

SPIN TESTS OF 1/16-SCALE MODELS OF THE  
N3N-3 LANDPLANE AND SEAPLANE

By CHARLES J. DONLAN

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Memorial Aeronautical  
Laboratory.*

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January 12, 1940

CONFIDENTIAL MEMORANDUM REPORT

for

Bureau of Aeronautics,  
Navy Department

SPIN TESTS OF 1/16-SCALE MODELS OF THE  
N3N-3 LANDPLANE AND SEAPLANE

By CHARLES J. DONLAN

INTRODUCTION

The investigation herein described was undertaken at the request of the Bureau of Aeronautics, Navy Department. The test program included the study of both the seaplane and landplane types. On both versions of the model the effects of loading changes and control dispositions were examined and on the seaplane the effect of the cowled and uncowed engine was investigated.

Preliminary results of the investigation have already been forwarded to the Bureau of Aeronautics.

APPARATUS AND MODEL

The tests were conducted in the N. A. C. A. free-spinning wind tunnel as described in reference 1.

The 1/16-scale model was built by the N. A. C. A. in accordance with drawing Nos. 68077, 67716, 67717, 67718, and 67719 of the Naval Aircraft Factory, Philadelphia. The full-scale weights, the center-of-gravity locations, the moments of inertia, and the length and location of the mean aerodynamic chord were

obtained from Report No. 2075 and drawings Nos. 67718 and 67719 of the Naval Aircraft Factory.

The model was constructed principally of balsa wood. A clockwork mechanism was installed to move the controls during recovery tests. Lead ballast, suitably located, served to bring the weights, the moments of inertia, and the center-of-gravity locations to their appropriate scaled-down values. The model was first tested as a seaplane, with one main float and two auxiliary wing-tip floats. For the landplane tests, the floats were removed and a fixed type landing gear installed.

The weight and the mass distribution of the model were held to the true scaled-down values within the following limits:

Weight - - - - -  $\pm 1$  percent.

Center-of-gravity locations  $\pm 1$  percent.

|                                                                                     | <u>Seaplane</u>                                                                                                                            | <u>Landplane</u>                                                                                                                            |
|-------------------------------------------------------------------------------------|--------------------------------------------------------------------------------------------------------------------------------------------|---------------------------------------------------------------------------------------------------------------------------------------------|
| Moments of Inertia $\left\{ \begin{array}{l} I_x \\ I_y \\ I_z \end{array} \right.$ | $\begin{array}{l} -3 \text{ to } 3 \text{ percent.} \\ -6 \text{ to } 0 \text{ percent.} \\ -1 \text{ to } 5 \text{ percent.} \end{array}$ | $\begin{array}{l} -4 \text{ to } 2 \text{ percent.} \\ -1 \text{ to } -2 \text{ percent.} \\ -5 \text{ to } 1 \text{ percent.} \end{array}$ |

Figures 1 and 2 show the N3N-3 model as a seaplane and landplane, respectively.

# TEST CONDITIONS

Maximum control displacements of  $\pm 30$  degrees for the rudder and 35 degrees up and 25 degrees down for the elevator were supplied by the Naval Aircraft

Factory. The relative stick and aileron movements were obtained from the drawing No. 67717 of the Naval Aircraft Factory.

The normal loading condition of the seaplane and landplane models corresponded, within the limits of accuracy previously mentioned, to the following mass distribution of the full-scale airplanes:

|        | <u>Seaplane</u>                 | <u>Landplane</u>                 |
|--------|---------------------------------|----------------------------------|
| Weight | 2,935 Lb.                       | 2,803 Lb.                        |
| x/c    | 0.287                           | 0.293                            |
| z/c    | 0.080 (Below )<br>(Thrust Line) | -0.017 (Above )<br>(Thrust Line) |
| $I_X$  | 2,204 slug-ft. <sup>2</sup>     | 1,583 slug-ft. <sup>2</sup>      |
| $I_Y$  | 2,627 slug-ft. <sup>2</sup>     | 2,363 slug-ft. <sup>2</sup>      |
| $I_Z$  | 3,921 slug-ft. <sup>2</sup>     | 3,437 slug-ft. <sup>2</sup>      |

where

x/c is the ratio of the distance of the center of gravity aft of the leading edge of the mean aerodynamic chord to the mean aerodynamic chord.

z/c is the ratio of the distance of the center of gravity from the thrust line to the mean aerodynamic chord.

$I_x$ ,  $I_y$ , and  $I_z$  are the moments of inertia about the body axes X, Y, and Z, respectively.

The seaplane model was tested at an equivalent test altitude of 6,000 feet ( $\rho = 0.001988 \text{ slugs/ft.}^3$ ). The landplane model was tested at an equivalent test altitude of 10,000 feet ( $\rho = 0.001756 \text{ slugs/ft.}^3$ ).

Tests were also conducted with the models in other loading conditions. To obtain these loading variations, provision was made for distributing weight along the wings (changing the moments of inertia  $I_x$  and  $I_z$ ), for distributing weight along the fuselage (changing the moments of inertia  $I_y$  and  $I_z$ ), and for moving fuselage weights fore and aft (changing the center-of-gravity position).

Apart from a few comparison tests conducted with the cowled engine, all of the seaplane tests were performed with the uncowed engine. All of the landplane tests were performed with the cowled engine. During the landplane tests the tail wheel was purposely omitted as this tends toward more conservative model test results.

#### PRECISION

The accuracy of the model results is believed to be within the following limits:

|                    |           |                  |
|--------------------|-----------|------------------|
| V                  | - - - - - | $\pm 2$ percent. |
| $\Omega b/2V$      | - - - - - | $\pm 3$ percent. |
| $\alpha$           | - - - - - | $\pm 3$ degrees. |
| $\beta$            | - - - - - | $\pm 1$ degree.  |
| Turns for recovery | - - -     | $\pm 1/4$ turn.  |

The preceding limits may be exceeded for certain cases where it is difficult to handle the spin in the tunnel due to the high air speed or to the wandering or oscillatory nature of the spin.

## RESULTS

The results of the investigation are presented in charts I - XV, table 1, and figure 3. The steady-spin parameters presented in the charts were determined by methods described in reference 1, and have been converted to the corresponding full-scale values. The angle  $\alpha$  is the angle between the thrust axis and the vertical.  $\phi$  is the angle between the span (Y) axis and the horizontal, and is considered positive when the in-board wing is below the horizontal. The angle of sideslip,  $\beta$ , is approximately equal to  $\phi - \sigma$ , where  $\sigma$  is the helix angle (angle between flight path and vertical). Inward sideslip is considered positive;  $\sigma$  is also considered positive. For the H3N-3 model spins  $\sigma$  averaged about 7 degrees for the seaplane and about 6 degrees for the landplane.

Recoveries were measured by the number of turns the spinning model made from the time the controls were observed to move until the spinning rotation ceased.

## DISCUSSION OF RESULTS

Comparison between model and airplane spin results (reference 2) indicates that because of scale and tunnel

effects, absence of detail in the model, and differences in operator's technique, the spin-tunnel results are not always in complete agreement with full-scale spinning data. In general, for a given loading condition and control setting, the model steady-spin results have shown a somewhat smaller angle of attack, a somewhat higher rate of descent, and, at a given angle of attack, from 5 degrees to 10 degrees more outward sideslip. The comparison made in reference 2 showed that 80 percent of the model recovery tests predicted satisfactorily the corresponding full-scale recoveries, 10 percent overestimated the full-scale recoveries, and 10 percent underestimated the full-scale recoveries.

With the exception of a few left spins which were run for purposes of comparison, the majority of the test spins were made to the right.

#### SEAPLANE INVESTIGATION

Charts I through VIII present graphically the results of the seaplane investigation. Chart I shows the effect of various control dispositions on the steady spin and recovery characteristics of the N3N-3 seaplane model in the normal loading condition. Recoveries by rudder reversal alone were generally satisfactory for all control settings. In recoveries from elevator down spins there was a tendency for the model to enter an inverted spin.

Difficulty was experienced in attempting to reproduce several of the spins in the normal loading condition, particularly if the elevators were full down. The spinning equilibrium was apparently unstable and if the model were not launched in the appropriate attitude for each spin, the model would recover of its own accord.

The fastest recoveries occurred with the elevator full up. The aileron effect was small. With the elevators full-up, deflection of the ailerons in either direction had little effect on either the steady spins or recoveries. With the elevator neutral deflecting the ailerons full with the spin tended to steepen the spin, while deflecting the ailerons against the spin produced oscillatory spins which were very difficult to hold in the tunnel. With the elevators full down the model would not spin with the ailerons full with the spin.

The model would not spin with the rudder held neutral or against the spin for any elevator-aileron disposition, except with the rudder held neutral, the elevators up, and the ailerons full with the spin.

Chart II emphasizes the importance of a rapid and complete reversal of the rudder during the recovery maneuver. Recoveries effected by simply neutralizing the rudder were slow, particularly with the elevators down. Recoveries by releasing the rudder, although



faster than those obtained by neutralizing the rudder, were much inferior to those obtained by complete rudder reversal.

The effects of various mass changes were also determined. Chart III illustrates the effect of center-of-gravity movement, movement of the center of gravity either slightly forward or aft had no great effect, and similarly increasing the lateral mass distribution by adding weights along the wings (increasing  $I_x$  and  $I_z$ ) (chart IV) had little effect on recovery characteristics.

Increasing the longitudinal mass distribution by extending weights along the fuselage (increasing  $I_y$  and  $I_z$ ) was somewhat detrimental to recovery and the results of this investigation are presented in chart V. The loading associated with the slowest recoveries (and hereafter referred to as the "worst load") was obtained by increasing  $I_y$  and  $I_z$  by 22 percent  $I_y$ . This loading has been investigated more thoroughly than the others and the results are presented in chart VI.

In general, the spins with the worst load were slightly flatter than the corresponding spins in normal load and the recoveries were longer, the greatest effect being with the elevators down. The model would not spin with the rudder held neutral <sup>or</sup> against the spin.

This particular loading, however, is unlikely to be encountered in practice.

Chart VII affords a comparison between left erect spins, right erect spins, and right inverted spins. Considerable difficulty was experienced obtaining left spins because of their wandering nature, particularly when the ailerons were set against the spin.

Although the left and right spins were in general agreement, there were a few marked differences leading to the inference that in these instances the model may have been capable of spinning in two distinct regimes. Recoveries from all the spins, however, were rapid.

The seaplane model would spin inverted for only three aileron-elevator dispositions. Recoveries by rudder reversal were effected rapidly.

Chart VIII indicates that both the cowed and uncowed engines gave substantially the same results.

#### LAND PLANE INVESTIGATION

The remaining charts, charts IX through XV, pertain to the N3N-3 model tested as a landplane. The landplane was spun with the cowed engine in place, and, in view of the negative results previously obtained with the seaplane, no check spins to determine the effect of the cowling were made. The tail wheel was off for all landplane tests.

Chart IX shows the effect of control disposition on the steady spin characteristics of the W3N-3 land-plane model in the normal loading condition. In general, the steepest spins and fastest recoveries occurred with the elevator full up. The steady spin became flatter for neutral and down elevator settings and the slowest recoveries were generally effected from spins with the elevator full down. Ailerons set with the spin steepened the spin somewhat and assisted recovery. Ailerons set half-with the spin produced lesser effects. Ailerons set one half against the spin flattened the spin somewhat but had little effect on the recovery characteristics. With the exception of the elevator-up spin, setting the ailerons full against the spin also flattened the spin although the recoveries were slightly faster than those obtained with the ailerons only half against or neutral.

With the exception of one spin, rudder neutral, elevator up, ailerons with the spin, the model would not spin with the rudder held neutral or full against the spin. It will be observed that these results agree qualitatively with those obtained for the seaplane (chart I).

Chart X brings out the importance of complete rudder reversal for recovery. This chart shows that recovery by neutralizing the rudder was much slower than

recovery effected by complete and rapid reversal of the rudder alone. Failure to consider this point may lead to slower recoveries in flight than anticipated.

As in the case of the seaplane, moderate changes in c. g. location (chart XI) and in distribution of weight along the wings (chart XII) had negligible effect and the loading associated with the slowest recoveries (and called the worst load) was obtained by adding weights along the fuselage. For the landplane this loading consisted of increasing  $I_y$  and  $I_z$  by 30 percent of  $I_y$ , and the effects of this loading are presented in charts XIII and XIV.

In general, the effect of the worst load was to flatten the spin somewhat and increase the turns for recovery. The slowest recoveries again were obtained from the elevator-down spins, and the fastest, from elevator-up spins. The aileron effect was quite marked. In the worst load, ailerons with the spin gave steeper spins than those which were obtained with the normal loading. Ailerons against the spin flattened the spin. It should be noticed that simply neutralizing the rudder failed to effect a recovery from the elevator-down spin with the ailerons neutral. As is apparent from the chart, the model would spin in the worst loading with the rudder neutral for several elevator-aileron dispositions, but would not spin for any elevator-aileron dispositions with the rudder against the spin.

Chart XV contains a comparison of left and right spins of the landplane model in the worst loading condition. These tests were run in the worst rather than the normal load to facilitate testing. The left spins were apparently slightly steeper than the right spins but the discrepancies are small and might easily be due to asymmetry in the model incurred during the prolonged series of tests.

Chart XV also includes inverted right spins of the landplane model in the normal loading condition. In this condition the model would spin for only three control dispositions and recoveries from all three spins were rapid.

#### COMPARISON OF LANDPLANE AND SEAPLANE

In the normal loading condition the landplane spins were generally easier to hold than the seaplane spins. The landplane tended to give slightly flatter spins with consequent longer recoveries than the seaplane. Both models were prone to give spins with the rudder held neutral, the elevator up, and the ailerons full with the spin.

The effect of control disposition on the spinning characteristics of the two models was similar. Parallel effects were also observed for variations in mass distribution and center-of-gravity location.

A rigid comparison of the two models cannot be

made at this time since the two versions of the model were tested at slightly different equivalent altitudes (6,000 feet for the seaplane, 10,000 feet for the landplane). It might be mentioned, however, that in general it has been observed that the effect of an increase in altitude is to flatten the spin somewhat and to retard recovery. This may partially account for the inferior recovery characteristics exhibited by the landplane model.

#### TAIL MODIFICATION TESTS

Although both the seaplane and landplane models exhibited good spinning characteristics, it was thought advisable to run a series of tests with various tail modifications in order to determine which locations for additional area were most effective. These tests were performed on the landplane/<sup>model</sup> but it is believed that the results are equally applicable to the seaplane model as well. A number of auxiliary fins (of approximately equal area) were tried in various locations about the rudder and their effects studied. Figure 3 shows three of the best and one of the worst auxiliary fins tried.

The investigation is summarized in table 1.

The region in the vicinity of auxiliary fin C appears to be the optimum location for additional fin area. The upper rear portion of the rudder (fin A) also seems quite good. The region immediately forward of the vertical fin is definitely a poor location.

#### CONCLUSIONS

On the basis of the model tests, the following recovery technique is recommended for both the N3N-3 landplane and seaplane:

During the steady spin hold the stick full back, the rudder full with the spin, and the ailerons neutral. For recovery, rapidly move the rudder full against the spin and later, after an additional one-half turn has been made, briskly move the elevator full down.

Slow control movements should be avoided, and after recovery has been effected the stick should be eased back to avoid the possibility of entering an inverted spin.

The model tests of the N3N-3 also indicate the following spin characteristics:

##### Seaplane:

1. In the normal loading condition the airplane should recover rapidly by complete reversal of the rudder. The fastest recoveries

should be effected with the elevators full up. In general, recoveries effected by neutralizing the rudder will be unsatisfactory.

2. There exists some danger of entering an inverted spin from recoveries with the elevators full down.

3. Reasonable changes in mass distribution should have no great effect upon recovery characteristics.

4. Recovery from any inverted spin by reversal of the rudder should be rapid.

5. The effect of a cowl over the engine should be negligible.

Landplane:

1. In the normal loading condition the airplane should spin normally. Lowering the elevators will tend to flatten the spin. Putting the ailerons with the spin will tend to steepen the spin. Recoveries effected by rudder reversal should be fastest with the elevator up. Recoveries effected by neutralizing the rudder should be slow.

2. There exists some danger of entering an inverted spin from recoveries with the elevator full down.



3. Reasonable changes in mass distribution should have no great effect upon recovery characteristics although increased loading along the fuselage <sup>tend to</sup> should/retard recovery.

4. Recovery from any inverted spin by rudder reversal should be rapid.

5. The effective regions for additional vertical tail area are below the horizontal tail surfaces and around the rear upper portion of the rudder.

Langley Memorial Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., January 6, 1940.

*Charles J. Donlan*

Charles J. Donlan,  
Junior Aeronautical Engineer.

Approved:

*Elton W. Miller*

Elton W. Miller,  
Principal Mechanical Engineer.

PWP

REFERENCES

1. Zimmerman, C. H.: Preliminary Tests in the N. A. C. A. Free-Spinning Wind Tunnel. T. R. No. 557, N. A. C. A., 1936.
2. Seidman, O., and Neihouse, A. I.: Comparison of Free-Spinning Wind Tunnel Results with Corresponding Full-Scale Spin Results. Confidential Memo. Report, N. A. C. A., December 1938.

EFFECT OF AUXILIARY FINS

| TEST<br>CONDITIONS               | Control Setting |             |                 |               |                 |               |                                                  | Turns<br>for<br>Recovery |
|----------------------------------|-----------------|-------------|-----------------|---------------|-----------------|---------------|--------------------------------------------------|--------------------------|
|                                  | Ailerons        |             | Rudder          |               | Elevator        |               |                                                  |                          |
|                                  | Rt.<br>Deg.     | Lt.<br>Deg. | Initial<br>Deg. | Final<br>Deg. | Initial<br>Deg. | Final<br>Deg. |                                                  |                          |
| WORST LOADING ✓<br>NO ADDED FINS |                 |             | 0               | 0             | 30W 0           | 25D 25D       | ∞                                                |                          |
| FIN A ADDED ✓                    |                 |             | "               | "             | "               | "             | 4, 4                                             |                          |
| FIN B ADDED ✓                    |                 |             | "               | "             | "               | "             | 4 <sup>1</sup> / <sub>4</sub> , (4) <sup>B</sup> |                          |
| FIN C ADDED ✓                    |                 |             | "               | "             | "               | "             | 3, 2 <sup>1</sup> / <sub>2</sub>                 |                          |
| FIN D ADDED ✓                    |                 |             | "               | "             | "               | "             | ∞                                                |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |
|                                  |                 |             |                 |               |                 |               |                                                  |                          |

✓ Wts. added along fuselage ( $I_y$  and  $I_z$  increased 30% of  $I_y$ )

✓ Model hit net still spinning in 4 turns.

W = With; D = Down; ∞ = No recovery

NOTE: EACH FIN WAS TESTED SEPARATELY

L.F. 634

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BY: CSD  
CHK: J.W.D

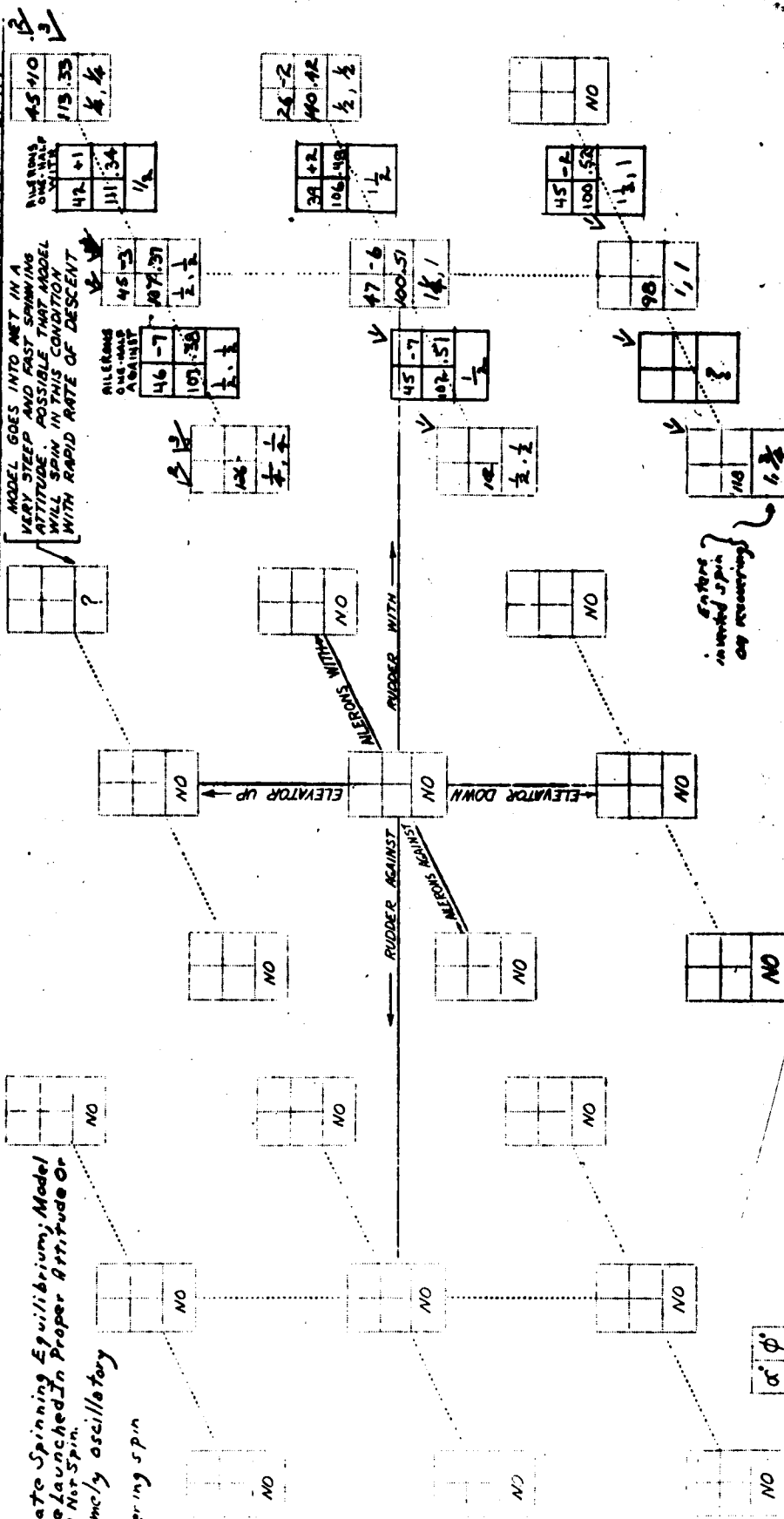
## SEAPLANE - UNCOWLED ENGINE

MODEL 43N-2 EFFECT OF CONTROLS LOADING NORMAL COCKPIT OPEN LANDING GEAR ALL FLOATS ON LAP SETTING NO FLAPS  
RECOVERY - BY MOVING RUDDER FROM POSITION INDICATED ON CHART TO FULL AGAINST SPIN

✓ Delicate Spinning Equilibrium; Model  
Must Be Launched In Proper Attitude Or  
It Will Not Spin.

It will Not Spin.  
Extremely oscillatory

$\frac{1}{3}$  und 6 uapm



NO MEANS MODEL WOULD NOT SPIN  
"? MEANS MODEL WOULD SPIN BUT NOT

KEY -

|        |               |
|--------|---------------|
| $\phi$ | $\frac{A}{B}$ |
| $x$    | $\frac{C}{D}$ |

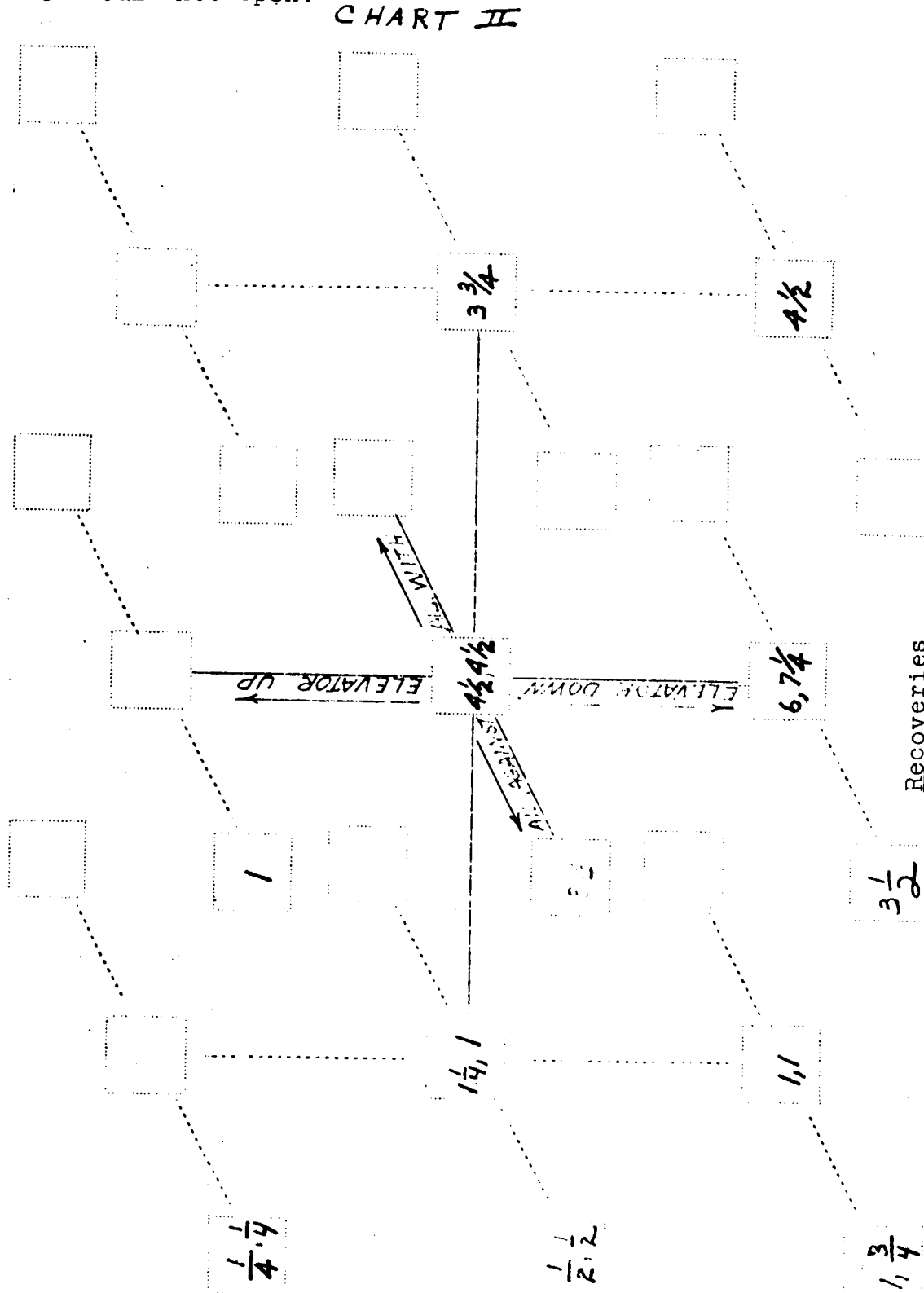
**OR REASONS NOTED.**

22. BY JMD.  
55. BY JMD.

Note: Chart shows control disposition prior to recovery attempt. Figures in squares are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART II

RECOVERY BY COMPLETE RUDDER REVERSAL      RECOVERY BY BRINGING RUDDER TO NEUTRAL      RECOVERY BY RELEASING RUDDER



Recoveries  
UNCONTROLLED Effect of RUDDER MOVEMENT ON RECOVERY  
ENSURING gear ALL FLOWS ON Flap setting NO FLAPS  
Model A34-23 SEABEAC  
Cockpit ODEA  
Loading NCGM-1  
Recovery BY METEOR 1000000

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1000000

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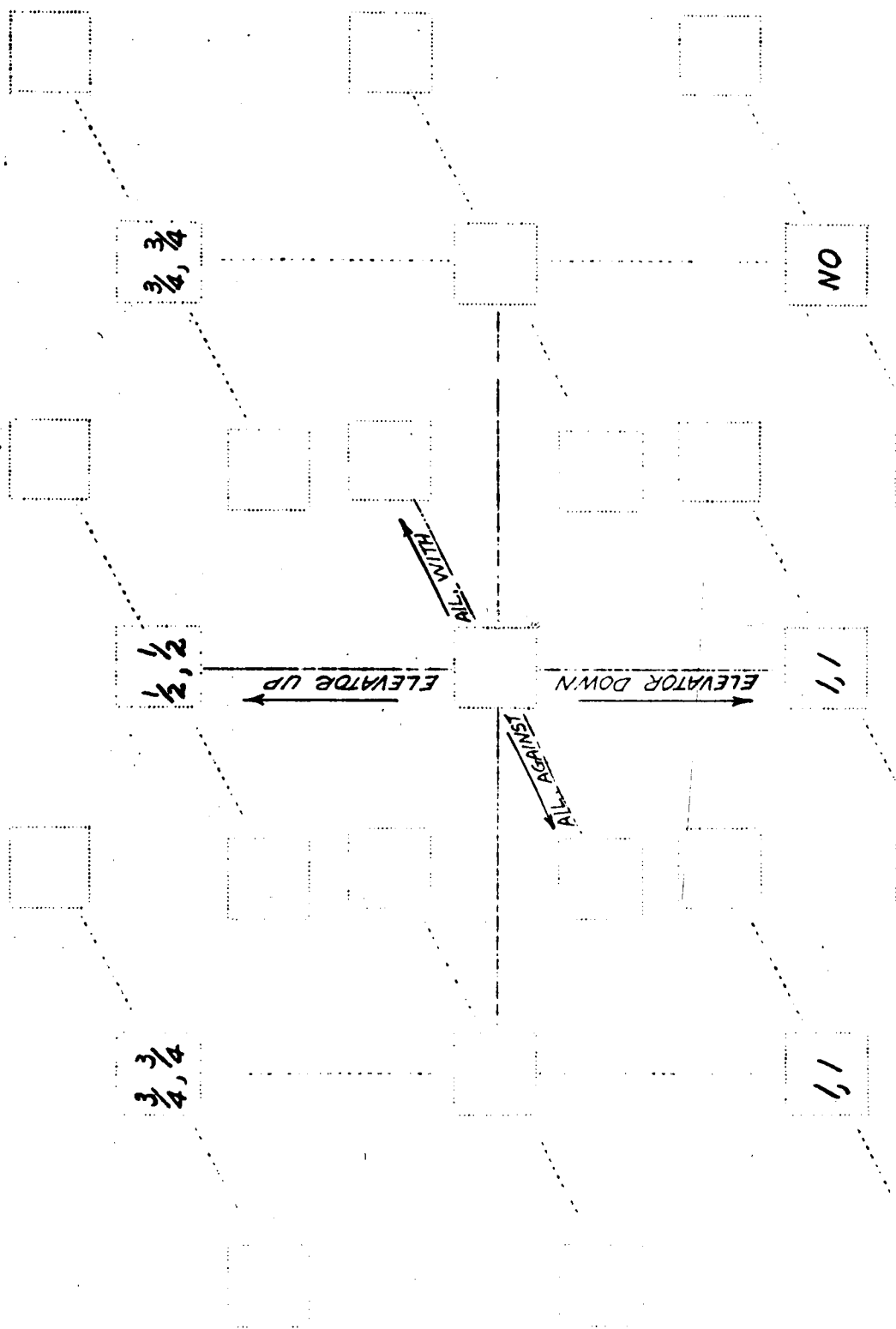
Note: Chart shows control disposition prior to recovery attempt. Figures in circles are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

### CHART III

C.G. FORWARD 4% M.A.C.  
(2.16 INCHES FULL-SCALE)

NORMAL

C.G. AFT. 4% M.A.C.  
(2.16 INCHES FULL-SCALE)



Recoveries

Model N3N-3 SEAPLANE UNKOWNL Effect of C.G. POSITION

RECOVERY NORMAL BY RUDDER REVERSAL ALONE (RUDDER WITH SPIN PRIOR TO RECOVERY ATTEMPT) ALL FLOATS ON Plan setting NO FLAPS

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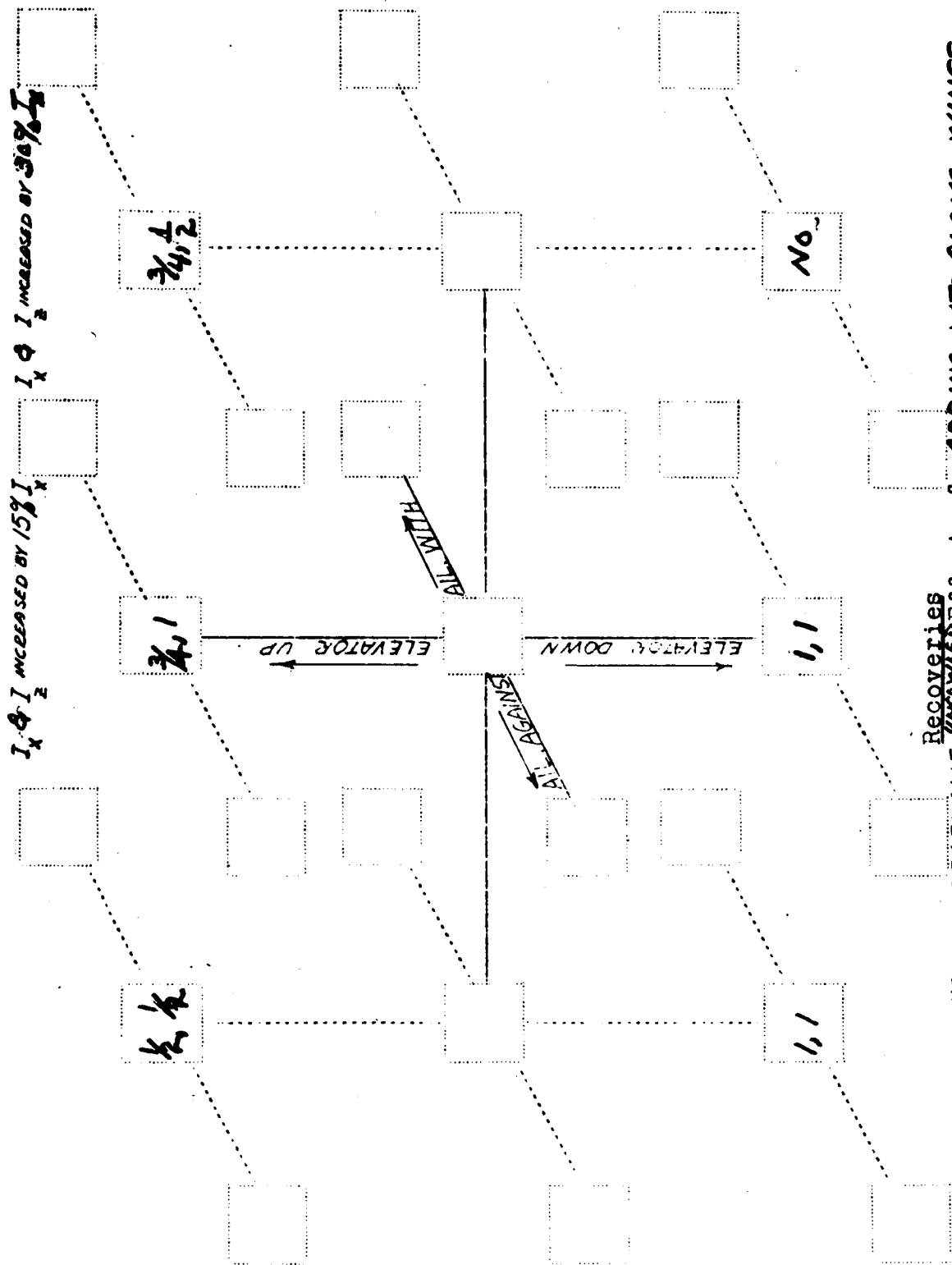
National Advisory Committee for Aeronautics

Note: Chart shows control disposition prior to recovery attempt. Figures in circles are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

# CHART - IV

WT. ADDED ALONG WINGS

NORMAL



Recoveries

Model N3N-3 SEAPLANE - UNDEVELOPED Effect of ADDING WTS ALONG WINGS  
 Loading AS INDICATED Cockpit OPEN ALL FLAPS ON Flap setting NO FLAPS  
 Recovered BY RUDDER REVERSAL ALONE (RUDDER WITH SPIN PRONE TO RECOVERY ATTEMPT)

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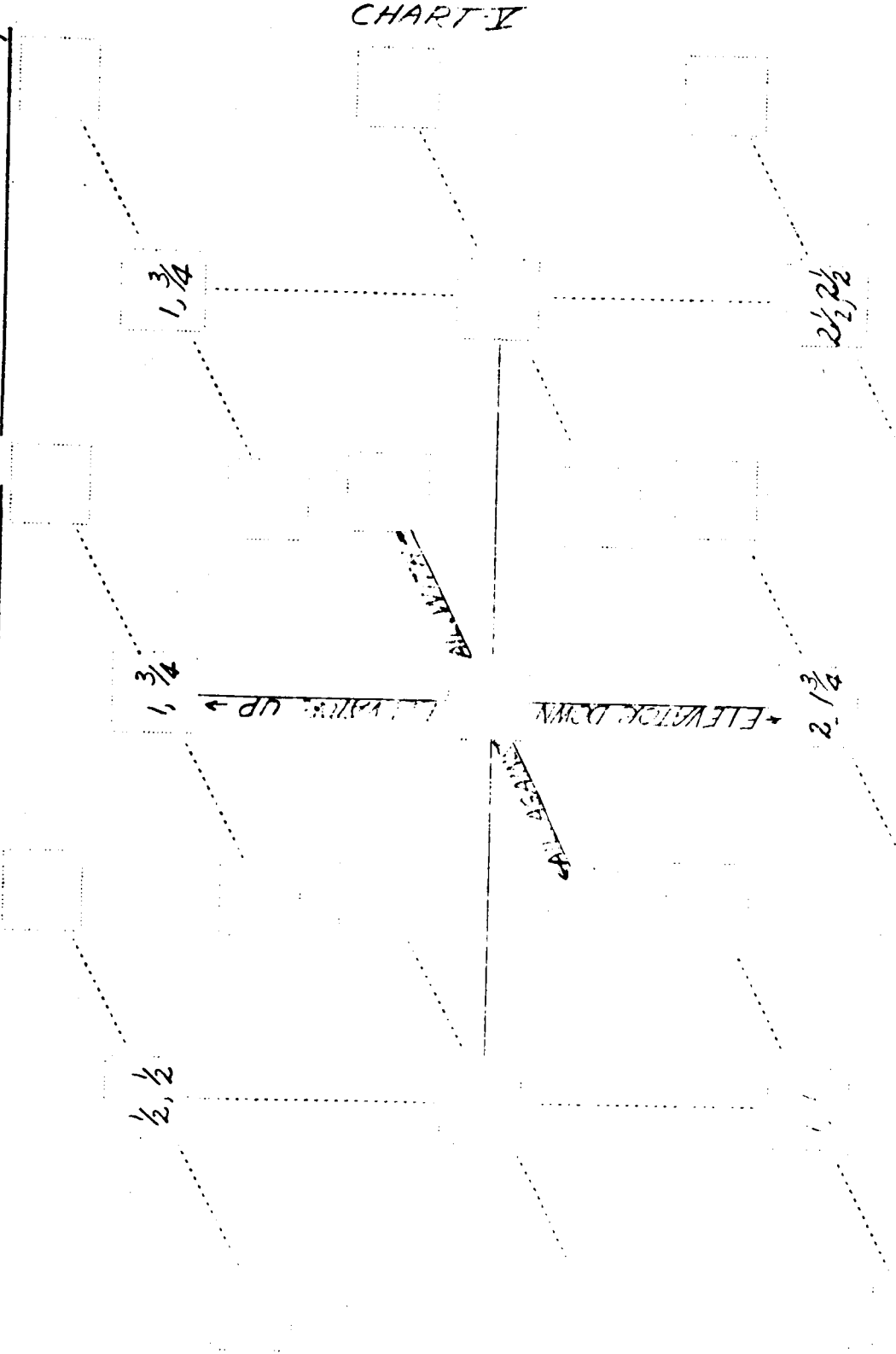
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DR. H. T. ...  
 C. K. B. ...

NORMAL LOAD

WTS. EXTENDED ALONG FUSELAGE

$I_Y$  &  $I_Z$  INCREASED 15%  $I_Y$ ,  $I_Y$  &  $I_Z$  INCREASED 22%  $I_Y$ \*



\* WORST LOAD

Recoveries

Model N2V-3 SLAFV...  
Loading AS...  
Recovery...  
Cockpit OPEN...  
Engine...  
Control...  
Disposition...  
Flaps...  
Spin...  
Prior...

Effect of EXTENDING WTS. ALONG FUSELAGE  
on ALL FLAPS ON Flap setting NO FLAPS  
CONTROL DISPOSITION - RECOVERIES WITH SPIN PRIOR

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CR-285



# CHART - VII

SEAPLANE - UNCOWLED MOTOR

SPIN CHARACTERISTICS

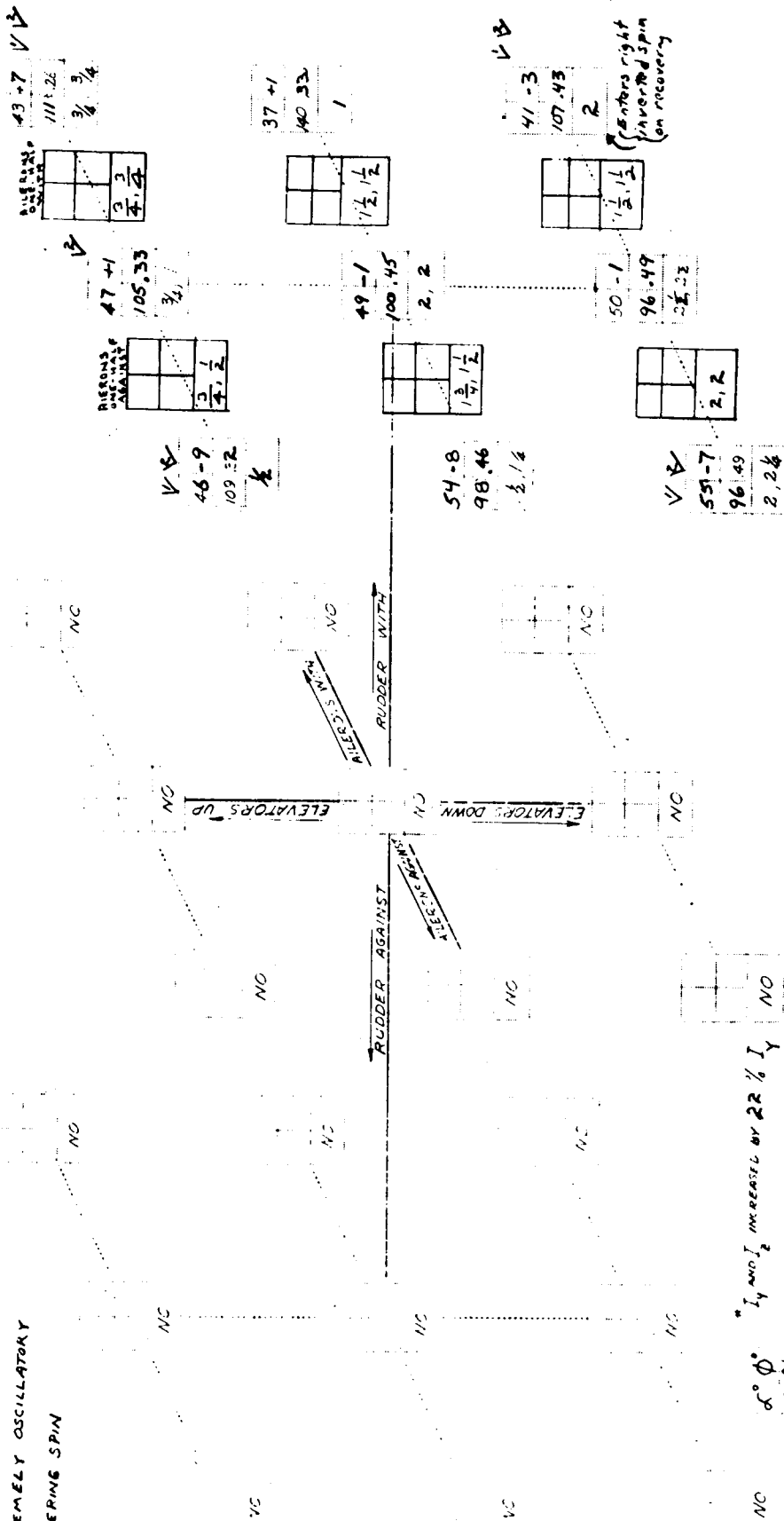
MODEL UN-3 EFFECT OF CONTROLS LOADING WORST\* COCKPIT OPEN

RECOVERY - BY MOVING RUDDER FROM POSITION INDICATED ON CHART TO FULL AND 157 SPIN

LANDING GEAR ALL FLOATS ON LAP SETTING NO FLAPS

✓ EXTREMELY OSCILLATORY

✓ WANDERING SPIN



\*  $I_y$  and  $I_z$  INCREASED BY 22%  $I_y$

\* NO MEANS MODEL WOULD NOT SPIN

\* YES MEANS MODEL WOULD SPIN BUT NO DATA OBTAINED FOR REASONS NOTED

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KEY -  
 NO  
 YES  
 RECOVERY

DR. BY - J.M.D.  
 CK. BY - G.D.

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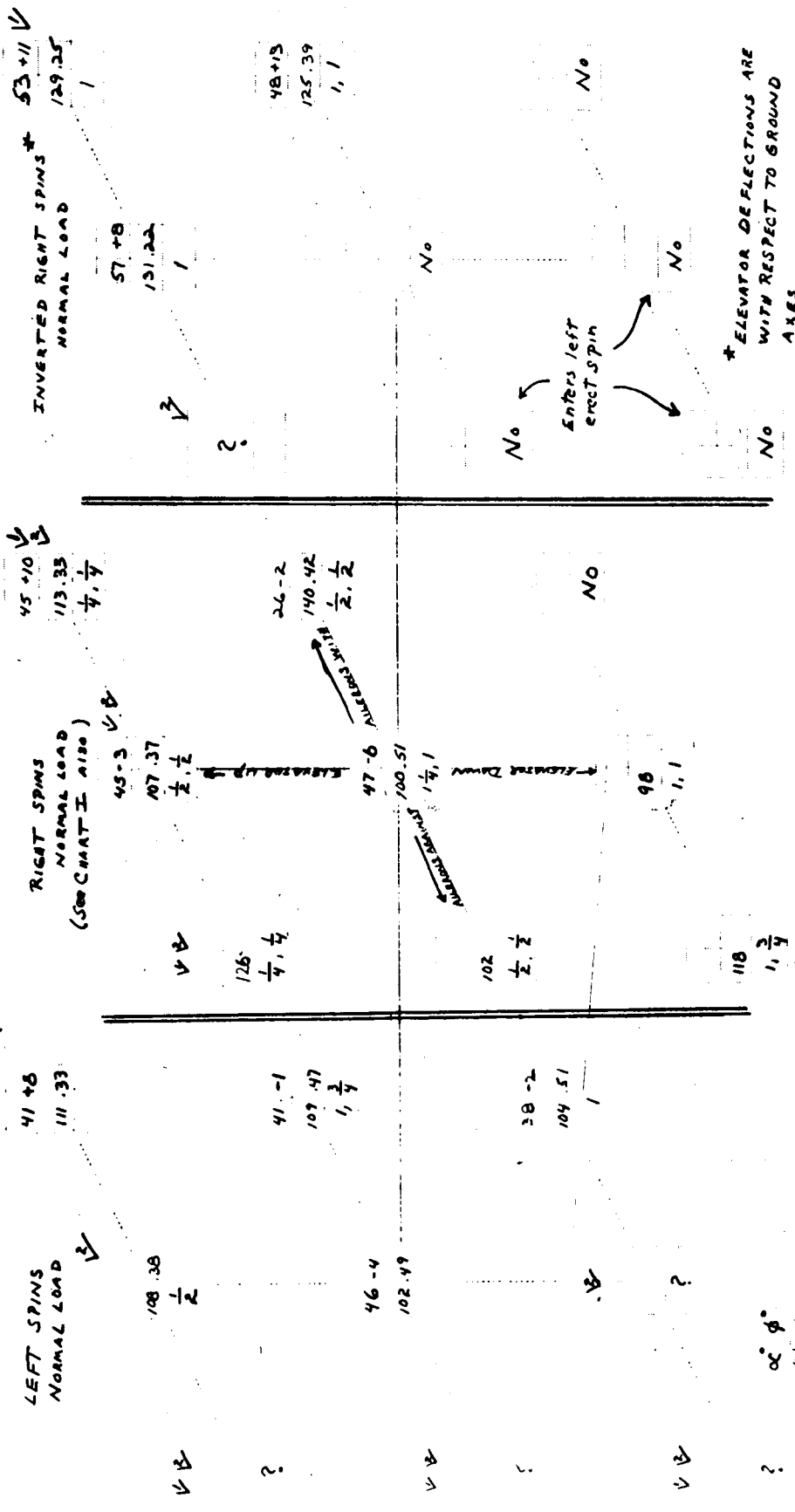
SEAPLANE-UNCOWLED ENGINE

MODEL 134-3 EFFECT OF SPIN DURING RECOVERY

RECOVERY BY RUDDER REVERSAL ALONE (RUDDER WITH SPIN PRIOR TO RECOVERY ATTEMPT)

# SPIN CHARACTERISTICS CHART - VII

COCKPIT OPEN - HANDING GEAR ALL FLOORS ON FLAP SETTING NO FLAPS



NO MEANS MODEL WOULD NOT SPIN  
 ? MEANS MODEL WOULD SPIN BUT NO DATA OBTAINED FOR REASONS NOTED  
 V OSCILLATORY; B WANDERING SPIN

831029  
 001 2719

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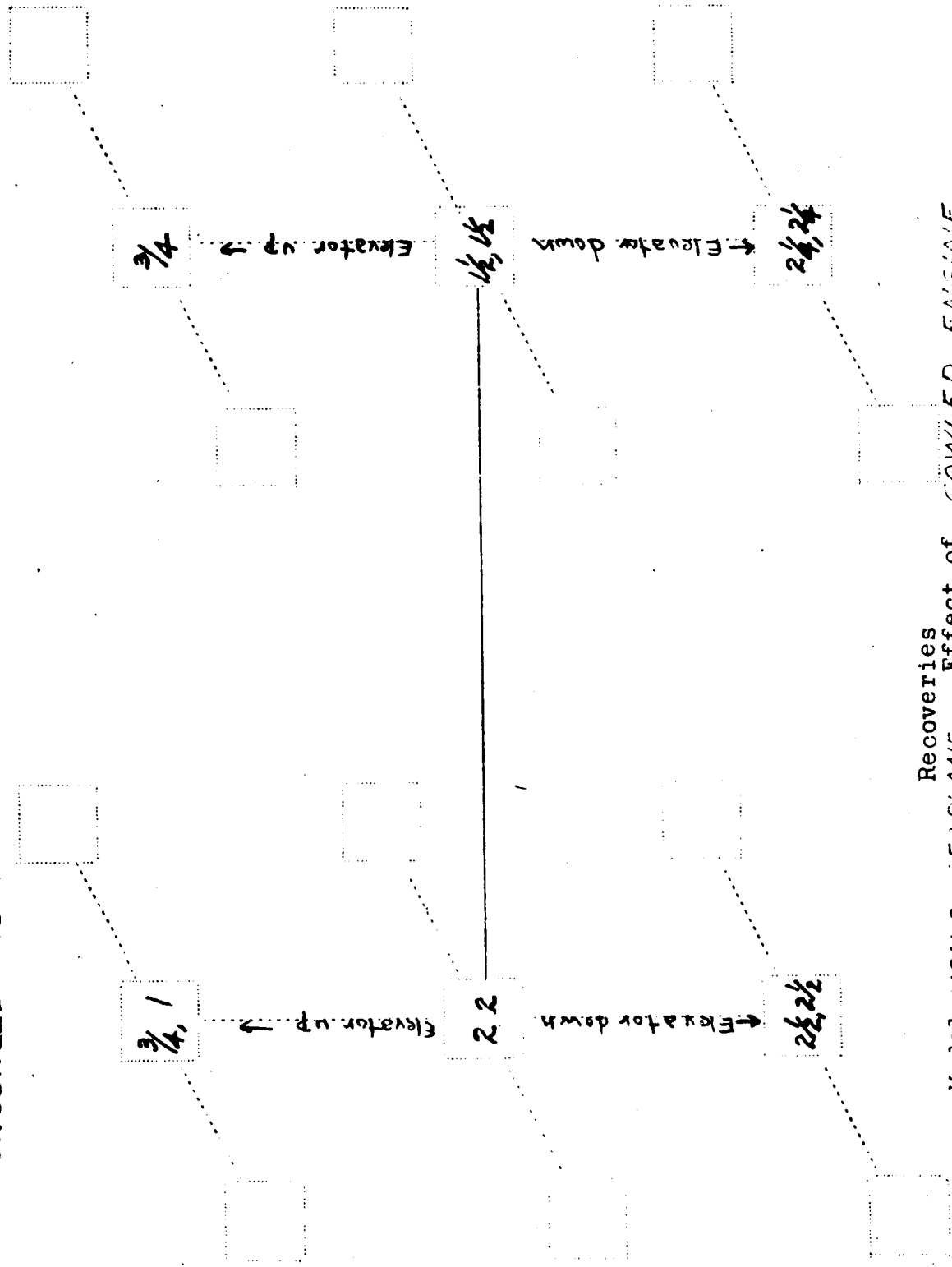
CONFIDENTIAL

Note: Chart shows control disposition prior to recovery attempt. Figures in ~~squares~~ are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART - VIII

COWLED ENGINE

UNCOWLED ENGINE



Recoveries

Effect of COWLED ENGINE

Model N3N-3 SEAFLENE

Loading WEIGHT

Cockpit DOWN

Landing gear ALL FLOATS ON Flap setting NO F.A.P.S.

Recovery BY RUDDER, REVERSER

(RUDDER WITH SPIN PRIOR TO RECOVERY)

CR. 100-1000

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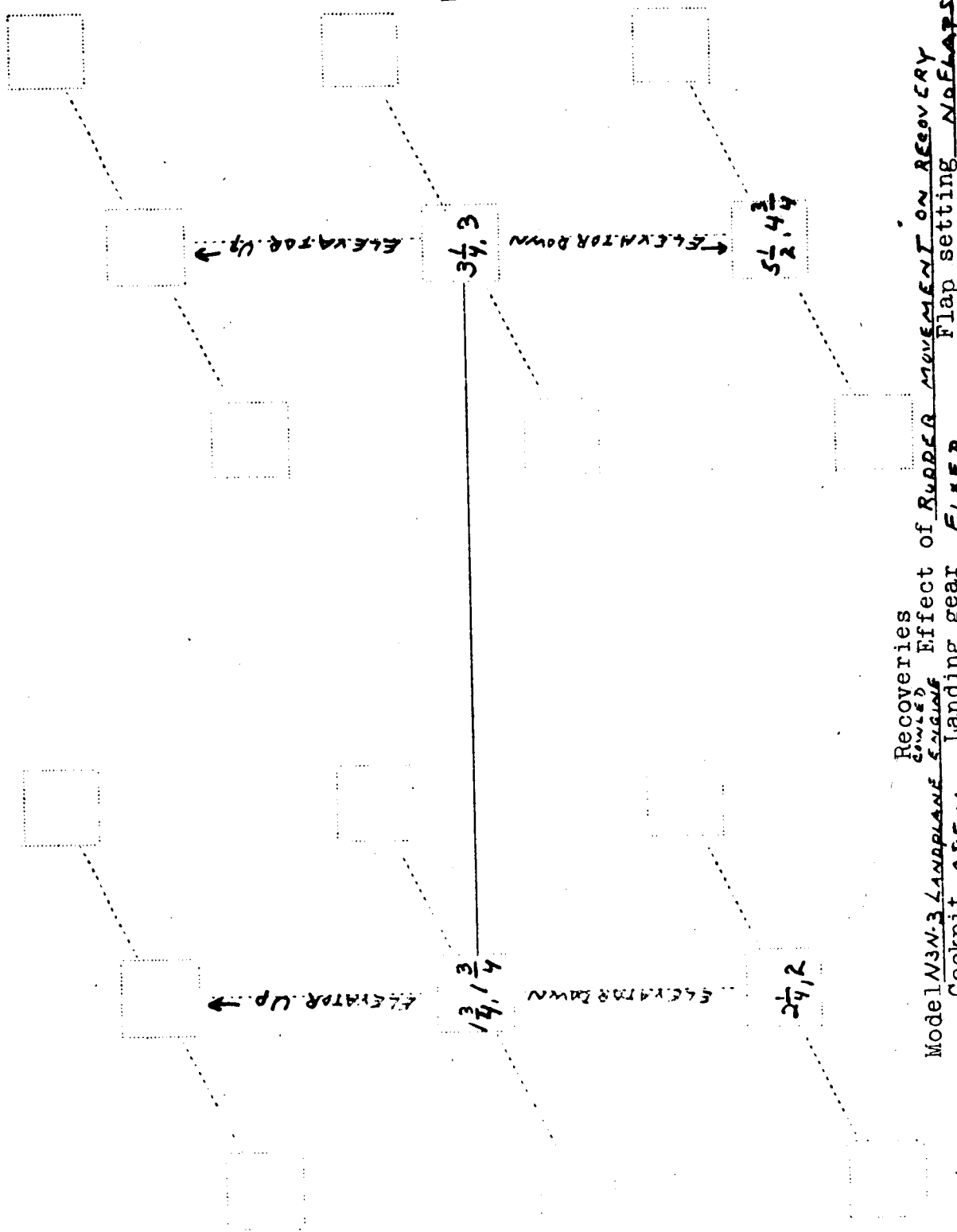
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RECOVERY BY COMPLETE  
RUDDER REVERSAL

RECOVERY BY NEUTRALIZING  
RUDDER

CHART X



Recoveries  
Model W3N-3  
Cockpit OPEN  
Landing gear  
FLAP setting  
NO FLAPS

Recovery BY METHOD INDICATED

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Note: Chart shows control disposition prior to recovery. Figures in ~~54001~~ are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

Note: Chart shows control disposition prior to recovery attempt. Figures in SQUARES are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART - XI

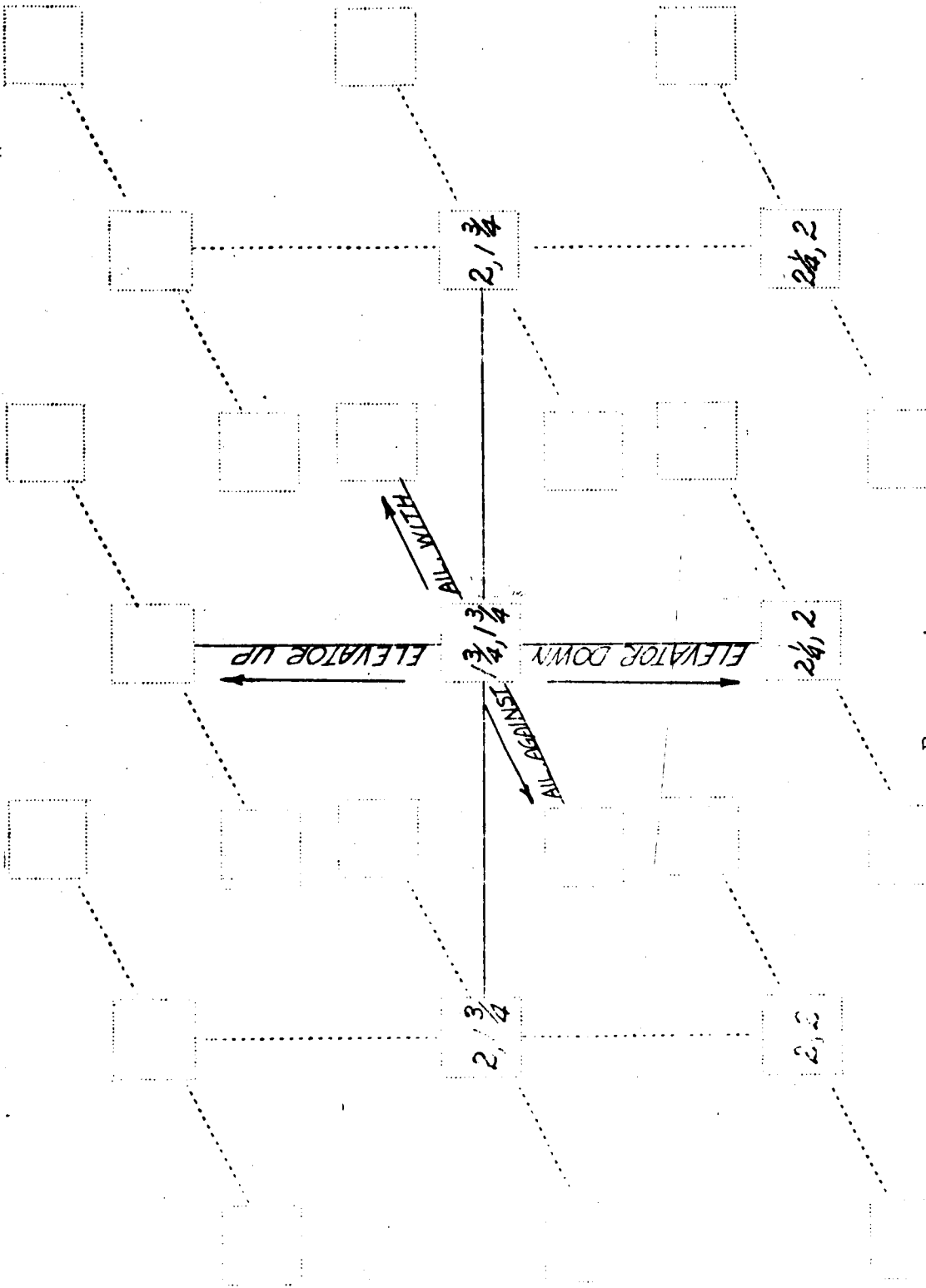
C.G. FORWARD 4% MAC.

2.16 INCHES FULL SCALE

NORMAL C.G. POSITION

C.G. AFT 4% MAC.

2.16 INCHES FULL SCALE



Recoveries

Model 13M3, ENGINE EFFECT of C.G. MOVEMENT

Cockpit OPEN Landing gear FIXED Flap setting NO FLAPS

BY RUDDER REVERSAL FROM INDICATED CONTROL DISPOSITION - RUDDER WITH PRIOR TO RECOVERY ATTEMPT

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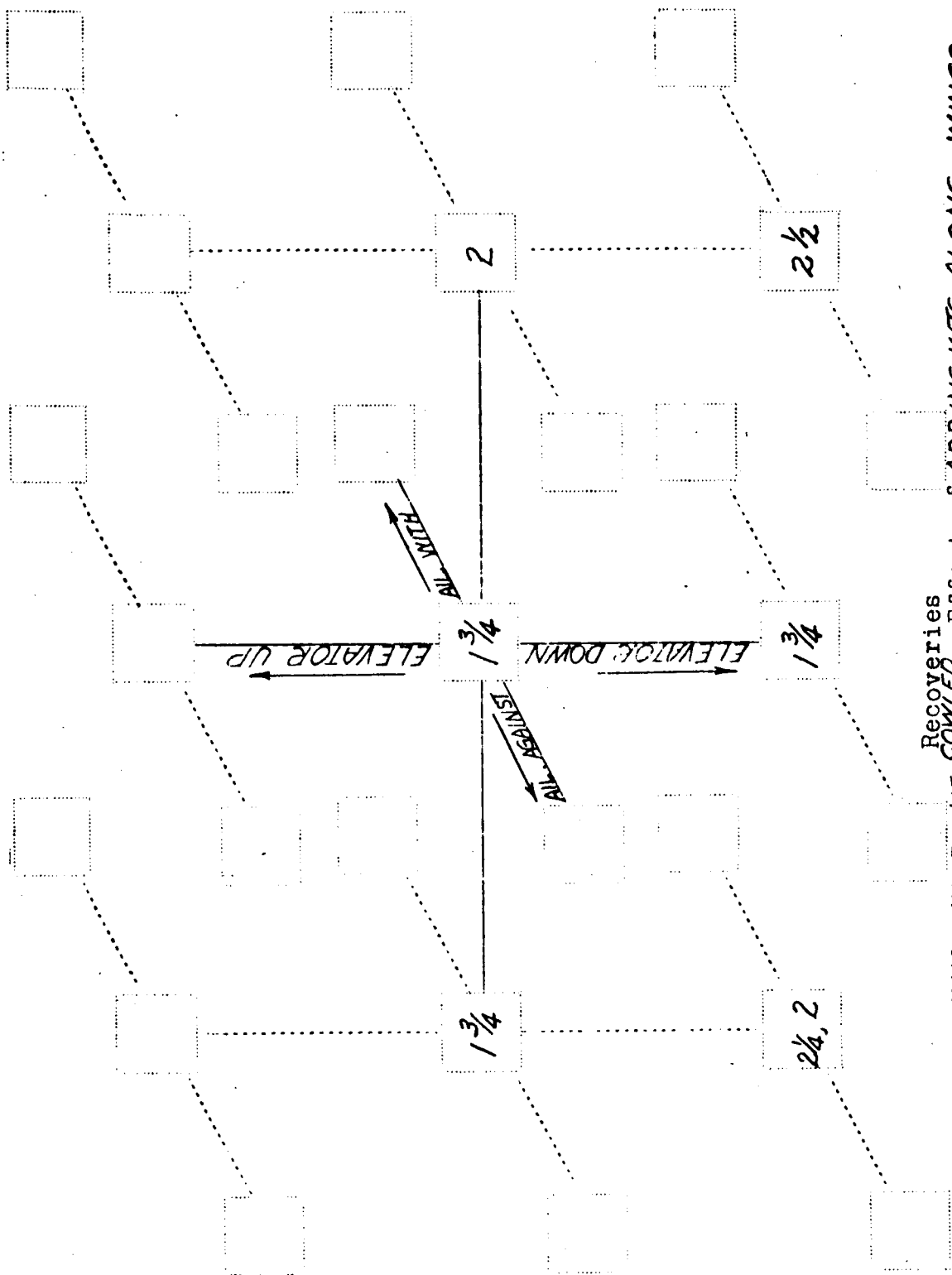
National Advisory Committee for Aeronautics

DR. J. W. O  
CC - 622

Note: Chart shows control disposition prior to recovery attempt. Figures in ~~circles~~<sup>SQUARES</sup> are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART-XII

NORMAL LOAD  
WT. ADDED ALONG WINGS  
( $I_x$  &  $I_z$  INCREASED 15%  $I_x$ ) ( $I_x$  &  $I_z$  INCREASED 30%  $I_x$ )



Recoveries

Model N3N-3-LANDPLANE  
Cockpit OPEN  
ENGINE  
Landing gear  
FIXED  
Flap setting NO FLAPS

Recovery BY RUDDER REVERSAL ALONE - RUDDER WITH SPIN PRIOR TO RECOVERY ATTEMPT

CONFIDENTIAL

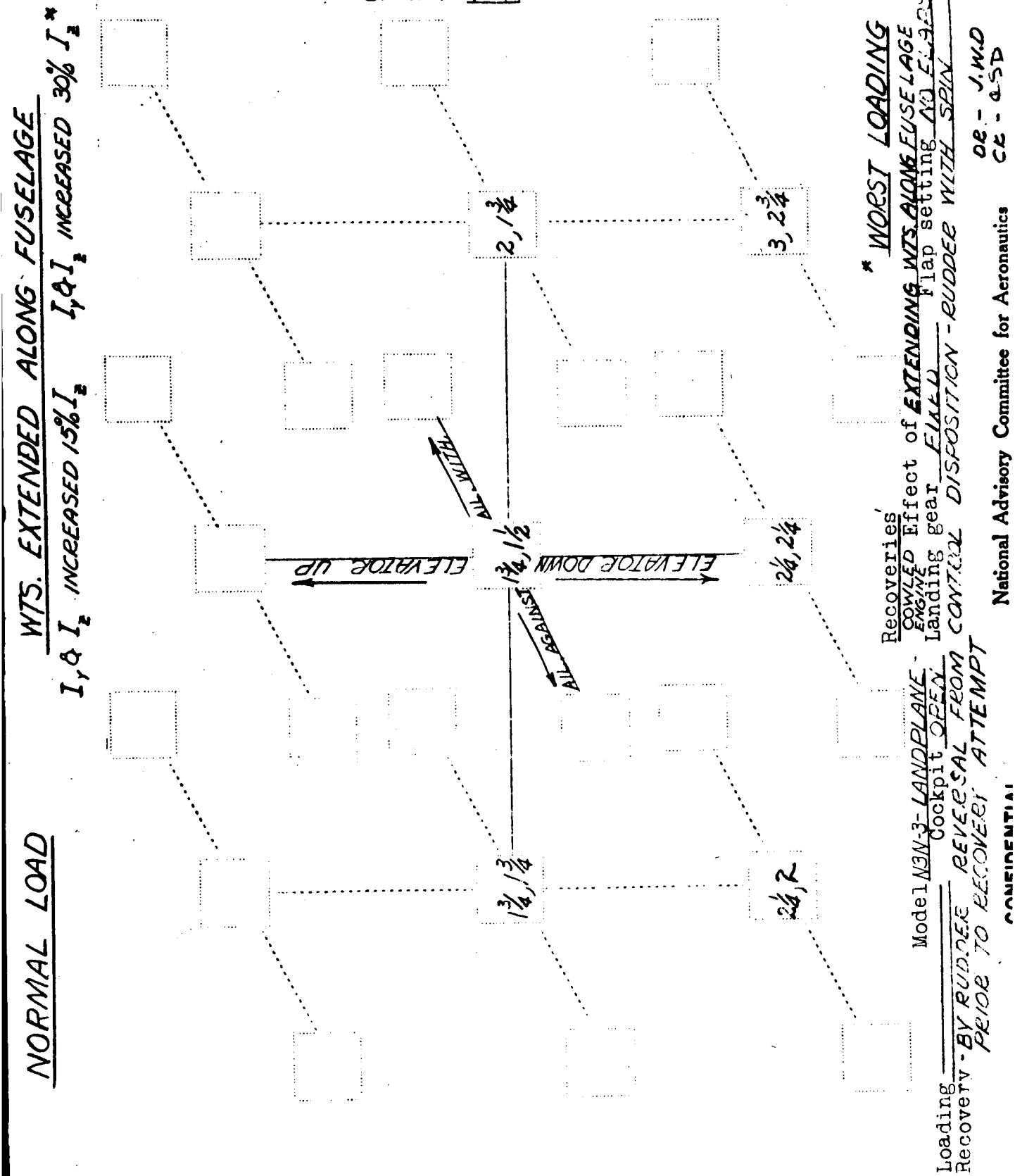
National Advisory Committee for Aeronautics

DR. - J.W.D.

CK - 050

Note: Chart shows control disposition prior to recovery attempt.  
 Figures in ~~circles~~ <sup>SQUARES</sup> are turns for recovery. "?" means model would spin but no data obtained for reasons noted. "No" means model would not spin.

CHART-XIII

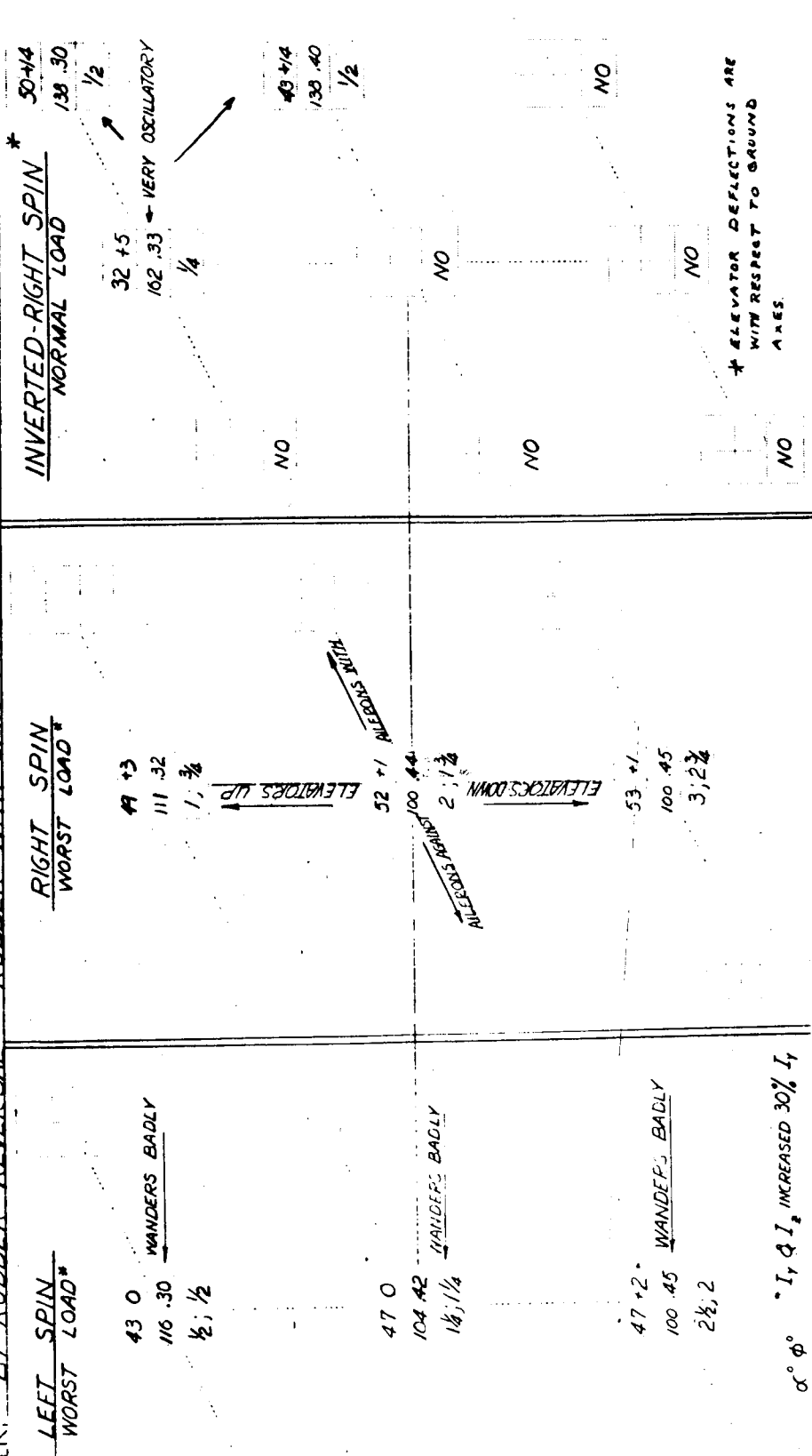






LAND PLANE - COWLED ENGINE  
MODELING EFFECT OF SPIN DIRECTION  
LOADING  
RECOVERY BY RUDDER REVERSAL - RUDDER WITH SPIN PRIOR TO RECOVERY ATTEMPT

SPIN CHARACTERISTICS CHART - XV  
COCKPIT OPEN LANDING GEAR FIXED FLAP SETTING NO FLAPS



\*  $I_1 \neq I_2$  INCREASED 30%  $I_1$

KEY -  
 $\alpha^\circ \phi^\circ$   
VEL ALB  
R/SEC 2V  
TURNS  
FOR  
RECOVERY

NO MEANS MODEL WOULD NOT SPIN  
MEANS MODEL WOULD SPIN BUT NO DATA OBTAINED FOR REASONS NOTED

DR. - JMD  
CK - a 35

National Advisory Committee for Aeronautics

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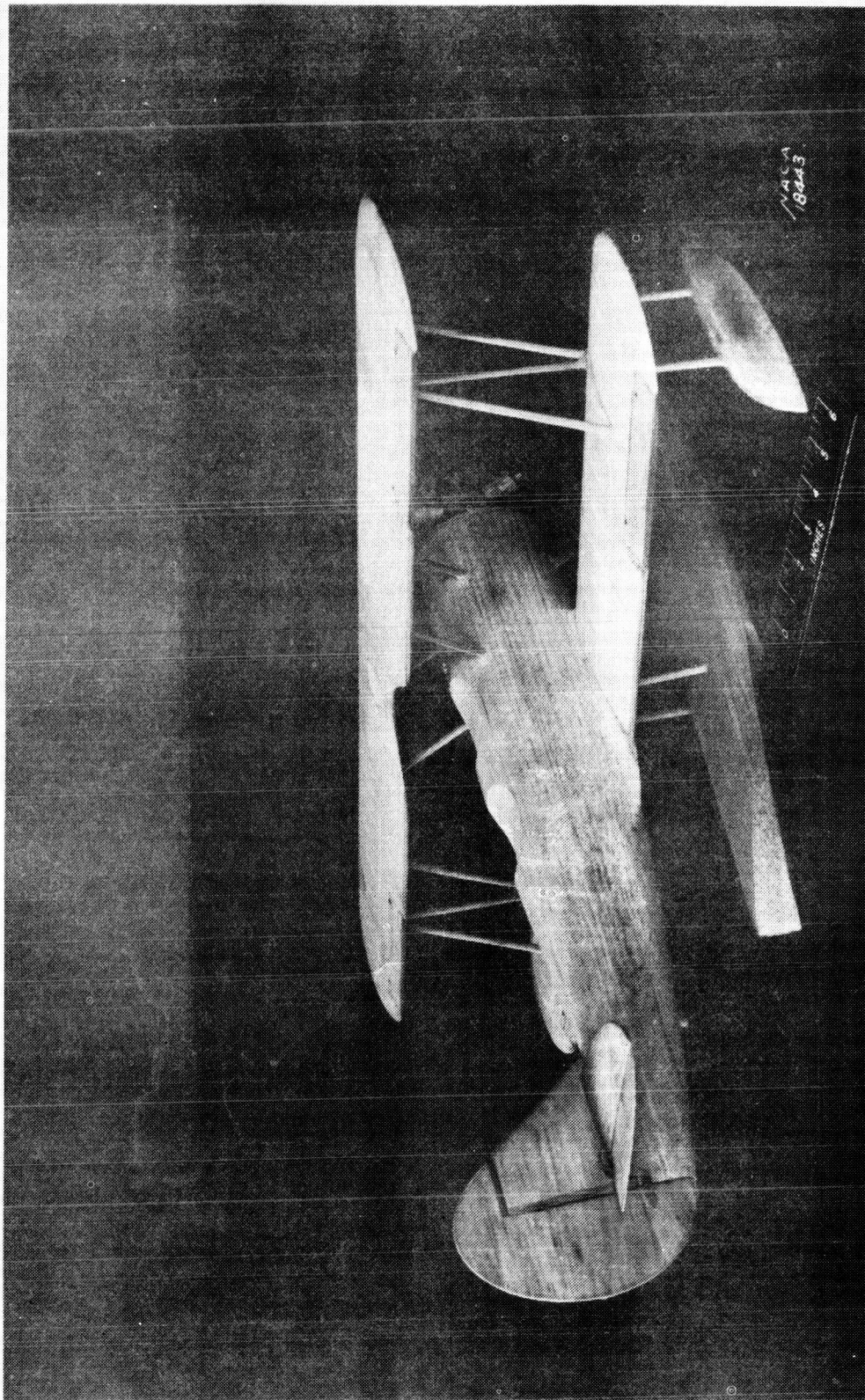


Figure 1. - 1/16-scale model of the N3N-3 seaplane.

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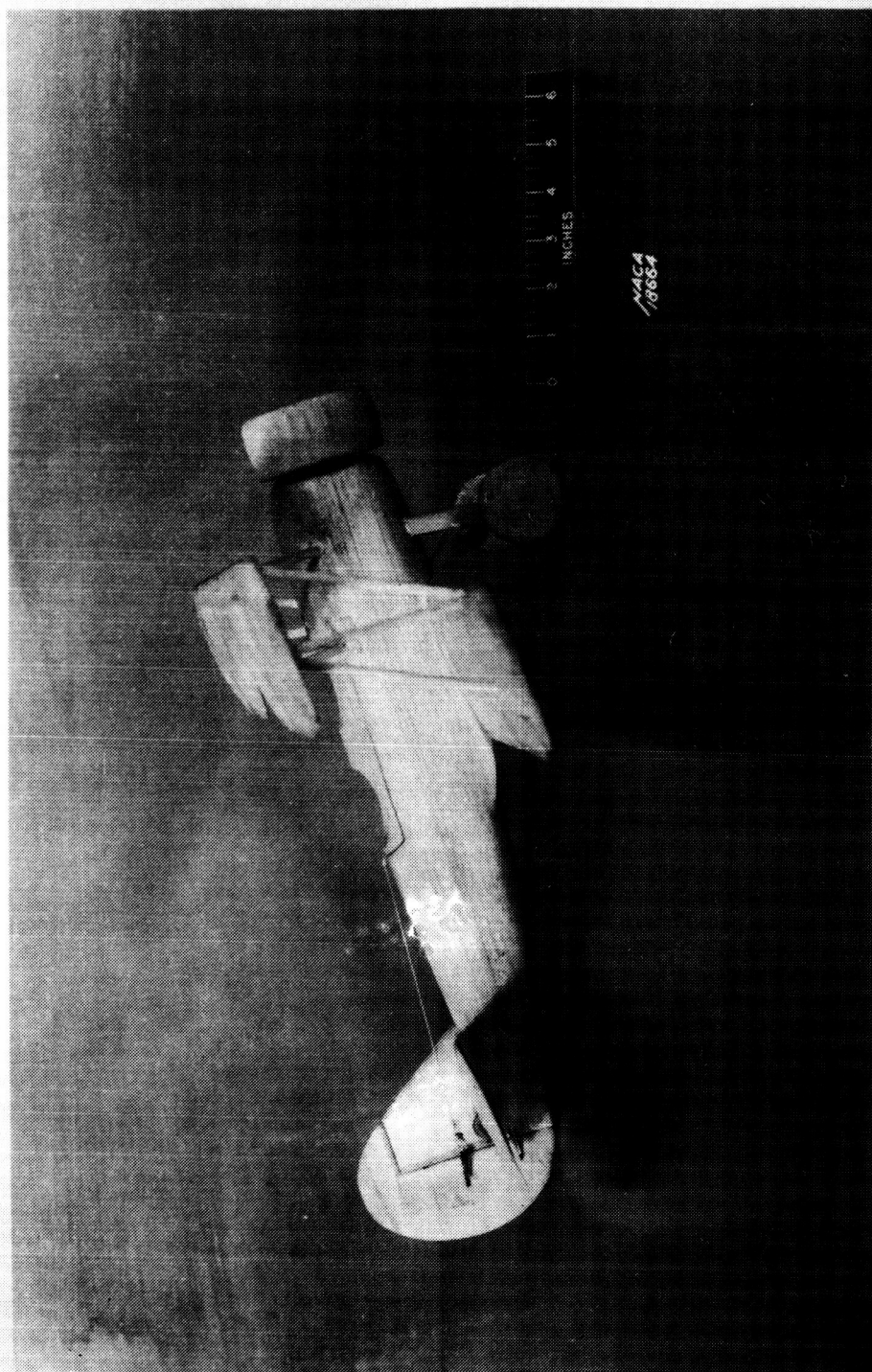


Figure 2. - 1/16-scale model of the N3N-3 landplane.

NOTE:

1. ALL AREAS ARE  
IN SQUARE FEET

$A - AREA = 3.02 \square'$

$D - AREA = 3.27 \square'$

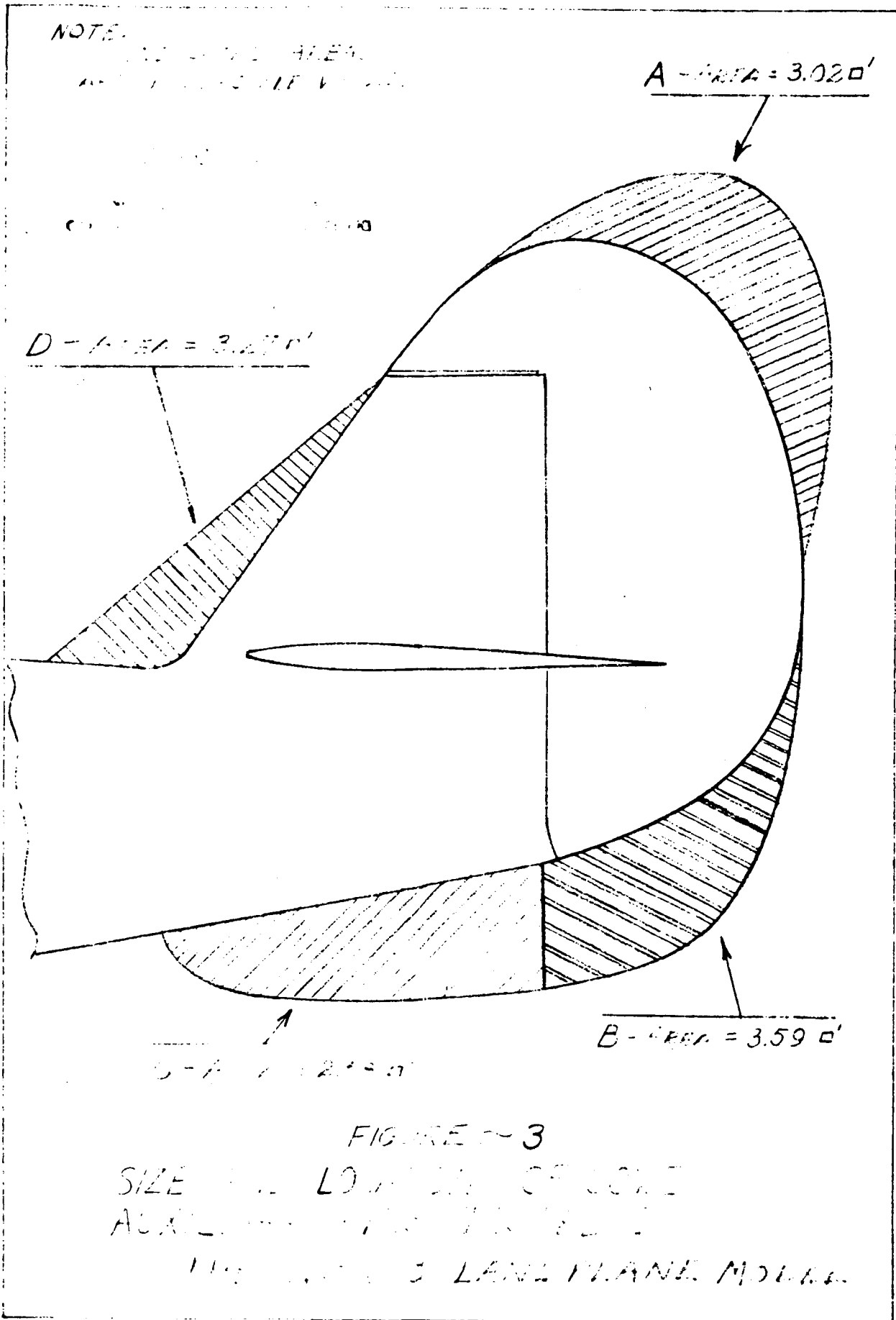
$B - AREA = 3.59 \square'$

$C - AREA = 2.22 \square'$

FIGURE - 3

SIZE AND LOCATION OF COKE  
ACCUMULATION TANKS

THE 100% LANE PLANE MODEL





# CLASSIFIED DOCUMENT

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## TECHNICAL NOTES

### NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 807

#### SPIN TESTS OF TWO MODELS OF A LOW-WING MONOPLANE TO INVESTIGATE SCALE EFFECT IN THE MODEL TEST RANGE

By Charles J. Donlan  
Langley Memorial Aeronautical Laboratory

Washington  
May 1941

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TO BE  
MAILED  
MAY 1941

# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## TECHNICAL NOTE NO. 807

### SPIN TESTS OF TWO MODELS OF A LOW-WING MONOPLANE TO INVESTIGATE SCALE EFFECT IN THE MODEL TEST RANGE

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#### SUMMARY

Concurrent tests were performed on a 1/16- and a 1/20-scale model (wing spans of 2.64 and 2.11 ft, respectively) of a modern low-wing monoplane in the NACA 15-foot free-spinning wind tunnel. Results are presented in the form of charts that afford a direct comparison between the spins of the two models for a number of different conditions.

Qualitatively, the same characteristic effects of control disposition, mass distribution, and dimensional modifications were indicated by both models. Quantitatively, the number of turns for recover and the steady-spin parameters, with the exception of the inclination of the wing to the horizontal, were usually in good agreement.

The results presented indicate that, within the range of Reynolds numbers used in the present investigation, such factors as difficulty of ballasting and testing are more important in determining proper model size than the changes in scale effect likely to result from the use of different sizes of models.

#### INTRODUCTION

The size of models used for testing in the NACA free-spinning wind tunnel is usually dictated by considerations of tunnel operating technique and ease of ballasting. With large models the actual testing is often difficult; with small models the proper mass or inertial balance is difficult to obtain. In general, the particular choice of model size is somewhat arbitrary because usually more than one size can be tested. It was therefore considered expedient to determine to what extent the experimental results vary with the actual size of the model tested.

At the present time, little information is available concerning the effect of size or scale within the model test range on the spin characteristics of dynamic scale models. With the exception of a British report (reference 1), which contains some rolling-balance results for two similar models, and of reference 2, which mentions the effect of scale on the data obtained from the spinning balance, previous scale-effect investigations have been concerned with the comparison of model results and full-scale results.

This paper presents the results of an investigation made in the NACA free-spinning wind tunnel to compare the spin characteristics of a 1/16- and a 1/20-scale model of a modern low-wing monoplane. The investigation included a comparison of results for the steady-spin and the recovery characteristics of the two models as regards the effects of control disposition, mass distribution, and dimensional modifications.

#### SYMBOLS

|                 |                                                                                                                 |
|-----------------|-----------------------------------------------------------------------------------------------------------------|
| $I_x, I_y, I_z$ | moments of inertia about model body axes, X, Y, and Z, respectively                                             |
| $b$             | span                                                                                                            |
| $c$             | mean aerodynamic chord of wing                                                                                  |
| $x/c$           | ratio of distance of center of gravity back of leading edge of mean aerodynamic chord to mean aerodynamic chord |
| $z/c$           | ratio of distance of center of gravity below thrust line to mean aerodynamic chord                              |
| $\alpha$        | angle of attack                                                                                                 |
| $V$             | air speed                                                                                                       |
| $\phi$          | angle of span (Y) axis to horizontal (positive when right wing is below the horizontal)                         |
| $R_A$           | Reynolds number of full-scale airplane                                                                          |
| $R_M$           | Reynolds number of model                                                                                        |



- N scale of model (1/16, 1/20, etc.)
- $\delta_e$  elevator deflection (positive up)
- $\Omega$  resultant angular velocity

### APPARATUS AND MODELS

The tests were performed in the NACA 15-foot free-spinning wind tunnel, as described in reference 3.

A 1/16- and a 1/20-scale model of a modern low-wing monoplane trainer with fixed landing gear were tested. The wing spans of the models were 2.64 and 2.11 feet, respectively. Photographs of the models are shown in figure 1. The models were constructed principally of balsa. For both models, wing and tail-surface contours were held to their true dimensions to within  $\pm 0.01$  inch; all other dimensions under 6 inches were held to within  $\pm 0.02$  inch; all other dimensions over 6 inches were held to within  $\pm 0.05$  inch; and angular relationships, such as wing setting, sweepback, and control settings, were held to within  $\pm 0.5^\circ$ .

Lead ballast added at suitable locations served to bring the weight, the moments of inertia, and the center-of-gravity locations to their appropriate values. A clockwork mechanism was installed on each model to hold the controls in position during the steady spins and to actuate the controls during the recovery tests. The weights, the moments of inertia, and the center-of-gravity positions of the two models were held to their true scaled-down full-scale values within the following limits:

Weight, percent . . . . .  $\pm 1$

Center-of-gravity position, percent M.A.C. . . . .  $\pm 1$

Moments of inertia, percent:

1/20-scale model

$I_x$  . . . . . -1 to 5

$I_y$  . . . . . -1 to 5

$I_z$  . . . . . -6 to 0

## 1/16-scale model

|                 |         |
|-----------------|---------|
| $I_X$ . . . . . | -3 to 3 |
| $I_Y$ . . . . . | 0 to 6  |
| $I_Z$ . . . . . | 1 to 7  |

The maximum control displacements used during the tests were  $\pm 30^\circ$  for the rudder,  $30^\circ$  up and  $20^\circ$  down for the elevator, and  $30^\circ$  up and  $17^\circ$  down for the ailerons.

## TEST CONDITIONS AND METHODS

Tests were performed with the two models representing the same equivalent full-scale conditions. The normal model loading conditions corresponded, within the limits of accuracy previously indicated, to the following full-scale mass distribution. This mass distribution is considered to be typical of a modern low-wing monoplane.

|                                        |       |
|----------------------------------------|-------|
| Weight, lb . . . . .                   | 4340  |
| $x/c$ . . . . .                        | 0.248 |
| $z/c$ . . . . .                        | 0.126 |
| $I_X$ , slug-ft <sup>2</sup> . . . . . | 2479  |
| $I_Y$ , slug-ft <sup>2</sup> . . . . . | 3876  |
| $I_Z$ , slug-ft <sup>2</sup> . . . . . | 5776  |

The model tests were performed under conditions that were equivalent to spinning the full-scale airplane at an altitude of 7000 feet.

Tests were performed on the two models to compare the effect of changing the mass distribution. The particular mass variation investigated consisted in increasing the moments of inertia  $I_Y$  and  $I_Z$  by 30 percent of  $I_Y$ . This loading was obtained on the models by extending weights along the fuselage; it is hereinafter referred to as the "modified" loading condition.

Tests were conducted to determine the effect of di-

mensional modifications on both the normal and the modified loading conditions. Two auxiliary fins of the size and location shown in figure 2 were tested independently on the two models.

Concurrent tests were run on the two models in each test condition for various control dispositions. The results of the investigation are presented in figures 3 to 8. In order to permit a direct comparison of effects due to differences in size, the steady-spin parameters presented in the figures (determined by methods described in reference 3) have been converted to the corresponding full-scale values. If each model gave a similar representation of the motion of the airplane, the results for the two models as plotted on the figures would be identical. The angle of sideslip is approximately equal to  $\phi$  minus the helix angle (angle between flight path and vertical). For the recorded spins, the helix angle averaged about  $5.5^\circ$  for both models.

Recoveries were measured by the number of turns the spinning model made from the instant the controls were observed to move until the spinning rotation ceased.

For convenience, the results are presented in two sections. The first section contains a comparison of the model results for the normal loading condition, including dimensional modification on the models; the second section presents a similar comparison of the models in the modified loading condition ( $I_y$  and  $I_z$  increased by 30 percent of  $I_y$ ). All the results are for right spins.

In several instances comparable data on the two models are lacking, particularly for spins involving upward settings of the elevators, because these spins were too difficult to hold in the tunnel.

In a comparison of the number of turns required for recovery, it should be remembered that, for an oscillatory spin, recoveries depend somewhat on the phase of the oscillation at which the controls are manipulated and that, for such spins, it is difficult to obtain consistent results. This effect may account for a difference of one-half turn or more in recovery results for oscillatory spins.

## PRECISION

The precision of the measurement made in the spin tunnel is believed to be within the following limits:

|                                                    |           |
|----------------------------------------------------|-----------|
| Velocity $V$ , percent . . . . .                   | $\pm 2$   |
| Angular velocity $\Omega$ , percent . . . . .      | $\pm 1$   |
| Angle of attack $\alpha$ , deg . . . . .           | $\pm 3$   |
| Angle of wing to horizontal $\phi$ , deg . . . . . | $\pm 2$   |
| Turns for recovery . . . . .                       | $\pm 1/4$ |

The preceding limits may be exceeded in instances where it is difficult to handle the spin in the tunnel owing to a high rate of descent or to the wandering or oscillatory nature of the spin.

## RESULTS FOR NORMAL LOADING CONDITIONS

## Normal Flying Condition (Fig. 3)

Qualitative comparison of trends indicated by each model.— In the normal loading and the normal flying conditions, both models exhibited similar characteristics. With the ailerons neutral, raising the elevator from neutral generally steepened the spins, increased the vertical velocity, slightly decreased the angular velocity, tended to lower the right (inboard) wing, and tended to improve recovery. Ailerons with the spin effected similar changes in the steady spins except that the angular velocity increased instead of decreasing. Ailerons against the spin tended to flatten the spin slightly and to produce more critical oscillatory spins. Neither model would spin steadily with the rudder neutral and no results are presented for this control setting.

Quantitative comparison of results for the two models.— A study of figure 3 reveals that the results for the two models are in general quantitative agreement in regard both to steady-spin parameters and to turns for recovery except for spins with the ailerons set full with the spin. With this aileron disposition, the 1/20-scale model spun steeper,

faster, steadier, and with its right wing from  $7^\circ$  to  $14^\circ$  higher than that of the 1/16-scale model. The large change in  $\phi$  with aileron setting should be noted. (See figs. 3(h) and 3(i).)

#### Fin 1 in Place (Fig. 4)

Qualitative effect of the fin as shown by both models.- The effect of fin 1 forward of the vertical tail was small and inconsistent. With the ailerons neutral and the elevator down, both models gave flatter spins with the added fin area. With raised elevator, however, the effect on either model was slight. Ailerons with the spin resulted in steeper spins; whereas ailerons against the spin produced more oscillatory and irregular spins. The corresponding velocities, however, did not appear to vary consistently with angle of attack.

Quantitative comparison of results for the two models.- The wandering and the oscillatory nature of the spins, particularly when the ailerons were used, makes a rigorous comparison between the two models difficult. With the ailerons neutral, however, both the steady-spin parameters and the recoveries are in fairly close agreement, excepting the velocities that accompanied the spins with the elevator  $20^\circ$  down. The tendency of the 1/20-scale model to spin with its right wing higher than that of the 1/16-scale model, when the ailerons are with the spin, should be noted. The two types of spin exhibited by the 1/20-scale model with the ailerons against the spin (fig. 4(c)) should also be observed.

#### Fin 2 in Place (Fig. 5)

Qualitative effect of the fin as shown by both models.- In general, both models indicate a favorable effect of adding area below the horizontal tail. With neutral ailerons the spins were slightly steeper and the recoveries faster, although the information on the 1/16-scale model is limited. Ailerons with the spin produced steep spins similar to those obtained without the added fin area. With the ailerons against the spin, neither model would spin consistently.

Quantitative comparison of results for the two models.- Oscillatory spins and fluctuating air speeds make

comparison of the results for the two models difficult, particularly as regards the velocity and the inclination of the wings to the horizontal. With the ailerons neutral, however, the other parameters are in good agreement. For the ailerons with the spin, the 1/20-scale model definitely spun steeper, faster, and with its right wing considerably higher ( $10^{\circ}$  to  $14^{\circ}$ ) than that of the 1/16-scale model.

#### RESULTS FOR MODIFIED LOADING CONDITIONS

( $I_y$  AND  $I_z$  INCREASED BY 30 PERCENT  $I_y$ )

Normal Flying Condition (Fig. 6)

Qualitative effect of the change in loading.— Both models were similarly affected by the change in loading. The effect of the modified load on both models was to flatten the spin, decrease the rate of descent, and decrease the rate of rotation, for all control dispositions except those involving the ailerons set with the spin. With this control disposition, the reverse effect on the angle of attack and the velocity was obtained, but both models were prone to spin with this aileron disposition when the elevators were down, even when the rudder was neutral (fig. 6(c)). Recoveries were not greatly different from those obtained in normal loading, but both models indicated a slight adverse effect of the modified loading.

Quantitative comparison of results for the two models.— Quantitatively, the results for the two models in the normal flying condition check well; the greatest discrepancies occur for the ailerons with the spin and the elevator neutral. An examination of figure 6(i) indicates that, for the ailerons with the spin, the 1/20-scale model tended to spin with its right wing higher than that of the 1/16-scale model.

Fin 1 in Place (Fig. 7)

Qualitative effect of the fin as shown by each model.— A comparison of figures 6 and 7 indicates that the detrimental effect of the added fin area was quite pronounced when the models were in the modified loading condition.

The presence of the fin caused both models to spin flatter and at a lower velocity and increased the number of turns for recovery. For the ailerons with the spin, however, the effects were not very definite.

Quantitative comparison of the results for the two models.- With the exception of the spins in which the ailerons were with the spin, the steady-spin parameters and recoveries for the models with fin 1 are in good agreement; the largest discrepancy appears in figure 7(o).

For the ailerons with the spin, elevators down, the 1/20-scale model spun flatter, at a lower air speed, and with its right wing  $5^\circ$  higher, than the 1/16-scale model. It should be observed, however, that occasionally a steeper spin was obtained with the 1/20-scale model, but no quantitative data could be secured (fig. 7(c)).

#### Fin 2 in Place (Fig. 8)

Qualitative effects of the fin as shown by each model.- A comparison of figure 5 (normal load) and figure 8 (modified load) reveals that, with the additional fin in place, the effect of the modified loading on both models was, in general, an increase in angle of attack, a decrease in vertical velocity, a decrease in angular velocity, and an increase in turns for recovery, for all control dispositions not involving ailerons with the spin. For the ailerons with the spin, the modified loading appeared favorable.

A comparison of figures 6 and 8 indicates that, for the models with the modified loading, the addition of the auxiliary fin below the fuselage tended to increase the rate of descent but had little other effect.

Quantitative comparison of the results of the two models.- With the ailerons either neutral or against the spin, the 1/20-scale model spun slightly flatter than the 1/16-scale model for all elevator settings, but the differences in the other parameters were small. For the ailerons with the spin, a comparison can be made only for the elevator-down spins. With this control disposition, the velocity of the 1/20-scale model was greater and its right wing was a few degrees higher than that of the 1/16-scale model.

## DISCUSSION

## Reynolds Number Range Covered by Investigation

The relationship between the test Reynolds number of a dynamically similar scale model tested in air of normal density and the Reynolds number of the full-scale airplane can be expressed as follows:

$$R_M = R_A N^{3/2}$$

For the 1/20- and the 1/16-scale models used in these experiments, the foregoing relationship becomes

$$R \text{ for } 1/20\text{-scale model} = R_A (1/20)^{3/2} = 0.011 R_A$$

$$R \text{ for } 1/16\text{-scale model} = R_A (1/16)^{3/2} = 0.0156 R_A$$

The range of Reynolds numbers investigated - based on the mean wing chord, a mean value of the kinematic viscosity of 0.000165 foot<sup>2</sup> per second, and the measured rates of descent - is tabulated below:

|         | Test       | Model R | Corresponding full-scale R |
|---------|------------|---------|----------------------------|
| Minimum | 1/20 model | 62,500  | 5,680,000                  |
|         | 1/16 model | 91,400  | 5,850,000                  |
| Maximum | 1/20 model | 113,500 | 10,280,000                 |
|         | 1/16 model | 148,000 | 9,480,000                  |

Because of the turbulence in the tunnel, the effective Reynolds number is greater than the Reynolds number of the test model by a factor 1.8 (reference 4). The effective test Reynolds number thus ranged from 112,500 (for the 1/20-scale model) to 266,400 (for the 1/16-scale model).

## Correlation between Results for the Two Models

On the basis of the information contained in figures 3 to 8, the following conclusions have been reached:

1. The same qualitative effects of control disposition, mass distribution, and dimensional modifications were indicated for the two models.



2. The most difficult spins to correlate were those involving aileron deflections. When the ailerons were with the spin, the 1/20-scale model generally spun steeper in the normal loading condition than the 1/16-scale model. In the modified loading condition, although there was generally little difference in results for the two models, spins were obtained for which the reverse was true. For the ailerons against the spin, there existed a tendency for the 1/16-scale model to spin steeper than the 1/20-scale model, regardless of the mass distribution.

3. All of the steady-spin parameters were in fair agreement with the exception of the angle of the wing to the horizontal, which varied considerably for the two models, particularly when the ailerons were used. In general, when the ailerons were with the spin, the 1/20-scale model tended to spin with the right wing higher than that of the 1/16-scale model, that is, with more outward sideslip. (It will be observed in going from the larger model to the smaller model that the change in angle of sideslip was in the same direction as that found in going from full-scale data to model data in reference 2.)

4. The size of the model had little influence on the number of turns for recovery, even for spins in which the angle of the wing to the horizontal was noticeably different for the two models. The relationship existing between the angle  $\phi$  and the number of turns for recovery is exceedingly complex and, consequently, the significance of the aforementioned result is not completely understood. From a practical point of view, the number of turns for recovery is usually considered to be the most important parameter of the motion insofar as the correlation of model results and full-scale results is concerned.

#### Comparison with Flight Results

Spin-test results of the full-scale airplane represented by the two models are presented in reference 5. Unfortunately, the control settings used in these full-scale tests are not the same as those used on the models in this investigation, and therefore a rigorous comparison cannot be made. A qualitative comparison, however, seems to indicate that the effect of scale is of much greater significance when the results for either model are compared with the full-scale results than when the results for either model are compared with the results

for the other. It would therefore appear that, within the range of the model sizes investigated, such factors as difficulty of construction and testing are more important in determining proper model size than are the changes in scale effect likely to exist between extreme sizes feasible for test in the 15-foot tunnel.

### Comparison with Other Results

The investigation reported in reference 1 included a comparison of rolling-balance measurements made in a 7-foot vertical tunnel on a 1/10- and on a 1/17.5-scale model of a British fighter airplane. The resultant aerodynamic moments about the spinning axis for several rates of rotation were measured on both models for a single angle of attack ( $37.9^\circ$ ). The rates of rotation were measured by the quantity  $\Omega b/2V$  and the values of this parameter ranged from 0.3 to 0.6. Similar measurements were made on the 1/17.5-scale model in a 4-foot tunnel to determine the effect of tunnel size. The tunnel effect was found to be small. The sets of measurements made in the 7-foot tunnel agreed closely with each other, but the results for either model disagreed considerably with the corresponding results for the full-scale airplane. It will be observed that this effect of scale is consistent with the comparison of the results of the present investigation with the full-scale results of reference 5.

The results in reference 2 indicate that, within the range of Reynolds numbers tested (of the same order of magnitude as the tests of the present investigation), the scale effect was negligible.

### Suggestions for Future Research

In this investigation the actual difference in the size of the models used did not completely cover the greatest range of sizes likely to be encountered in spin-tunnel test work. It would therefore appear advisable to supplement the present investigation with data representative of a much greater variation in model size.

The model-recovery results in this investigation were not particularly sensitive to the modifications tried. It is suggested that, in future investigations, modifications be tested that markedly affect the recovery characteristics of the models.

## CONCLUSIONS

On the basis of the results obtained in the investigation, the following conclusions can be drawn:

1. Qualitatively, the same characteristic effect of control disposition, mass distribution, and dimensional modifications were indicated for both models.

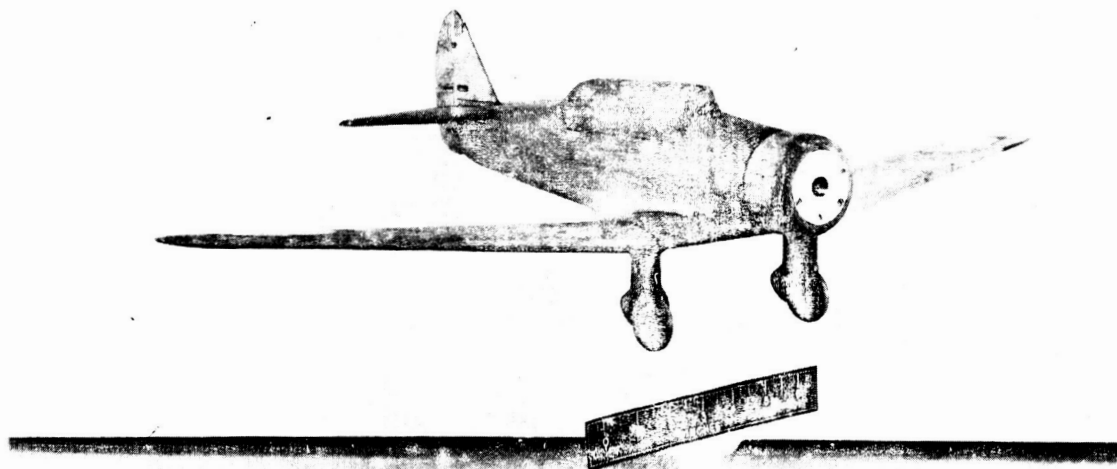
2. The number of turns for recovery, probably the most important parameter of the spin for practical purposes, were in good agreement for both models.

3. It would appear that, for the 15-foot tunnel, such factors as difficulty of construction and testing are more important in determining proper model size than are the changes in scale effect likely to exist between the different sizes of models that are practicable for the 15-foot spin tunnel. This conclusion is based entirely on the results presented in this report. The investigation should be extended to include a greater range of model sizes and more extreme modifications.

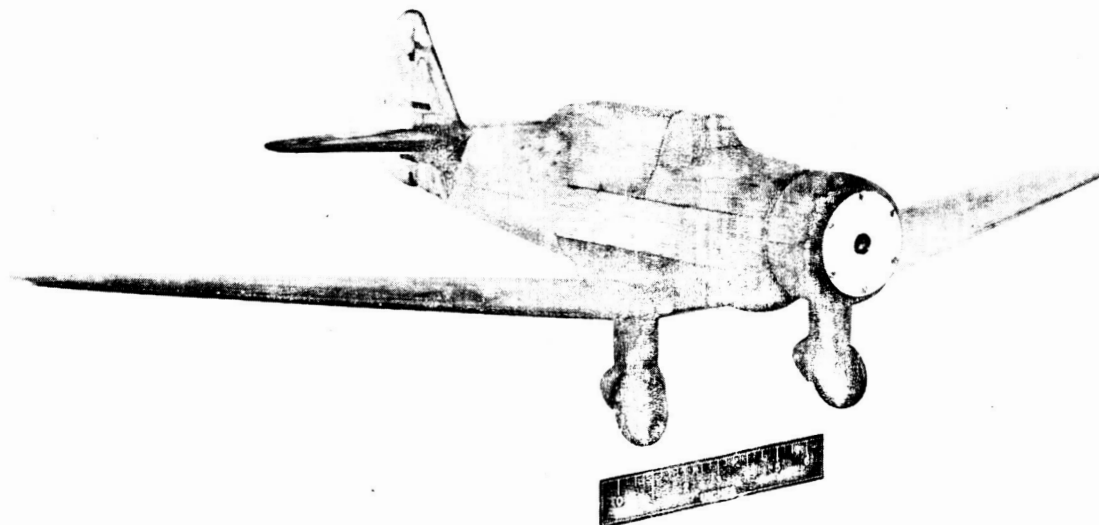
Langley Memorial Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., April 16, 1941.

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4. Platt, Robert C.: Turbulence Factors of N.A.C.A. Wind Tunnels as Determined by Sphere Tests. Rep. No. 558, NACA, 1936.
5. Seidman, Oscar, and McAvoy, William H.: Spin Tests of a Low-Wing Monoplane in Flight and in the Free-Spinning Wind Tunnel. T.N. No. 769, NACA, 1940.



(a) The 1/20 scale model.



(b) the 1/16 scale model.

Figure 1.- Photographs of the two models used in the investigation.

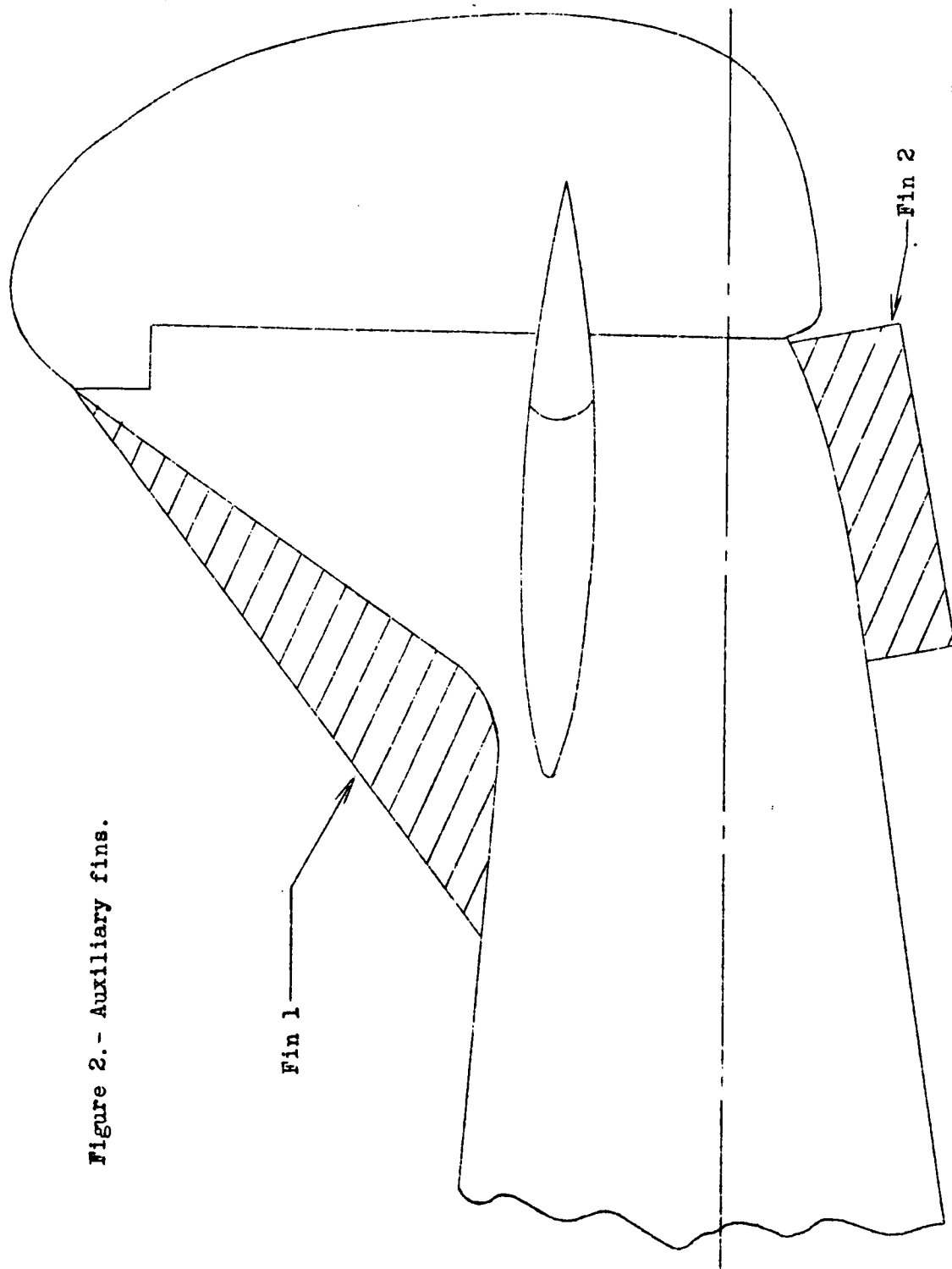


Figure 2.- Auxiliary fins.

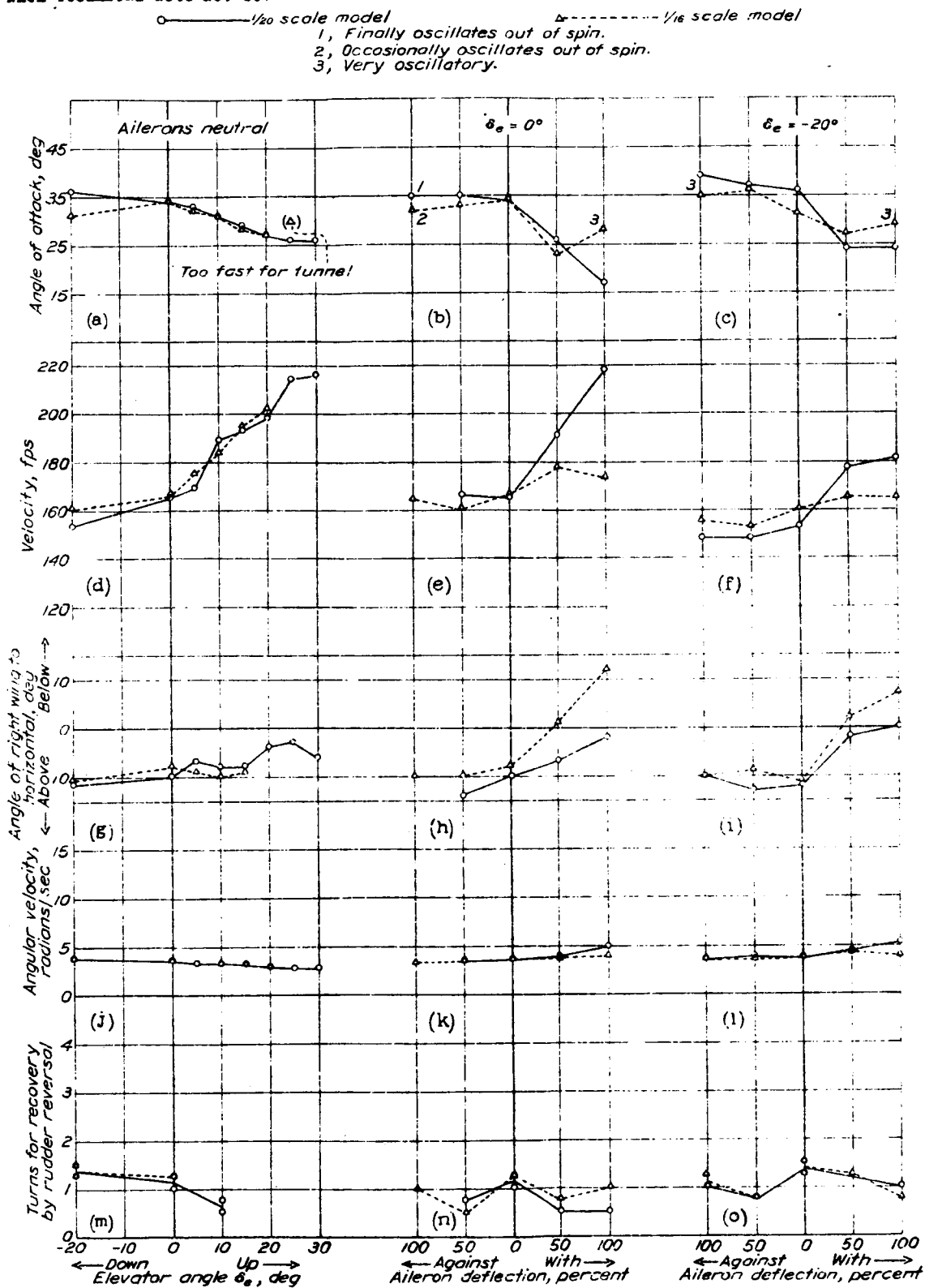


Figure 3.- Effect of scale on two models in the NACA 15-foot free-spinning wind tunnel. Normal load; rudder  $30^\circ$  with spin; right spine.

- ——— 1/20 scale model  
 Δ ——— 1/16 scale model  
 1, Oscillatory, in 1/20-scale model checks, the model would recover of its own accord when forced to spin.  
 2, Model gradually steepens and recovers.  
 3, Note two types of spin here.  
 4, Depends on oscillation present when controls move.  
 5, Model would recover of its own accord when forced to spin.

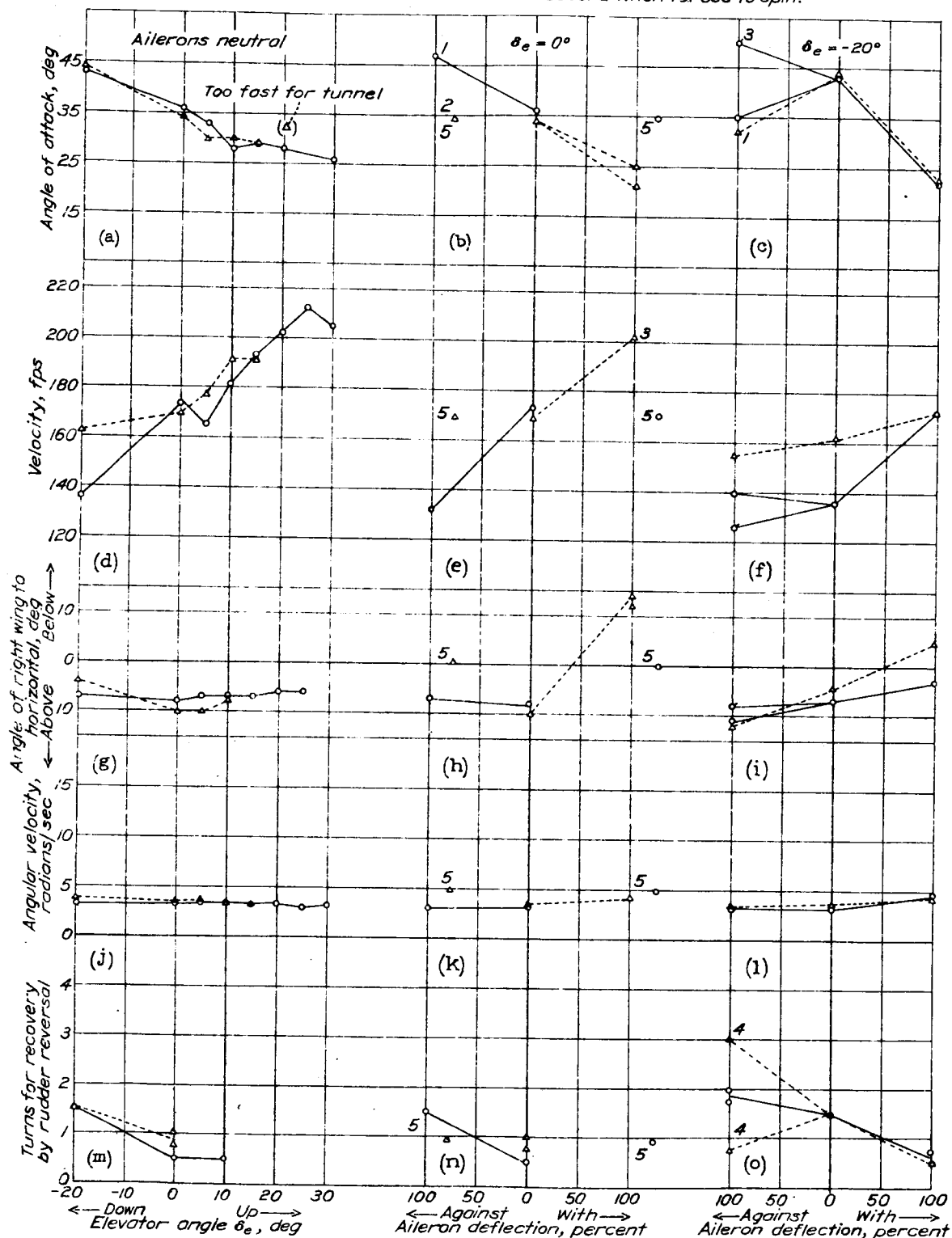


Figure 4.- Effect of scale on two models in the NACA 15-foot free-spinning wind tunnel.  
 Normal load; auxiliary fin 1 in place; rudder  $30^\circ$  with spin; right spins.



○ ———  $1/20$  scale model△ - - - - -  $1/16$  scale model

1, Oscillatory spin.

2, In  $1/20$ -model checks, the model would recover of its own accord when forced to spin.

3, Air speed fluctuates.

4, Model would recover of its own accord when forced to spin.

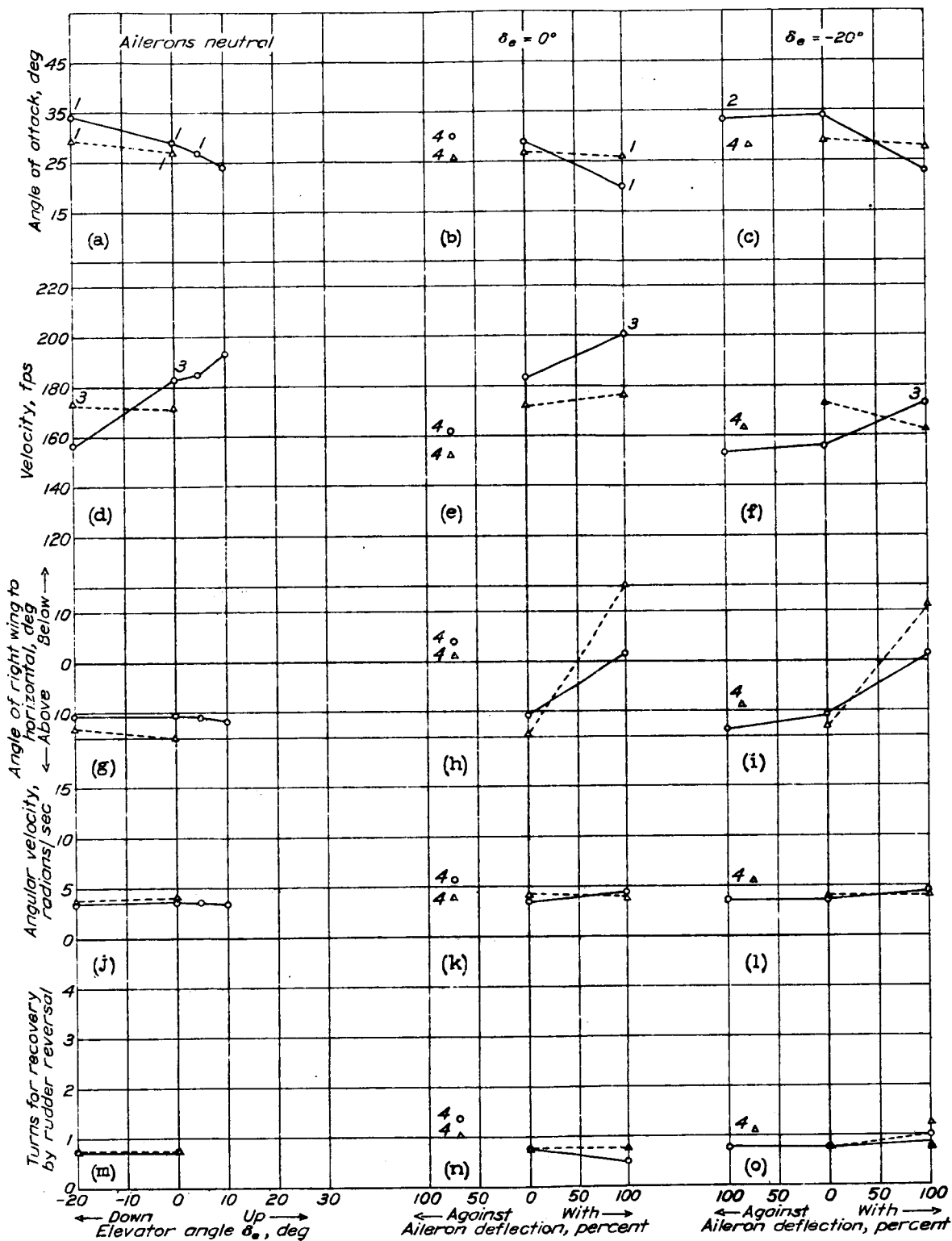


Figure 5.- Effect of scale on two models in the 15-foot free-spinning wind tunnel.

Normal load; auxiliary fin 2 in place; rudder  $30^\circ$  with spin; right spins.

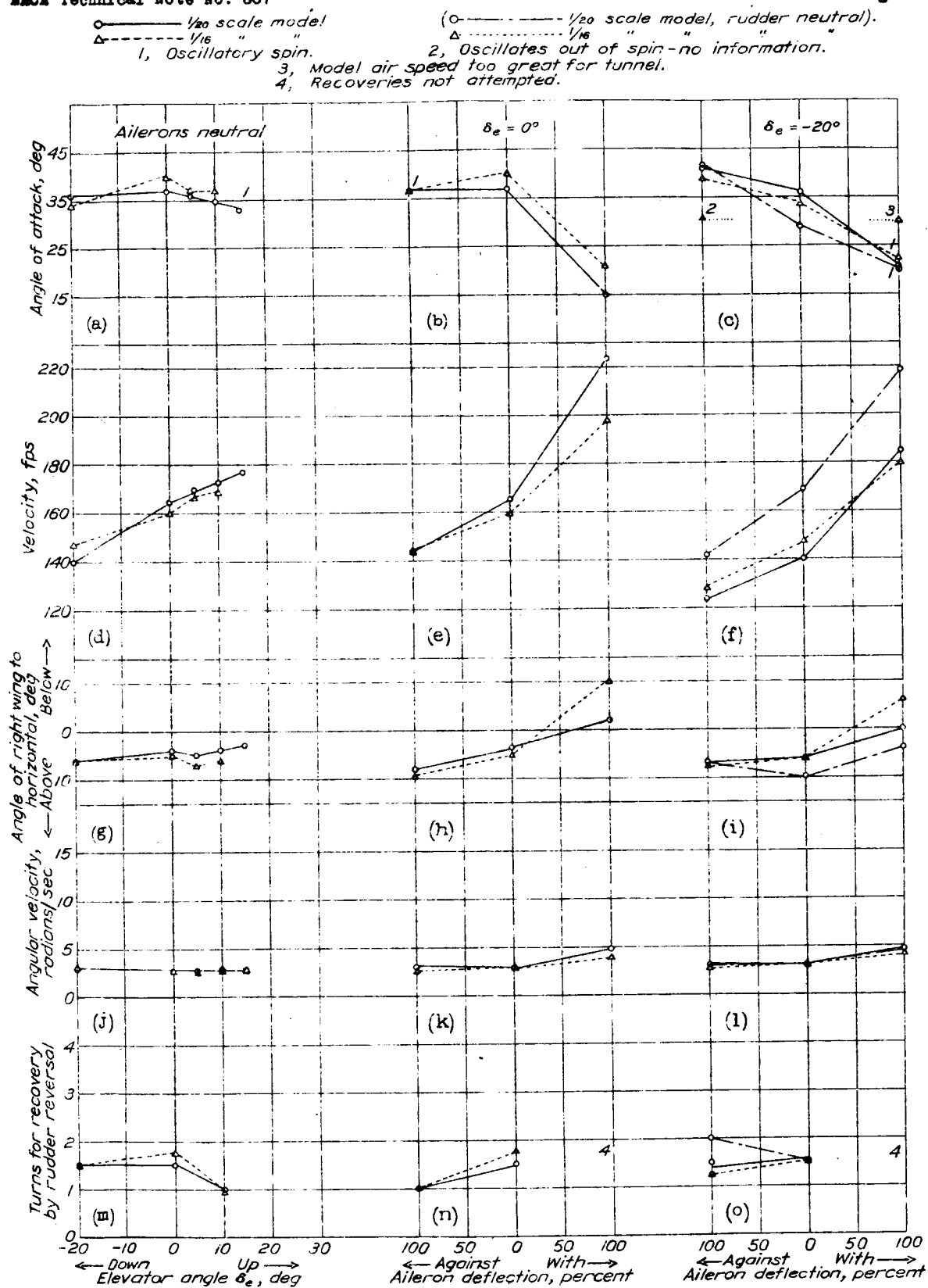


Figure 8.— Effect of scale on two models in the NACA 15-foot free-spinning wind tunnel. Modified load( $I_y$  and  $I_z$  increased 30 percent  $I_y$ ); rudder  $30^\circ$  with spin; right spins.

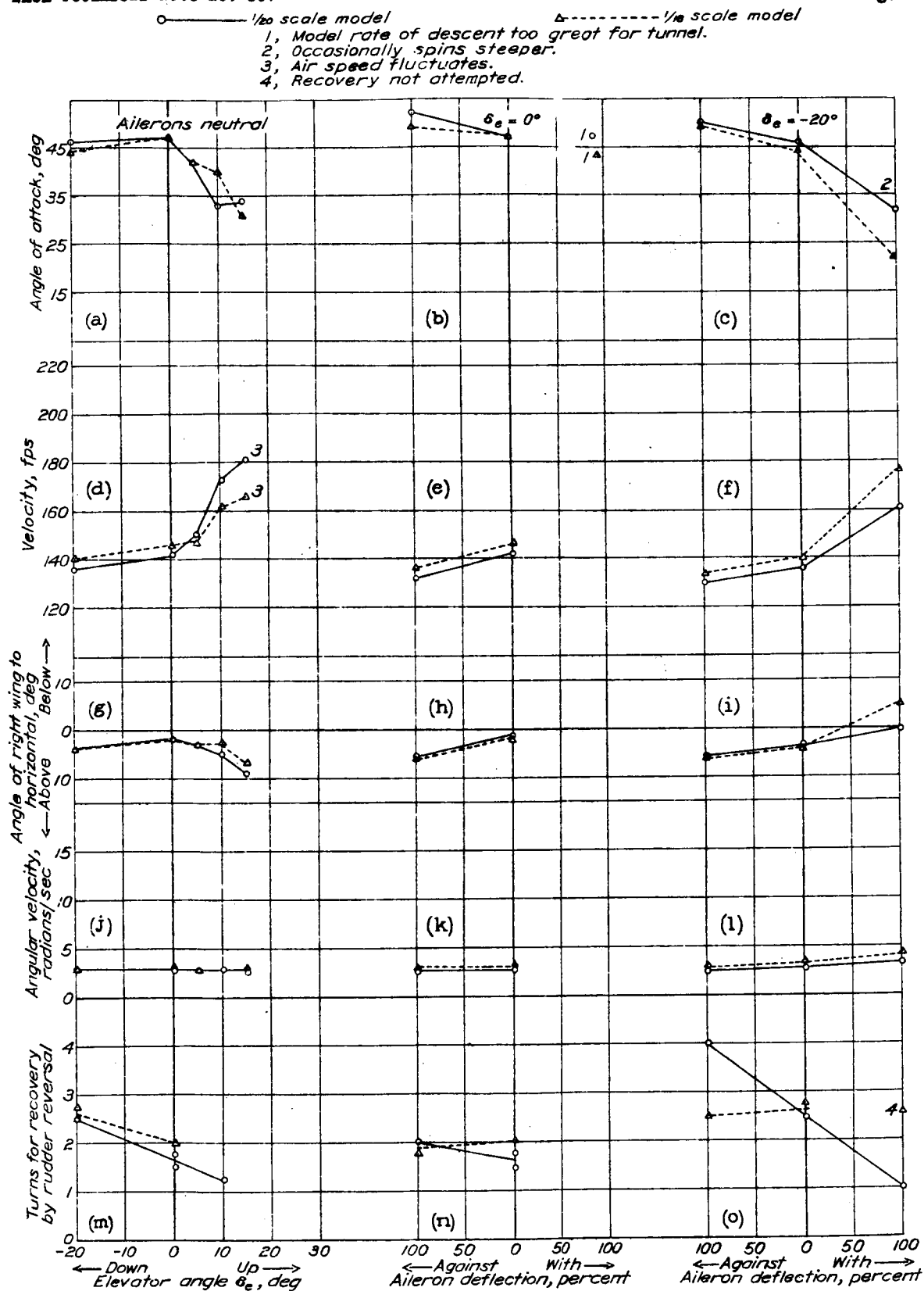


Figure 7.— Effect of scale on two models in the NACA 15-foot free-spinning wind tunnel. Modified load ( $I_y$  and  $I_z$  increased 30 percent  $I_y$ ); auxiliary fin 1 in place; rudder  $30^\circ$  with spin; right spins.

○ — 1/20 scale model  
 △ — 1/6 scale model  
 1, Model rate of descent too great for tunnel.  
 2, Oscillatory spin.  
 3, Recovery not attempted.

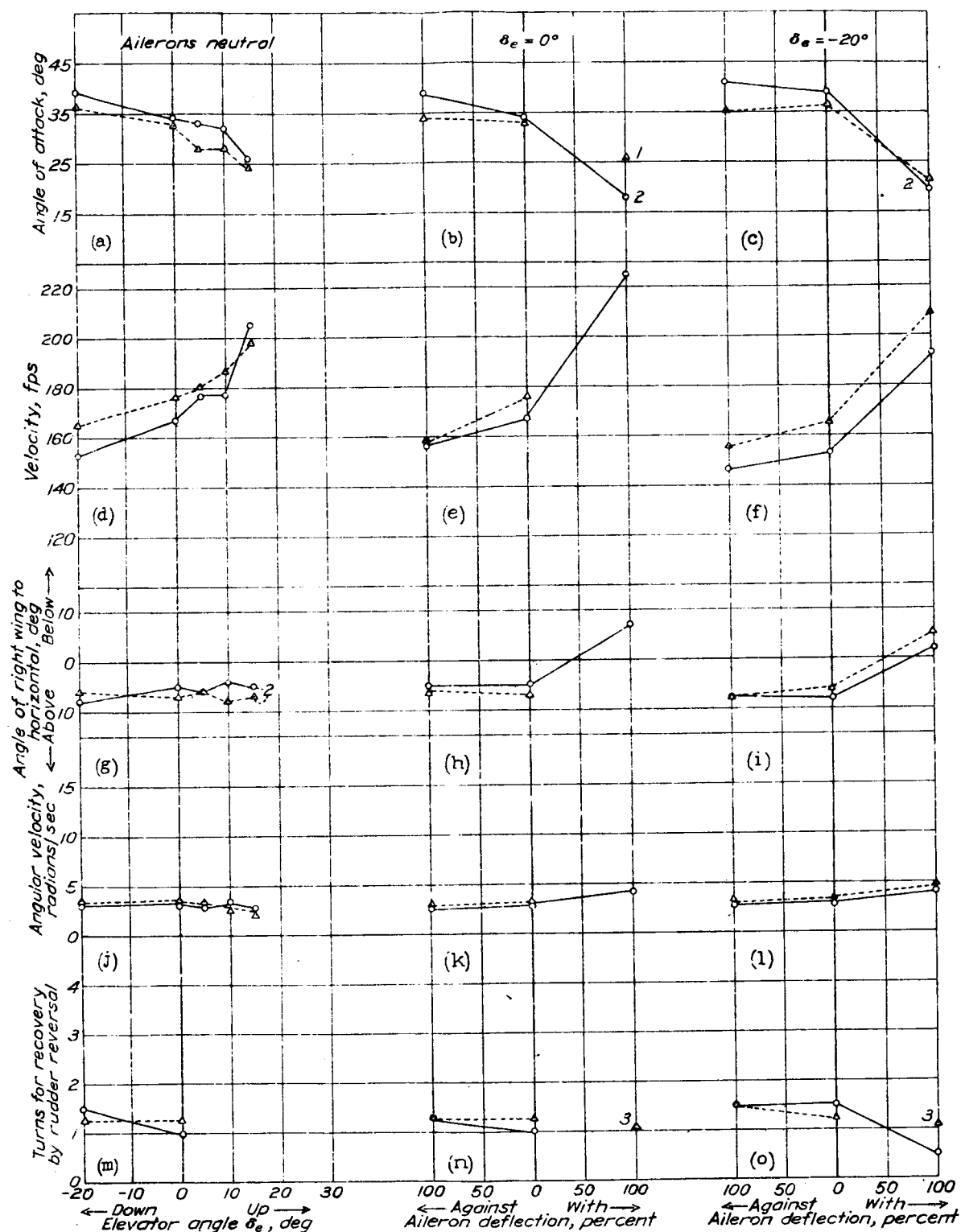


Figure 8.- Effect of scale on two models in the NACA 15-foot free-spinning wind tunnel. Modified load ( $I_y$  and  $I_z$  increased 30 percent  $I_y$ ); auxiliary fin 2 in place; rudder  $30^\circ$  with spin; right spins.

# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## TECHNICAL NOTE NO. 828

### METHODS OF ANALYZING WIND-TUNNEL DATA

#### FOR DYNAMIC FLIGHT CONDITIONS

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#### SUMMARY

The effects of power on the stability and the control characteristics of an airplane are discussed and methods of analysis are given for evaluating certain dynamic characteristics of the airplane that are not directly discernible from wind-tunnel tests alone. Data are presented to show how the characteristics of a model tested in a wind tunnel are affected by power.

The response of an airplane to a rolling and a yawing disturbance is discussed, particularly in regard to changes in wing dihedral and fin area. Solutions of the lateral equations of motion are given in a form suitable for direct computations. An approximate formula is developed that permits the rapid estimation of the accelerations produced during pull-up maneuvers involving abrupt elevator deflections.

#### INTRODUCTION

Some time ago, the NACA undertook an investigation to determine the flying qualities of a low-wing, pursuit monoplane. This airplane (fig. 1), like many of its type, was found to possess several undesirable flying characteristics. Accordingly, a number of modifications to the airplane were recommended. In order to study the effects of the proposed modifications and in order to investigate the reliability of wind-tunnel data for estimating the behavior of the full-scale airplane, a scale model of the airplane, with and without the proposed modifications, was tested in a wind tunnel. Tests were made with and without the propeller operating in order to study the effect of power on the characteristics of the model.

During the course of these wind-tunnel tests, several interesting results were obtained concerning the effects of power on the characteristics exhibited by the model. In the evaluation of the data, certain methods of analysis were employed that were found useful in estimating the dynamic behavior of the airplane, particularly as regards the effects of the proposed modifications. These subjects are treated in the present paper. It is felt that this paper will serve as an aid to other wind-tunnel investigators seeking to evaluate the flight characteristics of an airplane from wind-tunnel results.

The first portion of the paper is concerned with the effect of power on certain of the wind-tunnel characteristics exhibited by a model. Graphs are presented that emphasize the importance of obtaining wind-tunnel data with powered models. The second portion of the paper discusses the lateral motions of the airplane. Concrete applications are included showing how the solutions to the lateral equations of motion were used to estimate the change in dynamic flight characteristics likely to result from the use of the recommended modifications to the airplane. Finally, the longitudinal motion of the airplane is considered. An approximate formula is developed that was found useful in estimating the accelerations and the stick forces likely to develop in an airplane during an abrupt pull-up maneuver.

#### THE IMPORTANCE OF POWERED MODELS IN WIND-TUNNEL TESTS

Although the influence of the slipstream on the characteristics of an airplane has been appreciated for many years, wind-tunnel tests with propellers operating have not been the usual procedure. The frequency with which undesirable flying characteristics appear in modern airplanes has led to some apprehension concerning the reliability of wind-tunnel data obtained without powered models. The particular testing technique employed in the present investigation will be reserved for detailed discussion in a future paper. It will suffice to say here that the power-on test procedure should be such that data may be obtained for thrust conditions of the model corresponding to those of the full-scale airplane at the lift coefficient and the flight attitude under consideration.

Figure 2 is a plot of the elevator angle required for trim against the indicated air speed at which trim occurs. These curves were evaluated from data secured from (1) wind-tunnel tests with no propeller on the model, (2) actual flight tests with the full-scale airplane, and (3) wind-tunnel tests with a powered model. The shapes of the curves are indices of the static stability (see later discussion of longitudinal motion) displayed by the airplane in flight and by the models in the wind tunnel. The agreement between the flight results and the results obtained with a powered model is good, particularly for the flap-up condition. The larger discrepancy existing for the flap-down condition is partly attributable to the rather large tab deflection ( $11^\circ$  nose up) used on the full-scale airplane for this test. The related variation of the elevator stick force required for trim with indicated air speed is not presented here because the model data and the flight data were not directly comparable owing to the absence of a trim tab on the model. In the evaluation of the information obtained from wind-tunnel tests, certain methods of presenting the data have been found to facilitate the analysis. The resulting pitching moment acting on an airplane trimmed for steady equilibrium flight is zero. This condition is true whether or not the airplane is flying yawed. As far as the pilot is concerned, the dihedral effect of the wing manifests itself as a pure rolling moment about an axis lying in the plane of symmetry of the airplane. In wind-tunnel investigations, on the other hand, the rolling moment is usually measured about an axis coincident with the tunnel axis. Consequently, when the model is yawed relative to the tunnel axis, the measured rolling moment about this axis will contain a component of the pitching moment acting on the model unless the model is trimmed for zero pitching moment. In the comparison of the slopes of different rolling-moment curves obtained in yaw tests, it is therefore necessary to know whether the measurements were made with the model trimmed in pitch.

If the system of axes discussed in the subsequent section on lateral motion is employed for the presentation of wind-tunnel data, the necessity for continually checking the trim of the model during yaw tests is avoided. In addition, the data in this form can be used directly in the evaluation of the stability derivatives necessary for the solutions of the equations of motion.

In figure 3 are plotted for various power conditions the rates of change with angle of yaw of the yawing-moment coefficient  $dC_N/d\psi$ , the rolling-moment coefficient  $dC_l/d\psi$ , and the lateral-force coefficient  $dC_Y/d\psi$ . The decrease in dihedral effect  $dC_l/d\psi$  with the application of power should be observed. In the condition presented (flap up), the decrease in dihedral effect with the application of power is not great, but, in the flap-down condition and at high lift coefficients, the effect of power may be critical.

An inspection of figure 4 will suffice to illustrate the effect of the slipstream on the characteristics of a control surface placed within its boundaries. The negative yawing moment that exists with neutral rudder when the propeller is operating should be observed. The moment is apparently caused by the rotation of the slipstream. It is, however, recognized that the slipstream rotational characteristics exhibited by the model in this test are probably not identical with the slipstream rotational characteristics that would be obtained in flight with the full-scale airplane, even under similar thrust conditions. The results do indicate, nevertheless, that with the present trend toward greater and greater power, the effect of this slipstream rotation may be a critical factor in tail design.

Figures 2, 3, and 4 have been included to illustrate the effects that power may have on the stability characteristics of an airplane and on the effectiveness of the horizontal and the vertical tail surfaces. These effects are difficult to estimate and consequently recourse must be had to wind-tunnel tests of a model equipped with running propellers. Wind-tunnel tests of a powered model require more care and are more expensive to perform than tests of the conventional type of model. By a judicious choice of tests, however, it is possible to secure the information of greatest value with a minimum of time and expenditure.

#### THE LATERAL MOTIONS OF THE AIRPLANE

Certain lateral-stability and control characteristics of an airplane can be directly determined from wind-tunnel tests of a model. These characteristics usually include



(1) the directional stability (fig. 3(a)), (2) the dihedral effect (fig. 3(b)), and (3) the static effectiveness of the control surfaces (fig. 4). This information is indispensable to the designer if he is to proportion his airplane properly and is also necessary for the evaluation of the stability derivatives from which the dynamic behavior of the airplane may be predicted.

The dynamic flight characteristics that an airplane will display are not, however, readily obtained. The behavior of an airplane subjected to a lateral disturbance is complicated by the coupling of the ensuing rolling and yawing motions. The coupling of these two motions makes it impossible to estimate the effectiveness of the ailerons and the rudder in flight from static tests of the controls alone. A sound evaluation of these factors can be made only if a knowledge of the motions produced by the controls is available.

The angle of bank produced, for example, in 1 second by the total deflection of the ailerons might be taken as a measure of the effectiveness of the ailerons in producing a sudden roll. If, on the other hand, the airplane is rolled slowly, a greater yawing motion being permitted to develop, the result may be to produce a rolling motion opposed to that generated by the ailerons and of such magnitude that the effect of the ailerons is completely nullified. The airplane may even roll against the ailerons.

In actual flight the pilot might counteract this adverse rolling tendency by coordinating the rudder with the ailerons. In the present analysis, however, only the inherent dynamic characteristics of the airplane are considered. It would be of interest, then, to know the variation of the angle of bank with time after small aileron deflections are applied. For similar reasons, the effectiveness and the sensitivity of the rudder can be most carefully judged if the angle of bank and the angle of yaw produced in a definite time interval by a small rudder deflection are known. Information of this type is particularly valuable in estimating the effect of various combinations of vertical fin area and wing dihedral on the dynamic behavior of an airplane.

Apart from actual flight tests, probably the best method of estimating the dynamic qualities of an airplane is by evaluating the dynamic equations of motion, the data obtained from wind-tunnel tests of a powered model being used.

Assumptions and symbols.- The assumptions generally made in the study of airplane stability are made here. The most important of these assumptions are:

1. The air forces and the moments resulting from displacements of the airplane relative to its steady condition of flight are proportional to the displacements or to their rates of change.
2. The components of moment due to different components of the motion are directly additive. (For example, the rolling moment due to combined rolling and sideslipping may be computed as though the rolling and the sideslipping had occurred separately.)

The axes used in specifying the moments, the angular velocities, and so forth are fixed in the airplane and move relative to the earth and the air. The X axis, passing through the center of gravity of the airplane, is in the plane of symmetry and is so oriented that it points into the relative wind when the airplane is flying steadily in unyawed flight. Also, the axes form a conventional orthogonal system intersecting at the center of gravity. The Z axis points directly downward in the plane of symmetry and the Y axis points along the direction of the right wing. The motions discussed are those of the moving axes relative to the undisturbed air with the exception of the angle of bank, which is measured from the horizontal.

The symbols used in the following analysis are defined in the appendix.

Although the axes change their orientation in the airplane with different lift coefficients and probably never coincide with the axes of the principal moments of inertia, the corrections in unstalled flight are small and have been neglected, as have the products of inertia.

Equations of motion.- If the airplane is considered capable of motion in all degrees of lateral freedom, the equations of motion with deflected controls (neglecting the small side forces developed by the deflected controls

and by the rolling and the yawing velocities) may be written:

$$\left. \begin{aligned} \frac{dp}{dt} &= pL_p + rL_r + \beta L_\beta + \delta L_\delta \quad (\text{in rolling}) \\ \frac{dr}{dt} &= pN_p + rN_r + \beta N_\beta + \delta N_\delta \quad (\text{in yawing}) \\ \frac{d\beta}{dt} &= \frac{g\phi}{U_0} - r + \frac{\beta Y_\beta}{U_0} \quad (\text{in sideslipping}) \end{aligned} \right\} \quad (1)$$

Also

$$\frac{d\phi}{dt} = p; \quad \frac{d\psi}{dt} = r; \quad \beta = \frac{v}{U_0}$$

In order to solve for any of the variables, it is necessary to integrate this system of linear simultaneous equations. For reasons that will be apparent later, it is convenient to solve the system of equations (1) for separate unit magnitudes of the control disturbance terms,  $\delta L_\delta$  and  $\delta N_\delta$ ; that is, one set of solutions is obtained by letting  $\delta L_\delta = 1$  and  $\delta N_\delta = 0$ , and another set of solutions is obtained by letting  $\delta L_\delta = 0$  and  $\delta N_\delta = 1$ . The unit disturbances are assumed to be instantly applied at zero time and to remain constant thereafter. In order to distinguish the separate solutions, the subscript L will be applied to the solutions obtained when  $\delta L_\delta = 1$  and  $\delta N_\delta = 0$ ; whereas the subscript N will be applied to solutions obtained by letting  $\delta L_\delta = 0$  and  $\delta N_\delta = 1$ . Thus,  $p_L$  represents the rolling velocity resulting from the application of a pure rolling disturbance of unit magnitude.

If the symbol D is substituted for  $d/dt$  and if  $\delta L_\delta = 1$  and  $\delta N_\delta = 0$ , equations (1) may be rewritten in the following form:

$$\left. \begin{aligned} p(D-L_p) - rL_r - \beta L_\beta &= 1 \\ p(-N_p) + r(D-N_r) - \beta N_\beta &= 0 \\ p\left(-\frac{K}{U_0}\right) + Dr + D\left(D - \frac{Y_\beta}{U_0}\right)\beta &= 0 \end{aligned} \right\} \quad (2)$$

It can be shown that the solutions of equation (2) are of the form:

$$p_L = p_{L_0} + p_{L_1} e^{\lambda_1 t} + p_{L_2} e^{\lambda_2 t} + p_{L_3} e^{\lambda_3 t} + p_{L_4} e^{\lambda_4 t} \quad (3)$$

where

$p_L$  resultant rolling velocity due to unit rolling disturbance, that is,  $\delta L_\beta = 1$

$p_{L_0} \dots p_{L_4}$  constants that depend only on values of stability derivatives and on type of disturbance involved

$\lambda_1 \dots \lambda_4$  roots of stability equation  $F(D) = 0$ .

where

$$F(D) = \begin{vmatrix} D-L_p & -L_r & -L_\beta \\ -N_p & D-N_r & -N_\beta \\ -\frac{K}{U_0} & D & D\left(D - \frac{Y_\beta}{U_0}\right) \end{vmatrix}$$

The constants in equation (3), together with the constants appearing in the expressions for all the remaining components of motion (including the solutions for the unit yawing disturbance  $\delta N_\beta = 1$ ), have been evaluated in terms of the stability derivatives and the roots of the stability equation. The solutions of equation (2) are tabulated in

the appendix in a form suitable for computation. The stability equation is also discussed and an alternative form of equation (3) is given for use when the solution of  $F(D) = 0$  includes conjugate complex roots of the form  $a \pm ib$ .

The expressions presented in the appendix were evaluated by applying the operational mathematics of Heaviside to equation (2). For the theory of operational methods the reader is referred to a standard text on the subject, for example, reference 1. Specific applications of the Heaviside treatment to other problems in airplane dynamics may be found in reference 2.

After the complete unit solutions have been obtained, time histories of the motion caused by the unit disturbances can be plotted with very little additional calculation. Because of the linearity of the equations of motion, the unit solutions may be compounded in any arbitrary manner. If, for example,  $\delta_a L_{\delta_a}$  represents the rolling acceleration created by the applied aileron rolling moment and  $\delta_a N_{\delta_a}$  represents the accompanying yawing acceleration, then, at time  $t$ ,

$$\phi_t = \delta_a L_{\delta_a} (\phi_L)_t + \delta_a N_{\delta_a} (\phi_N)_t \quad (4)$$

where

$\phi_t$  resultant angle of bank after  $t$  seconds

$(\phi_L)_t$  angle of bank after  $t$  seconds due to a unit rolling disturbance

$(\phi_N)_t$  angle of bank after  $t$  seconds due to a unit yawing disturbance

As in the case of the unit disturbance terms, the actual disturbances,  $\delta_a L_{\delta_a}$  and  $\delta_a N_{\delta_a}$ , are assumed to be suddenly applied at zero time and to remain constant thereafter.

Applications.- The practicability of using the solutions to the lateral equations of motion is best illustrated by concrete applications.

The airplane under consideration, as mentioned in the introduction, exhibited certain undesirable flying characteristics in flight tests: The dihedral effect was undesirably low at slow speeds with the flaps down (fig. 3(b) indicates satisfactory dihedral characteristics with the flaps up over the range considered), and it was impossible to raise a wing by use of the rudder alone. In an effort to improve the lateral flying qualities of the airplane, it was proposed to increase the wing dihedral. Because the lateral characteristics of an airplane depend not only on the absolute amount of dihedral but also on the relative amount of weathercock stability present, it was considered necessary to increase the vertical tail area as well as the wing dihedral. The ailerons were unchanged, but the rudder was so modified as to improve its hinge-moment characteristics. These modifications are shown on figure 1.

Wind-tunnel tests were made with a model equipped for power-on tests with and without the proposed modifications. The data from the comparative tests indicated that considerable improvement should result from the incorporation of the modifications on the full-scale airplane. For all flap and power conditions the dihedral effect  $dC_l/d\psi$  remained positive, the index of weathercock stability  $dC_n/d\psi$  remained negative, and the static characteristics of the rudder appeared satisfactory. Wind-tunnel tests of a model, however, provide no direct information pertaining to the dynamic flight characteristics of the full-scale airplane. Instances have occurred in which the incorporation of similar modifications on a full-scale airplane has affected the control characteristics of the airplane in an adverse manner, particularly at high speeds, in spite of the favorable static characteristics indicated from wind-tunnel tests. Accordingly, it was decided to investigate the response of the airplane, with and without the modifications to the aileron and the rudder controls, by evaluating the solutions to the equations of motion.

Aileron control.- In the particular problem considered here, undesirable aileron control characteristics are likely to be manifested in the form of aileron "heaviness" or

stiffness at high speeds. The subsequent analysis, however, is perfectly general and is in no way limited to this high-speed condition. This particular condition is treated, merely as an example, to indicate the general method of procedure. Because the ailerons are identical on both the original and the modified airplanes, aileron heaviness can be physically interpreted as an increase in stick force resulting from the increased aileron deflection necessary to reproduce a given rolling maneuver with the modified airplane. This interpretation of aileron heaviness suggested the following method of analysis:

(a) On the assumption that the original airplane (airplane A) was flying in steady high-speed flight, the angle of bank generated in 5 seconds by a small aileron deflection  $\delta_a$  was computed.

(b) Then, the aileron deflection  $\delta_a + \Delta\delta_a$  necessary to reproduce the identical maneuver with the modified airplane (airplane B) was calculated.

The time for the maneuver (5 sec in this case) is somewhat arbitrary. It should be of sufficient duration, however, to permit full development of the secondary rolling effects introduced by the induced yawing motion.

The magnitude of the increment  $\Delta\delta_a$  is a direct measure of the additional stick force required to perform the maneuver (because the ailerons are identical) and may be used as an index of aileron heaviness. If  $\Delta\delta_a$  is large and positive, the aileron stick forces on airplane B may be too large to be acceptable and modification of the aileron itself may prove desirable.

The stability derivatives of airplanes A and B were evaluated from the power-on wind-tunnel tests of the two models and from data in references 3 and 4. These derivatives are tabulated in table I, together with other information necessary to evaluate the solutions to the equations of motion.

In accordance with equation (4), the angle of bank assumed by airplane A in  $t$  seconds after the application of an aileron deflection  $\delta_a$  is

$$\phi_t = \delta_a L \delta_a (\phi_L)_t + \delta_a N \delta_a (\phi_N)_t$$

In order to evaluate the unit solutions  $\phi_L$  and  $\phi_N$ , it is necessary to obtain the roots of the stability equation  $F(D) = 0$ . If the appropriate values from table I are substituted into the expression for  $F(D)$  given in the appendix,

$$F(D) = D^4 + 20.4555D^3 + 52.7884D^2 + 347.8242D + 5.43760 = 0$$

With the use of the procedure outlined in the appendix for solving quartic equations, the roots of this equation were determined to be:

$$\lambda_1 = -0.01567$$

$$\lambda_2 = -18.6230$$

$$\lambda_3 = -0.908424 + 4.21991$$

$$\lambda_4 = -0.908424 - 4.21991$$

Substitution of the appropriate roots and derivatives in the expression for the angle of bank  $\phi_L$ , given in the appendix, yields

$$\begin{aligned} \phi_L = & 3.3471 - 3.3497e^{-0.01567t} + 0.002876e^{-18.623t} \\ & + \left( \frac{-0.46778 + 0.16310931}{1741.79 - 2270.451} \right) e^{(-0.9084 + 4.21991)t} \\ & + \left( \frac{-0.46778 - 0.16310931}{1741.79 + 2270.451} \right) e^{(-0.9084 - 4.21991)t} \end{aligned}$$

If the last two terms are combined in accordance with the transformation formula in the appendix, the final expression for  $\phi_L$  can be written:

$$\begin{aligned} \phi_L = & 3.3471 - 3.3497e^{-0.01567t} + 0.002876e^{-18.623t} \\ & - 0.00034623e^{-0.9084t} \cos 4.2199(t + 0.1354) \quad (5) \end{aligned}$$



Similarly,

$$\begin{aligned} \phi_N = & 11.5982 - 11.62275e^{-0.01587t} + 0.00038854e^{-18.623t} \\ & + 0.043548e^{-0.0084t} \cos 4.2199 (t + 0.23305) \end{aligned} \quad (6)$$

Practical solutions are most conveniently obtained by graphical addition and subtraction of the component parts of the motion. It can be seen that a subtraction involves the small difference of relatively large quantities. For this reason it is necessary to retain as many absolute figures as possible in the evaluation of the roots of the stability equation and in the evaluation of the individual components of the motion.

Equations (5) and (6) are plotted in figure 5 (a) for airplane A. The corresponding expressions for airplane B are plotted in figure 5 (b),

From figure 5 (a) the unit solutions for  $\phi_L$  and  $\phi_N$  after 5 seconds are

$$\phi_L_{5 \text{ sec}} = 0.25 \text{ radian}$$

$$\phi_N_{5 \text{ sec}} = 0.86 \text{ radian}$$

For a  $1^\circ$  aileron deflection from table I,

$$\delta_a L \delta_a = 1.54 \text{ per second}^2$$

and

$$\delta_a N \delta_a = 0$$

Hence, for airplane A,

$$(\phi)_{5 \text{ sec}} = (1.54) (0.25) = 0.385 \text{ radian} = 22.1^\circ$$

The total aileron deflection necessary to bank airplane B  $22.1^\circ$  in 5 seconds is given by

$$\delta_a = \frac{\phi_t}{L_{\delta_a} (\phi_L)_t + N_{\delta_a} (\phi_N)_t}$$

where

$$L_{\delta_a} = \frac{\left(\frac{dC_L}{d\delta_a}\right) qSb}{mk_X^2}$$

$$N_{\delta_a} = \frac{\left(\frac{dC_n}{d\delta_a}\right) qSb}{mk_Z^2}$$

For airplane B,  $L_{\delta_a} = 1.54$  per degree,  $N_{\delta_a} = 0$ ,

and  $\phi_L = 0.26$  radian; hence,

$$\delta_a = \frac{0.385}{(1.54)(0.26)} = 0.96^\circ$$

or

$$\Delta\delta_a = -0.04^\circ$$

In view of the fact that the ailerons on the two airplanes possess the same hinge-moment characteristics, it can be concluded that the aileron stick forces developed during maneuvers will at least be no greater (or heavier) on the modified airplane than on the original airplane.

Rudder control.— The sensitivity of the rudder control, particularly at high speeds, may be judged by the angle of bank and the angle of yaw generated, after a definite time interval, by the yawing moment impressed by the rudder. Here again the time must be of sufficient duration to permit the interaction of secondary rolling and yawing effects.

Airplane A was assumed to be flying level at high speed and to be suddenly subjected to a yawing moment impressed by a given rudder deflection. The angle of bank

$\phi$  and the angle of yaw  $\psi$ , generated after 5 seconds, were calculated. Under similar circumstances, the angle of bank  $\phi + \Delta\phi$  and the angle of yaw  $\psi + \Delta\psi$  developed by airplane B were similarly calculated. The magnitudes of  $\Delta\phi$  and  $\Delta\psi$  were taken as a measure of the relative sensitivity of airplanes A and B to their respective rudder controls. The computations follow.

From figure 5 and table I,

|                                            |                  | <u>Airplane A</u> | <u>Airplane B</u> |
|--------------------------------------------|------------------|-------------------|-------------------|
| $(\phi_L)_5 \text{ sec}$                   | radian - - - - - | 0.25              | 0.26              |
| $(\phi_N)_5 \text{ sec}$                   | radian - - - - - | .86               | .98               |
| $\delta_{rL}\delta_r$ , per $\text{sec}^2$ | - - - - -        | .308              | .410              |
| $\delta_{rN}\delta_r$ , per $\text{sec}^2$ | - - - - -        | .549              | .652              |

If a form similar to that of equation (4) is used,

$$\phi_t = (\delta_{rL}\delta_r) (\phi_L)_t + (\delta_{rN}\delta_r) (\phi_N)_t$$

and the angle of bank of airplane A, 5 seconds after the sudden application of a  $1^\circ$  rudder deflection, is

$$\begin{aligned} (\phi)_5 \text{ sec} &= (-0.308) (0.25) + (0.549) (0.86) \\ &= 0.395 \text{ radian} = 22.6^\circ \end{aligned}$$

For airplane B,

$$\begin{aligned} (\phi)_5 \text{ sec} &= (-0.410) (0.26) - (0.652) (0.98) \\ &= 0.532 \text{ radian} = 30.5^\circ \end{aligned}$$

and

$$\Delta\phi = 8^\circ$$

The angles of yaw generated by the unit disturbances were obtained in a similar manner: thus,

$$(\psi)_t = \delta_r L \delta_r (\psi_L)_t + \delta_r N \delta_r (\psi_N)_t$$

In the maneuver investigated, the unit solutions after 5 seconds were as follows:

|                                             | <u>Airplane A</u> | <u>Airplane B</u> |
|---------------------------------------------|-------------------|-------------------|
| $(\psi_L)_{5 \text{ sec}}$ radian - - - - - | 0.04              | 0.04              |
| $(\psi_N)_{5 \text{ sec}}$ radian - - - - - | .30               | .26               |

For airplane A,

$$\begin{aligned} (\psi)_{5 \text{ sec}} &= (-0.308)(0.04) + (0.549)(0.30) \\ &= 0.153 \text{ radian} = 8.8^\circ \end{aligned}$$

and, for airplane B,

$$\begin{aligned} (\psi)_{5 \text{ sec}} &= (-0.410)(0.04) + (0.652)(0.26) \\ &= 0.154 \text{ radian} = 8.8^\circ \end{aligned}$$

or

$$\Delta\psi = 0^\circ$$

It appears that, for equal rudder deflections, the modified airplane will tend to generate more bank than the original airplane. The increase in the wing dihedral and the modified fin and rudder may result, then, in a rudder control that will be slightly more sensitive at high speeds than the rudder control on the original airplane.

Hinge-moment measurements are, of course, necessary to determine whether the resultant rudder-pedal forces on the modified airplane will be greater or less than those on the original airplane. The hinge-moment characteristics

can be compared on the basis of the quantity

$$\left( \frac{s_r c_r}{\frac{1}{2} \rho v^2} \right) \left( \frac{\partial \theta_h}{\partial \delta_r} \right) = R$$

For airplane A,  $R = -0.297$ ; for airplane B,  $R = -0.150$ . The resultant pedal forces, on the other hand, depend not only on the hinge-moment characteristics of the rudder but also on its floating characteristics and on the particular deflection required to perform a stipulated maneuver. The resultant hinge moment per degree of rudder deflection may be expressed as follows:

$$\frac{H}{\delta_r} = R \left[ 1 - \left( \frac{\partial \delta_r}{\partial \psi} \right) c_{h_r} = 0 \frac{\psi}{\delta_r} \right]$$

The quantity  $\left( \frac{\partial \delta_r}{\partial \psi} \right) c_{h_r} = 0$  is theoretically a con-

stant for any given tail arrangement but actually it is very critical to interference effects at the tail and fluctuates considerably. For small angles of yaw ( $\pm 5^\circ$ ),

$\left( \frac{\partial \delta_r}{\partial \psi} \right) c_{h_r} = 0$  was practically zero for both airplanes A

and B; and the quantity  $R$  may, therefore, be taken as a measure of the pedal forces for a given rudder deflection.

## THE LONGITUDINAL MOTION OF THE AIRPLANE

When the longitudinal stability of an airplane is discussed, the characteristic usually referred to is the "static" longitudinal stability. If static longitudinal stability exists, the dynamic stability characteristics are of minor importance (reference 5).

The usual index for static longitudinal stability is the rate of change of pitching-moment coefficient with lift coefficient  $dC_m/dC_L$  or some quantity proportional to it. In flight the most convenient method of evaluating the amount of static stability present is to measure the elevator angle required to trim the airplane at various speeds; the slope,  $d\delta_e/dV$  or  $d\delta_e/d\alpha$ , is an index of the degree of static stability possessed by the airplane. The significance of the ratio  $d\delta_e/d\alpha$ , methods of evaluating it for power-off and windmilling conditions, and suggested design values are discussed in reference 6. The necessity for power-on wind-tunnel tests for securing the effect of power has already been discussed (fig. 2).

If tunnel data are available from which the floating angle of the elevator may be calculated for any lift coefficient, the elevator stick force required for trim at any lift coefficient can be calculated from the following formula:

$$P = \left( \frac{d\delta_e}{dx} \right) \left( \frac{\partial C_h}{\partial \delta_e} \Delta\delta_e^0 \right) \frac{1}{2} \rho V^2 S_e c_e \quad (7)$$

where

$P$  stick force for trim, pounds

$x$  linear travel of top of control column, feet

$\Delta\delta_e^0$  difference, in degrees, between elevator angle required for trim at lift coefficient under consideration and free-floating angle of elevator

The term  $\Delta\delta_e^0$  may fluctuate considerably with power.

The slope of the curve relating the variation in stick force with forward speed  $dP/dV$  depends on the speed at which the airplane is trimmed for zero stick force and, consequently, depends on the initial setting of the trimming tab. In the comparison of the change in the slope of the stick-force curve resulting from modifications to the elevator, care should therefore be taken to orient the elevator trimming-tab settings so that zero stick force

always occurs at the same speed. Otherwise, a superficial examination of the curves of stick-force variation with air speed may lead to incorrect conclusions concerning the effectiveness of the tail surfaces or the stability of the airplane.

In addition to providing a means of trimming the airplane in steady flight, the elevator must be capable of changing the airplane flight path. The rate at which this change is accomplished in a quick pull-up can, in a sense, be interpreted as a measure of the effectiveness of the elevator in maneuvers.

It is convenient to take the rate of change of the maximum normal acceleration per unit of elevator deflection as an index of elevator effectiveness in maneuvers. Care must be exercised, however, in interpreting this index. Although it is essential in a pursuit airplane to design an elevator sufficiently powerful to maneuver the airplane to the maximum lift coefficient of the wing, it has been found very undesirable if this condition is fulfilled with a minimum amount of elevator deflection. As discussed in reference 6, satisfactory static stability characteristics require the quantity  $d\delta/d\alpha$  to have a value around 0.5. In airplanes that required considerably less stick travel to trim the airplane over the angle-of-attack range inadvertent stalling has frequently occurred in accelerated maneuvers. The optimum value for the rate of change of normal acceleration per unit of elevator deflection is therefore conditioned by the requirements of satisfactory static longitudinal stability.

In the following section a simplified formula is developed that permits the rapid estimation of the normal accelerations developed in abrupt pull-up maneuvers.

Development of a simplified formula for normal accelerations produced in abrupt pull-ups.— To a first order of approximation, the equations of motion in the plane of symmetry involving a disturbance in pitch may be written as follows;

$$\left. \begin{aligned} \frac{du}{dt} &= uX_u + wX_w + qX_q - g\theta & (a) \\ \frac{dw}{dt} &= uZ_u + wZ_w + q \left( U_0 + Z_q \right) - g\theta_0\theta + \delta_e Z_{\delta_e} & (b) \\ \frac{dq}{dt} &= uM_u + wM_w + qM_q + \delta_e M_{\delta_e} & (c) \end{aligned} \right\} (8)$$

Also

$$\frac{d\theta}{dt} = q$$

where  $\theta_0$  is the initial angle the X axis makes with the horizontal axis.

The axes in which equations (8) are expressed are similar to those defined in the section on lateral motion except that the X axis is inclined at an angle  $\theta$  to the relative wind.

In general, this system of equations must be solved by methods analogous to those used in obtaining the solutions to the lateral equations of motion. The normal acceleration is given by the expression  $dw/dt - qU_0$ .

These equations have been solved for the airplane, the characteristics of which (evaluated, so far as possible, from wind-tunnel tests on a powered model) are presented in table II. The curve of the normal acceleration resulting from a  $1^\circ$  upward elevator deflection is plotted in figure 5 together with other components of the motion.

The variation of the components of motion shown in figure 6 is typical of an abrupt pull-up maneuver at high speed. The following facts are apparent from this figure:

1. During the time the airplane takes to attain maximum acceleration, the velocity  $V$  remains sensibly constant and is equal to  $U_0$ , the equilibrium velocity. Accordingly,  $u = du = 0$ , and equation (8a) can be neglected.
2. In the vicinity of the maximum acceleration, the time rate of change of pitching velocity  $dq/dt$  is approximately zero, that is,  $q$  is approximately constant.
3. In the vicinity of the maximum acceleration, the acceleration component  $dw/dt$  is almost zero and the acceleration thereafter is given almost entirely by the product  $qU_0$ .



If  $Z_{\delta_e}$ ,  $Z_q$ , and  $g\theta_0$  (all of which are small) are neglected and the substitutions  $\alpha Z_\alpha = w Z_w$  and  $w M_w = \alpha M_\alpha$  are made, equation (8), for an abrupt pull-up maneuver, can be simplified to the following form:

$$\left. \begin{aligned} qU_0 + \alpha Z_\alpha &= 0 \\ qM_q + \alpha M_\alpha &= -M_0 \end{aligned} \right\}$$

where

$$M_0 = \delta_e M_{\delta_e}$$

The solution for  $qU_0$  is

$$qU_0 = U_0 \left( \frac{M_0 Z_\alpha}{U_0 M_\alpha - Z_\alpha M_q} \right) \quad (9)$$

where

$$M_0 = \left( \frac{\partial C_m}{\partial \delta_e} \right) \delta_e \frac{1}{2} \frac{\rho S V^2 c}{m k_Y^2}$$

$$Z_\alpha = - \left( \frac{dC_L}{d\alpha} \right) \frac{1}{2} \frac{\rho S V^2}{m}$$

$$M_\alpha = \left( \frac{dC_m}{d\alpha} \right) \frac{1}{2} \frac{\rho S V^2 c}{m k_Y^2}$$

$$M_q = -F\eta_t \left( \frac{dC_L}{d\alpha} \right)' \frac{1}{2} \frac{\rho S V l^2}{m k_Y^2} \frac{S'}{S}$$

In the expression for  $M_q$ , the slipstream factor  $F$  normally has a value between 1 and 1.25 for high-speed flight. The tail efficiency factor  $\eta_t$  is always less than 1 and is generally about 0.9. The value of the product  $F\eta_t$  is therefore always about unity.

If the airplane is assumed to be flying level before the pull-up maneuver,  $mg = C_L \frac{1}{2} \rho S V^2 = C_L \frac{1}{2} \rho S U_0^2$ . If the substitutions  $U_0 = V$ ,  $\mu = \frac{m}{\frac{1}{2} \rho S l}$ , and  $F_{\eta t} = 1$  are made, equation (9) can be reduced to the following simplified form:

$$\frac{qU_0}{\delta_e} = - \frac{g\mu}{C_L} \left[ \frac{\partial C_m / \partial \delta_e}{\mu \frac{dC_m}{dC_L} - \frac{l}{c} \left( \frac{dC_L}{d\alpha} \right)' \frac{s'}{s}} \right] \quad (10)$$

where  $qU_0/\delta_e$  represents the change in normal acceleration per unit of elevator deflection.

Formula (10) gives a value of the normal acceleration produced during an abrupt pull-up maneuver that closely approximates the maximum acceleration which would be obtained by solving the more cumbersome equations of motion. It is the acceleration that will be obtained if the elevator is instantly deflected to its final position and held in that position until the maximum acceleration is reached. The formula gives less accurate results when  $dC_m/dC_L$  is small, that is,  $-\frac{dC_m}{dC_L} < 0.01$ . Several calculations made for smaller values of  $dC_m/dC_L$  gave results, however, that were in error by less than 7 percent. Greater accuracy is to be expected at small values of  $C_L$  on account of the approximations made involving  $\theta$ . At high values of  $C_L$ , these approximations introduce greater errors.

Maneuverability and stability.— The manner in which the normal acceleration produced in a pull-up maneuver is affected by  $dC_m/dC_L$  and  $\mu$  is shown in figure 7. An examination of this figure reveals that, for airplanes with high wing loadings (large  $\mu$ ) and low static stability (small  $dC_m/dC_L$ ), the normal acceleration produced per degree of elevator is affected by a small change in the

static stability. In accelerated maneuvers this effect, as far as the pilot is concerned, will manifest itself in the form of increased stick forces if it is assumed that the pilot wants to produce a given acceleration regardless of the degree of static stability present in the airplane.

The index of static stability  $dC_m/dC_L$  for an airplane with the characteristics given in table II is assumed to be -0.022. The curve of the normal acceleration produced in a pull-up from level flight at 448 feet per second is given in figure 6. The maximum change in the normal acceleration attained with an upward elevator deflection of  $1^\circ$  is about 84.5 feet per second.

From formula (10)

$$\frac{qU_0}{\delta_e} = -g \frac{34.4}{0.10} \left( \frac{-0.021}{(34.4)(-0.022) + (-2.01)} \right) = 2.6g \text{ per}$$

degree and  $qU_0 = 83.7$  feet per second<sup>2</sup>, an error of less than 1 percent.

If the static stability of the airplane is improved by moving the center of gravity forward 0.078c, the expression for the normal acceleration becomes

$$\begin{aligned} qU_0 &= -g \frac{34.4}{0.10} \left( \frac{-0.021}{(34.4)(-0.10) + (-2.07)} \right) \delta_e^\circ \\ &= (-1.37g) \delta_e^\circ \end{aligned}$$

If it is desired to load the airplane to its former additional load factor of 2.6g, the elevator movement required with the higher  $dC_m/dC_L$  would be

$$\delta_e = \frac{2.6g}{-1.37g} = -1.9^\circ$$

Accordingly, the pilot would have to exert nearly twice as much stick force to execute the identical pull-up maneuver because of the increased static stability of the airplane. The desirability of modifying the elevator

in order to change its hinge-moment characteristics thus partly depends on the relationship between the static stability desired and the magnitude of the stick forces acceptable in accelerated maneuvers.

### CONCLUSIONS

This paper is intended to illustrate primarily how power affects the characteristics of a model tested in the wind tunnel and how wind-tunnel data may be used to estimate flying qualities not directly discernible from wind-tunnel tests. The analyses presented in this paper permit the following conclusions to be drawn concerning the methods employed:

(1) In the prediction of the flight characteristics of an airplane operating with power, considerable error may be introduced if wind-tunnel data from tests of a model not equipped with an operating propeller are used.

(2) In the analyses of wind-tunnel rolling-moment data, care should be taken, in the determination of the dihedral effect, to allow for the contribution of the unbalanced pitching moment to the slope of the rolling-moment curve.

(3) The evaluation of the equations of motion permit estimates to be made of the relative effectiveness of the ailerons and of the rudder controls, particularly when changes in wing dihedral are involved as these effects are not readily discernible from static tests of the controls alone. The methods used have been found reasonably accurate, when wind-tunnel data are available, and are not difficult to employ.

(4) The rate of change of normal acceleration per unit of elevator deflection affords a convenient correlation between maneuverability, stability, and elevator-stick forces. The approximate formula developed for calculating the normal accelerations produced during an abrupt pull-up maneuver is simple to evaluate and has been found to yield reasonably accurate results, particularly at high speeds.

Langley Memorial Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., July 22, 1941.

## APPENDIX

## List of Symbols

|           |                                                                                                                             |
|-----------|-----------------------------------------------------------------------------------------------------------------------------|
| $U_0$     | velocity along axis in steady flight                                                                                        |
| $V$       | velocity along flight path                                                                                                  |
| $u$       | velocity along X axis                                                                                                       |
| $v$       | sideslipping component of velocity                                                                                          |
| $w$       | velocity along Z axis                                                                                                       |
| $p$       | angular velocity in roll                                                                                                    |
| $q$       | angular velocity in pitch                                                                                                   |
| $r$       | angular velocity in yaw                                                                                                     |
| $\phi$    | angle of bank                                                                                                               |
| $\psi$    | angle of yaw                                                                                                                |
| $\alpha$  | angle of attack                                                                                                             |
| $\beta$   | angle of sideslip ( $\omega v/U_0$ )                                                                                        |
| $\theta$  | angle X axis makes with horizontal                                                                                          |
| $\delta$  | control setting, with appropriate subscript<br>( $\delta^\circ$ indicates that values are in degrees<br>and not in radians) |
| $X, Y, Z$ | components of force along X, Y, and Z axes,<br>respectively                                                                 |
| $L$       | rolling moment about X axis                                                                                                 |
| $M$       | pitching moment about Y axis                                                                                                |
| $N$       | yawing moment about Z axis                                                                                                  |
| $H$       | hinge moment                                                                                                                |
| $b$       | wing span                                                                                                                   |
| $c$       | chord (of wing, unless otherwise subscripted)                                                                               |

- S** area (of wing, unless otherwise subscripted)
- l** tail length (distance from center of gravity to tail post)
- m** mass
- $\mu$**  relative density factor  $\frac{m}{1/2\rho S l}$
- $mk_x^2$**  moment of inertia about X axis
- $mk_y^2$**  moment of inertia about Y axis
- $mk_z^2$**  moment of inertia about Z axis
- t** time
- D**  $d/dt$
- $\rho$**  density
- g** gravity (32.2 ft/sec<sup>2</sup>)
- $C_l$**  rolling-moment coefficient  $\left(\frac{L}{1/2\rho V^2 S b}\right)$
- $C_m$**  pitching-moment coefficient  $\left(\frac{M}{1/2\rho V^2 c}\right)$
- $C_n$**  yawing-moment coefficient  $\left(\frac{N}{1/2\rho V^2 S b}\right)$
- $C_h$**  hinge-moment coefficient  $\left(\frac{H}{1/2\rho V^2 S c}\right)$
- $C_Y$**  lateral-force coefficient  $\left(\frac{Y}{1/2\rho V^2 S}\right)$
- $C_L$**  lift coefficient
- $\eta_t$**  tail efficiency factor
- F** empirical slipstream factor to account for contribution of fuselage, wing, and propeller to damping in pitch

$Y_\beta$  $L_p$  $L_r$  $L_\delta$  $L_\beta$  $N_p$  $N_r$  $N_\delta$  $N_\beta$ 

lateral-stability derivatives in terms of unit mass or moment of inertia of airplane. For example,

$$Y_\beta = \frac{\partial Y / \partial \beta}{m}$$

$$L_p = \frac{\partial L / \partial p}{mk_X^2}$$

$$N_r = \frac{\partial L / \partial r}{mk_Z^2}$$

 $X_u$  $X_w$  $X_q$  $Z_u$  $Z_w$  $Z_q$  $Z_\delta$  $Z_\alpha$  $M_u$  $M_w$  $M_q$  $M_\delta$  $M_\alpha$ 

longitudinal-stability derivatives in terms of unit mass or moment of inertia of airplane. For example,

$$X_u = \frac{\partial X / \partial u}{m}$$

$$Z_\delta = \frac{\partial Z / \partial \delta}{m}$$

$$M_u = \frac{\partial M / \partial u}{mk_Y^2}$$

Subscripts are defined:

L solutions obtained when  $\delta L_\delta = 1$  and  $\delta N_\delta = 0$

N solutions obtained when  $\delta L_\delta = 0$  and  $\delta N_\delta = 1$

a aileron

e elevator

r rudder

t time

o initial value

Primed symbols refer to horizontal tail.

SOLUTIONS OF THE LATERAL EQUATIONS OF MOTION WHEN  
THE ROLLING DISTURBANCE IS UNITY AND THE YAWING  
DISTURBANCE IS ZERO ( $\delta L_\delta = 1$ ;  $\delta N_\delta = 0$ )

The stability equation  $F(D) = 0$  is obtained from equation (2) of the text by expanding the third-order determinant formed by the coefficients of the variables. Thus,

$$F(D) = \begin{vmatrix} D-L_p & -L_r & -L_\beta \\ -N_p & D-N_r & -N_\beta \\ -\frac{L}{U_o} & D & D \left( D - \frac{Y_\beta}{U_o} \right) \end{vmatrix}$$

$$= (D-L_p) \begin{vmatrix} D-N_r & -N_\beta \\ D & D \left( D - \frac{Y_\beta}{U_o} \right) \end{vmatrix}$$



$$\begin{aligned}
& + N_p \begin{vmatrix} -L_r & -L_\beta \\ D & D \left( D - \frac{Y_\beta}{U_0} \right) \end{vmatrix} \begin{vmatrix} -\frac{\xi}{U_0} \\ D - N_r \end{vmatrix} \begin{vmatrix} -L_r & -L_\beta \\ D - N_r & -N_\beta \end{vmatrix} \\
& = D^4 - \left( L_p + N_r + \frac{Y_\beta}{U_0} \right) D^3 \\
& + \left[ \frac{Y_\beta}{U_0} (N_r + L_p) + N_r L_p - L_r N_p + N_\beta \right] D^2 \\
& + \left[ -\frac{Y_\beta}{U_0} (N_r L_p - L_r N_p) - L_p N_\beta + L_\beta N_p - \frac{\xi}{U_0} L_\beta \right] D \\
& + \frac{\xi}{U_0} (L_\beta N_r - L_r N_\beta) = 0
\end{aligned}$$

Let  $\lambda_1, \lambda_2, \lambda_3$ , and  $\lambda_4$  be the four roots of this equation  $F(D) = 0$  and form the following products:

$$\begin{aligned}
P &= \lambda_1 \lambda_2 \lambda_3 \lambda_4 = \frac{\xi}{U_0} (L_\beta N_r - L_r N_\beta) \\
Q &= \lambda_1 (\lambda_1 - \lambda_2) (\lambda_1 - \lambda_3) (\lambda_1 - \lambda_4) \\
R &= \lambda_2 (\lambda_2 - \lambda_1) (\lambda_2 - \lambda_3) (\lambda_2 - \lambda_4) \\
S &= \lambda_3 (\lambda_3 - \lambda_1) (\lambda_3 - \lambda_2) (\lambda_3 - \lambda_4) \\
T &= \lambda_4 (\lambda_4 - \lambda_1) (\lambda_4 - \lambda_2) (\lambda_4 - \lambda_3)
\end{aligned}$$

The solutions for any of the variables can be expressed in terms of the products of the roots just formed and the stability derivatives. By use of the auxiliary

relations  $\frac{d\phi}{dt} = p$  and  $\frac{d\psi}{dt} = r$  solutions can be written

directly for both  $\phi$  and  $\psi$ . As it is usually desirable in this work to determine  $\phi$  and  $\psi$  rather than  $p$  and  $r$ , it is convenient to solve for  $\phi$  and  $\psi$  directly and then to differentiate each of these solutions, respectively, to obtain  $p$  and  $r$  if these variables are wanted. The solutions follow:

Angle of Bank  $\phi_L$

$$\phi_L = \phi_{L_0} + \phi_{L_1} e^{\lambda_1 t} + \phi_{L_2} e^{\lambda_2 t} + \phi_{L_3} e^{\lambda_3 t} + \phi_{L_4} e^{\lambda_4 t}$$

where

$$\phi_{L_0} = \frac{N_\beta + \frac{N_r Y_\beta}{U_0}}{P}$$

$$\phi_{L_1} = \frac{\lambda_1^2 - \lambda_1 \left( N_r + \frac{Y_\beta}{U_0} \right) + \left( N_\beta + \frac{N_r Y_\beta}{U_0} \right)}{Q}$$

$$\phi_{L_2} = \frac{\lambda_2^2 - \lambda_2 \left( N_r + \frac{Y_\beta}{U_0} \right) + \left( N_\beta + \frac{N_r Y_\beta}{U_0} \right)}{R}$$

$$\phi_{L_3} = \frac{\lambda_3^2 - \lambda_3 \left( N_r + \frac{Y_\beta}{U_0} \right) + \left( N_\beta + \frac{N_r Y_\beta}{U_0} \right)}{S}$$

$$\phi_{L_4} = \frac{\lambda_4^2 - \lambda_4 \left( N_r + \frac{Y_\beta}{U_0} \right) + \left( N_\beta + \frac{N_r Y_\beta}{U_0} \right)}{T}$$

Angle of Sideslip  $\beta_L$ 

$$\beta_L = \beta_{L_0} + \beta_{L_1} e^{\lambda_1 t} + \beta_{L_2} e^{\lambda_2 t} + \beta_{L_3} e^{\lambda_3 t} + \beta_{L_4} e^{\lambda_4 t}$$

where

$$\beta_{L_0} = \frac{\xi}{U_0} \frac{-N_r}{P}$$

$$\beta_{L_1} = \frac{\xi}{U_0} \frac{(\lambda_1 - N_r) - \lambda_1 N_p}{Q}$$

$$\beta_{L_2} = \frac{\xi}{U_0} \frac{(\lambda_2 - N_r) - \lambda_2 N_p}{R}$$

$$\beta_{L_3} = \frac{\xi}{U_0} \frac{(\lambda_3 - N_r) - \lambda_3 N_p}{S}$$

$$\beta_{L_4} = \frac{\xi}{U_0} \frac{(\lambda_4 - N_r) - \lambda_4 N_p}{T}$$

 Angle of Yaw  $\psi_L$ 

$$\psi_L = \psi_{L_0} + \psi_{L_1} e^{\lambda_1 t} + \psi_{L_2} e^{\lambda_2 t} + \psi_{L_3} e^{\lambda_3 t} + \psi_{L_4} e^{\lambda_4 t}$$

where

$$\psi_{L_0} = \frac{N_\beta \frac{\xi}{U_0}}{P}$$

$$\psi_{L_1} = \frac{N_\beta \frac{\xi}{U_0} - \lambda_1 \frac{Y_\beta N_p}{U_0}}{Q}$$

$$\psi_{L_2} = \frac{N_\beta \frac{\xi}{U_0} - \lambda_2 \frac{Y_\beta N_p}{U_0}}{R}$$

$$\psi_{L_3} = \frac{N_\beta \frac{\xi}{U_0} - \lambda_3 \frac{Y_\beta N_p}{U_0}}{S}$$

$$\psi_{L_4} = \frac{N_\beta \frac{\xi}{U_0} - \lambda_4 \frac{Y_\beta N_p}{U_0}}{T}$$

Yawing Velocity  $r_L$

$$r_L = \frac{d}{dt} (\psi_L)$$

The yawing velocity is most easily obtained by direct differentiation.

Rolling Velocity  $p_L$

$$p_L = \frac{d}{dt} (\phi_L)$$

The rolling velocity is most easily obtained by direct differentiation.

SOLUTIONS OF THE LATERAL EQUATIONS OF MOTION WHEN  
 THE YAWING DISTURBANCE IS UNITY AND THE ROLLING  
 DISTURBANCE IS ZERO ( $\delta N_\delta = 1$ ;  $\delta L_\delta = 0$ )

Angle of Bank  $\phi_N$

$$\phi_N = \phi_{N_0} + \phi_{N_1} e^{\lambda_1 t} + \phi_{N_2} e^{\lambda_2 t} + \phi_{N_3} e^{\lambda_3 t} + \phi_{N_4} e^{\lambda_4 t}$$

where

$$\phi_{N_0} = \frac{-L_\beta - L_r \frac{Y_\beta}{U_0}}{P}$$

$$\phi_{N_1} = \frac{\left(-L_\beta - L_r \frac{Y_\beta}{U_0}\right) + \lambda_1 L_r}{Q}$$

$$\phi_{N_2} = \frac{\left(-L_\beta - L_r \frac{Y_\beta}{U_0}\right) + \lambda_2 L_r}{R}$$

$$\phi_{N_3} = \frac{\left(-L_\beta - L_r \frac{Y_\beta}{U_0}\right) + \lambda_3 L_r}{S}$$

$$\phi_{N_4} = \frac{\left(-L_\beta - L_r \frac{Y_\beta}{U_0}\right) + \lambda_4 L_r}{T}$$

Angle of Sideslip  $\beta_N$ 

$$\beta_N = \beta_{N_0} + \beta_{N_1} e^{\lambda_1 t} + \beta_{N_2} e^{\lambda_2 t} + \beta_{N_3} e^{\lambda_3 t} + \beta_{N_4} e^{\lambda_4 t}$$

where

$$\beta_{N_0} = \frac{\frac{gL_r}{U_0}}{P}$$

$$\beta_{N_1} = \frac{\frac{gL_r}{U_0} + \lambda_1 L_p - \lambda_1^2}{Q}$$

$$\beta_{N_2} = \frac{\frac{gL_r}{U_0} + \lambda_2 L_p - \lambda_2^2}{R}$$

$$\beta_{N_3} = \frac{\frac{gL_r}{U_0} + \lambda_3 L_p - \lambda_3^2}{S}$$

$$\beta_{N_4} = \frac{\frac{gL_r}{U_0} + \lambda_4 L_p - \lambda_4^2}{T}$$

Angle of Yaw  $\psi_N$ 

$$\psi_N = \psi_{N_0} + \psi_{N_1} e^{\lambda_1 t} + \psi_{N_2} e^{\lambda_2 t} + \psi_{N_3} e^{\lambda_3 t} + \psi_{N_4} e^{\lambda_4 t}$$

where

$$\psi_{N_0} = \frac{-L_\beta \frac{\xi}{U_0}}{P}$$

$$\psi_{N_1} = \frac{-\lambda_1^2 \frac{Y_\beta}{U_0} + \lambda_1 L_p \frac{Y_\beta}{U_0} - L_\beta \frac{\xi}{U_0}}{Q}$$

$$\psi_{N_2} = \frac{-\lambda_2^2 \frac{Y_\beta}{U_0} + \lambda_2 L_p \frac{Y_\beta}{U_0} - L_\beta \frac{\xi}{U_0}}{R}$$

$$\psi_{N_3} = \frac{-\lambda_3^2 \frac{Y_\beta}{U_0} + \lambda_3 L_p \frac{Y_\beta}{U_0} - L_\beta \frac{\xi}{U_0}}{S}$$

$$\psi_{N_4} = \frac{-\lambda_4^2 \frac{Y_\beta}{U_0} + \lambda_4 L_p \frac{Y_\beta}{U_0} - L_\beta \frac{\xi}{U_0}}{T}$$

Yawing Velocity  $r_N$

$$r_N = \frac{d}{dt} (\psi_N)$$

The yawing velocity is most easily obtained by direct differentiation.

Rolling Velocity  $p_N$

$$p_N = \frac{d}{dt} (\phi_N)$$

The rolling velocity is most easily obtained by direct differentiation.

DISCUSSION OF THE EQUATION  $F(D) = 0$ 

For most airplanes the solution of  $F(D) = 0$  will yield both real and imaginary roots. In the solution of the equation, the real roots can first be isolated (Horner's method) and extracted from the equation. The imaginary roots can then be found by solving the resulting quadratic. Because imaginary roots always occur in pairs, two of the roots will be of the form  $a \pm ib$ , where  $a$  and  $b$  are constants.

The components of the motion containing the imaginary roots can be combined conveniently into a single term involving only real numbers. In the case of  $\phi_L$ , the solution is

$$\phi_L = \phi_{L_0} + \phi_{L_1} e^{\lambda_1 t} + \phi_{L_2} e^{\lambda_2 t} + \phi_{L_3} e^{\lambda_3 t} + \phi_{L_4} e^{\lambda_4 t}$$

If  $\lambda_3$  and  $\lambda_4$  are the conjugate imaginary roots,

$\phi_{L_3} e^{\lambda_3 t} + \phi_{L_4} e^{\lambda_4 t}$  will also be imaginary.

Let  $\lambda_3 = a + b$ ; then  $\phi_{L_3}$  will be imaginary and can be reduced to the form  $\frac{E + iF}{G + iH}$ . Further,

$$\phi_{L_3} = \frac{E + iF}{G + iH} \frac{G - iH}{G - iH} = \frac{EG + FH - i(HE - FG)}{G^2 + H^2} = \frac{I + iJ}{K}$$

and

$$\phi_{L_3} e^{\lambda_3 t} + \phi_{L_4} e^{\lambda_4 t} = \frac{2\sqrt{I^2 + J^2}}{K} e^{at} \cos b(t + \gamma)$$

where

$$\gamma = \frac{1}{b} \tan^{-1} \left( \frac{J}{I} \right)$$



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TABLE I

| Airplane | $U_0$<br>(fps)                                          | $S$<br>(sq ft)                                          | $b$<br>(ft)                                             | $m$<br>(slugs)                                          | $mk_X^a$<br>(slug-ft <sup>2</sup> ) | $mk_Z^a$<br>(slug-ft <sup>2</sup> ) | $Y_\beta$<br>(ft/sec <sup>2</sup> ) |
|----------|---------------------------------------------------------|---------------------------------------------------------|---------------------------------------------------------|---------------------------------------------------------|-------------------------------------|-------------------------------------|-------------------------------------|
| A        | 448                                                     | 236                                                     | 37.3                                                    | 174                                                     | 2020                                | 8030                                | -166                                |
| B        | 448                                                     | 236                                                     | 37.3                                                    | 174                                                     | 2020                                | 6030                                | -185                                |
|          |                                                         |                                                         |                                                         |                                                         |                                     |                                     |                                     |
|          | $L_\beta$<br>(per sec <sup>2</sup> )                    | $N_\beta$<br>(per sec <sup>2</sup> )                    | $M_p$<br>(per sec)                                      | $N_r$<br>(per sec)                                      | $L_p$<br>(per sec)                  | $L_r$<br>(per sec)                  |                                     |
| A        | -62.7                                                   | 17.7                                                    | -0.076                                                  | -1.49                                                   | -18.6                               | 0.99                                |                                     |
| B        | -129                                                    | 31.0                                                    | -.088                                                   | -1.77                                                   | -18.6                               | 1.14                                |                                     |
|          |                                                         |                                                         |                                                         |                                                         |                                     |                                     |                                     |
|          | $\delta_a L \delta_a$<br>(per sec <sup>2</sup> )<br>(1) | $\delta_a N \delta_a$<br>(per sec <sup>2</sup> )<br>(1) | $\delta_r L \delta_r$<br>(per sec <sup>2</sup> )<br>(2) | $\delta_r N \delta_r$<br>(per sec <sup>2</sup> )<br>(2) |                                     |                                     |                                     |
| A        | 1.54                                                    | 0                                                       | -0.308                                                  | 0.549                                                   |                                     |                                     |                                     |
| B        | 1.54                                                    | 0                                                       | -.410                                                   | .652                                                    |                                     |                                     |                                     |

<sup>1</sup>  $\delta_a^0 = 1^0$ .

<sup>2</sup>  $\delta_r^0 = -1^0$ .

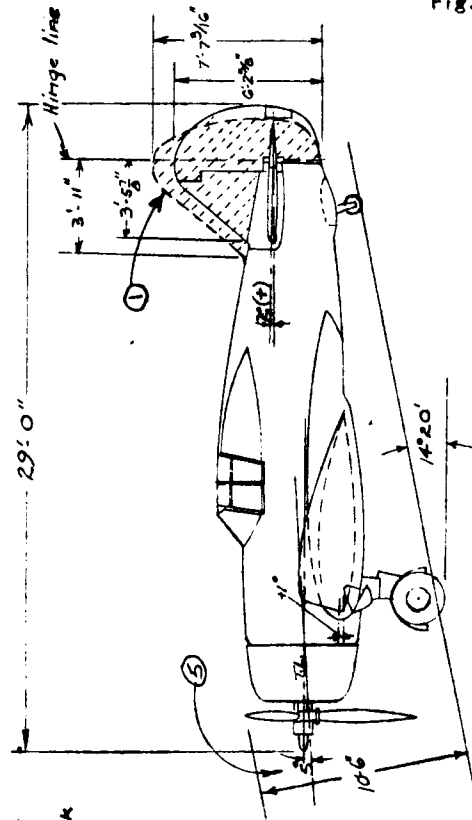
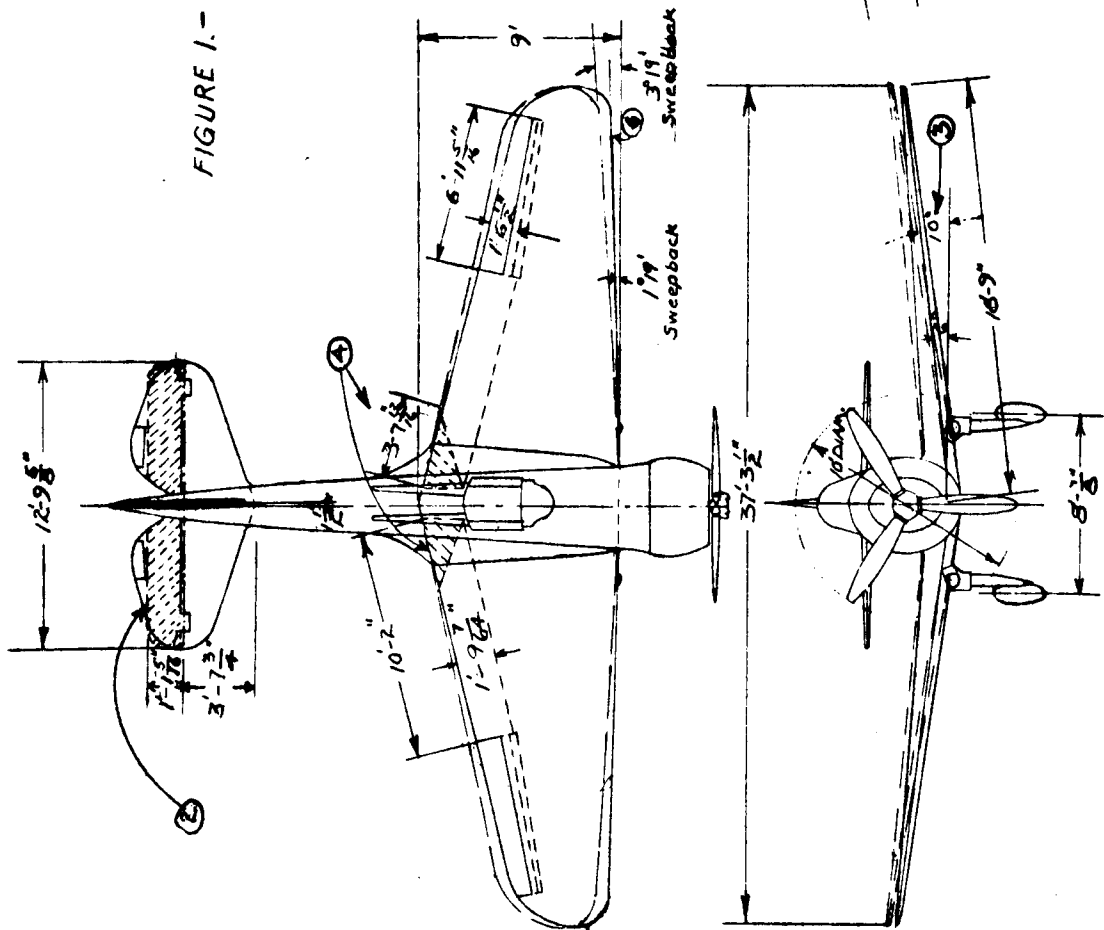
TABLE II

| $U_o$<br>(fps) | $S$<br>(sq ft) | $i$<br>(ft) | $c$<br>(ft) | $m$<br>(slugs) | $mky^2$<br>(slug-ft <sup>2</sup> ) | $Z_u$<br>(per sec) | $Z_w$<br>(per sec) | $Z_q$<br>(fps) |
|----------------|----------------|-------------|-------------|----------------|------------------------------------|--------------------|--------------------|----------------|
| 448            | 236            | 18          | 6.8         | 174            | 4470                               | -0.144             | -3.71              | -12.9          |

| $X_u$<br>(sec) | $M_u$<br>(per ft-sec)<br>(1) | $M_w$<br>(per ft-sec) | $M_q$<br>(per sec) | $\delta_e M \delta_e$<br>(per sec <sup>2</sup> ) |
|----------------|------------------------------|-----------------------|--------------------|--------------------------------------------------|
| -0.044         | -0.0005                      | -0.022                | -7.04              | -1.84                                            |

<sup>1</sup>See reference 7.

FIGURE 1.- A THREE-VIEW DRAWING OF AIRPLANE MODIFICATIONS.



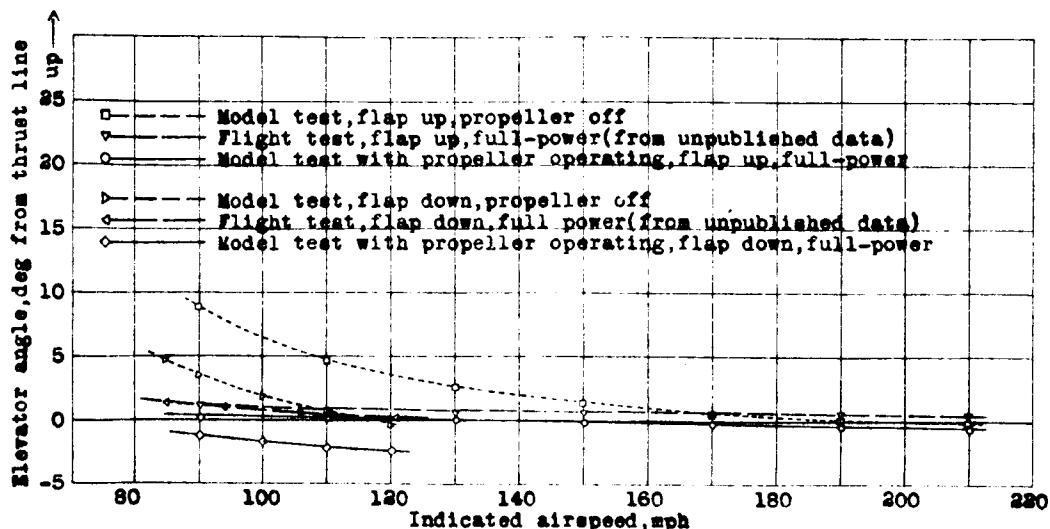
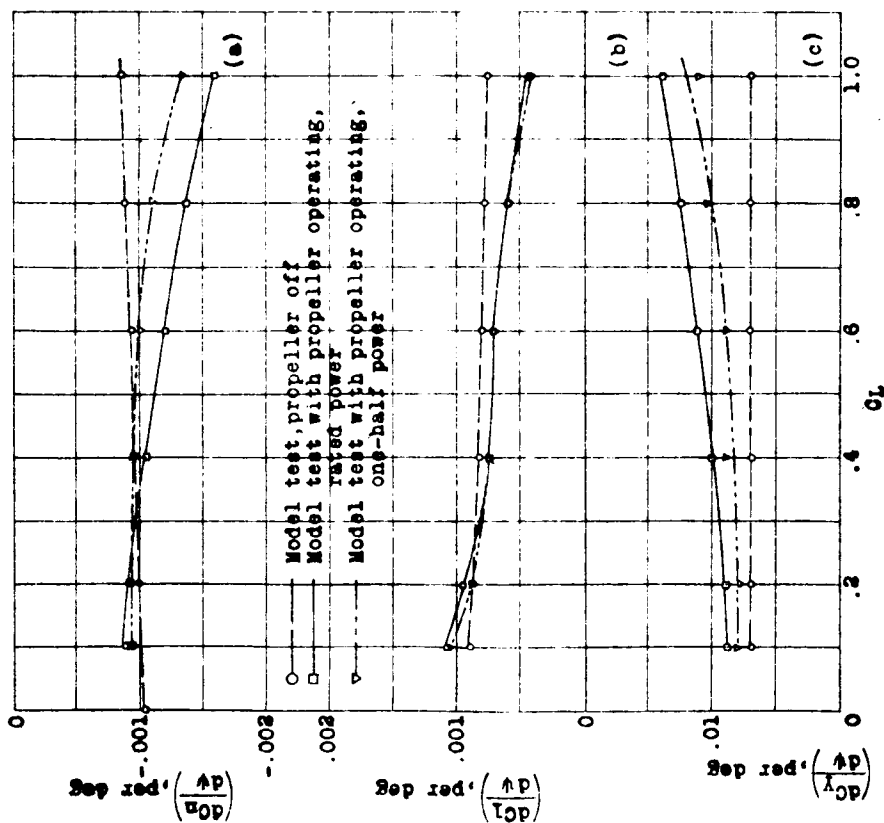


Figure 2.- Variation of elevator angle required for trim with indicated airspeed.



(a) Yawing moment, (b) Rolling moment, (c) Lateral force.  
Figure 3.- Effect of power on the slopes of yawing-moment, rolling-moment, and lateral-force curves. Flap up.

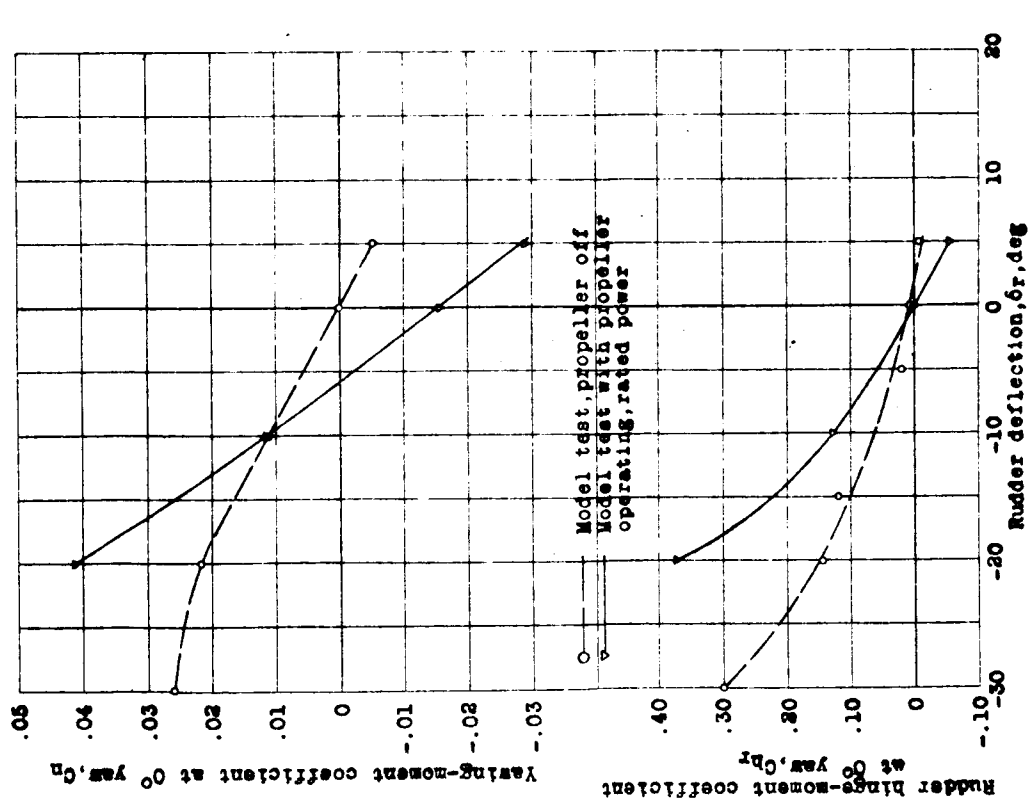
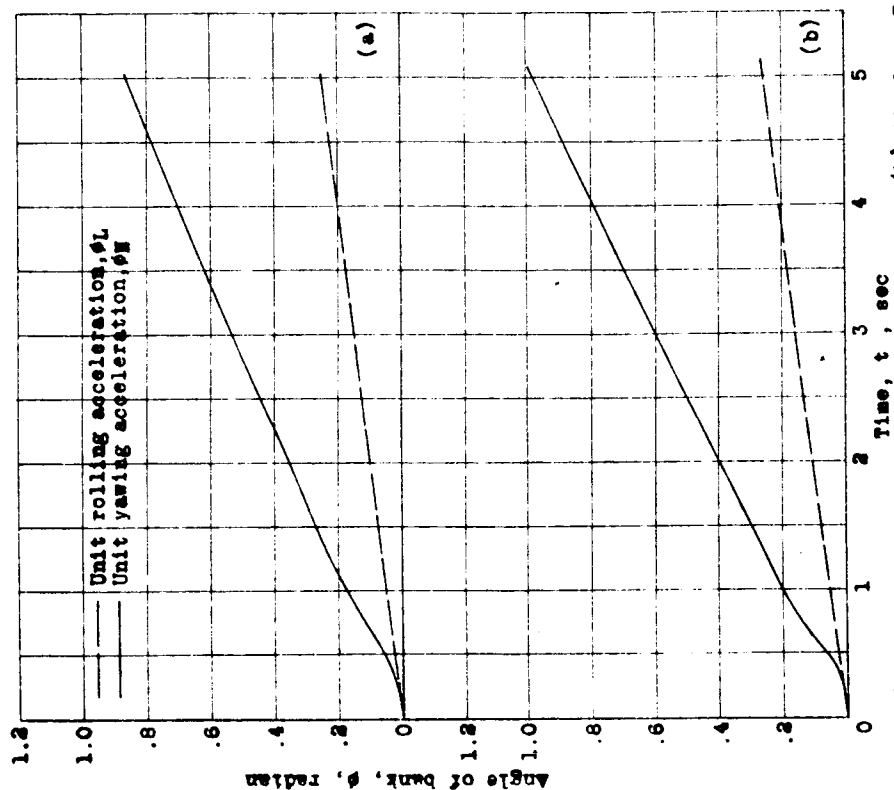


Figure 4.- Effect of power on rudder characteristics. The yawing-moment coefficients have been corrected for fin offset.  $C_L = 0.9$ .



(a) Airplane A. (b) Airplane B.  
Figure 5.- Angles of bank generated by unit rolling and yawing accelerations applied at  $t = 0$ .  $U_0$ , 448 feet per second.

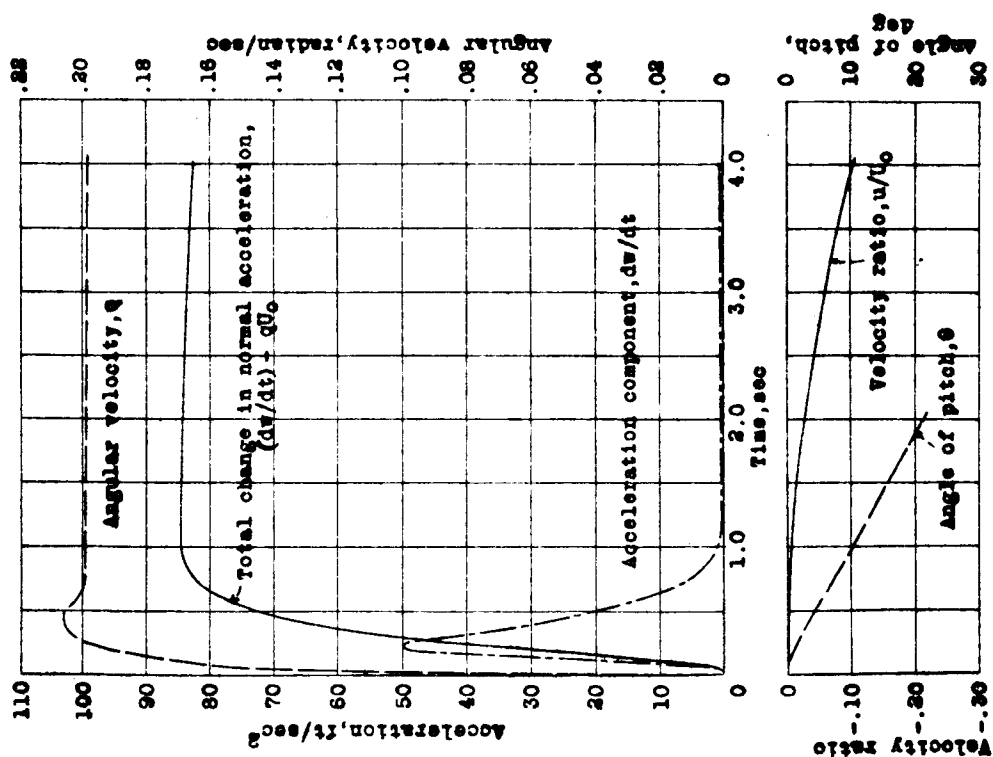


Figure 8.- Abrupt pull-up at high speed.  $\delta_e = 1^\circ$ .

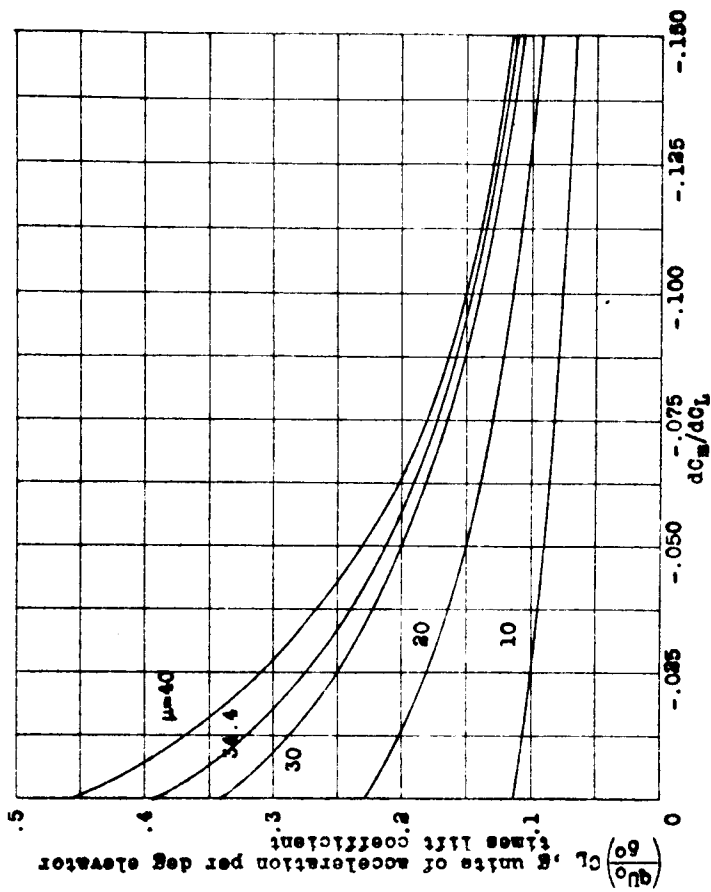


Figure 7.- Variation of normal acceleration per unit of elevation deflection with  $dC_m/dC_L$ . Abrupt pull-up maneuver.

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

# WARTIME REPORT

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THE EFFECT OF COWLING SHAPE ON THE STABILITY

CHARACTERISTICS OF AN AIRPLANE

By C. J. Donlan and W. Letko

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# NACA

WASHINGTON

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Langley Field, Va.

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L-343

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

ADVANCE RESTRICTED REPORT

THE EFFECT OF COWLING SHAPE ON THE STABILITY

CHARACTERISTICS OF AN AIRPLANE

By C. J. Donlan and W. Letko

SUMMARY

Three widely different nose shapes were tested on a fuselage alone and on a complete model in the NACA stability tunnel to investigate the effect of cowl shape on stability characteristics. The results are presented in the form of charts which show the variation in the aerodynamic characteristics with the three nose shapes for the propeller-removed condition over a wide range of angles of attack and yaw.

The results of the investigation indicated that large changes in the cowl shape produced relatively small changes in the aerodynamic characteristics. The effects may be appreciable, however, in the case of an airplane that has marginal stability.

INTRODUCTION

The trend in contemporary airplane design toward higher wing loadings, smaller wing areas with respect to fuselage areas, and larger moments of inertia makes the attainment of satisfactory stability characteristics increasingly difficult (reference 1). Small changes in the stability parameters, therefore, may produce significant changes in the stability characteristics of the airplane. One common change made on existing airplanes is a modification of the engine cowl. Recent flight investigations have suggested that changes in the type or the shape of engine cowl may alter the flying qualities of an airplane, but little direct information is available concerning these effects. Pressure-distribution measurements over cowlings of various shapes are given in references 2, 3, and 4. Reference 2 contains data on cowlings over a considerable range of angles of attack but does not show

the effect of the cowlings on the fuselage pressure distribution. References 3 and 4 contain extensive pressure-distribution measurements at low angles of attack on a number of cowlings-fuselage arrangements over a wide range of Mach numbers. Reference 5 shows the effect of a radial engine and cowlings on the unstable moment of a streamline body but does not show the effect on the weathercock stability of the complete airplane. The present investigation was made to provide direct information concerning the effect of cowlings shape on the longitudinal- and lateral-stability characteristics of the entire airplane.

Two cowlings representing extremes in contemporary design practice were tested. In addition, a streamline nose section was tested to provide a basis for comparison. Arrangements were made also for varying the volume of air flowing through the cowlings. The propellers were not represented. Tests were made of a fuselage alone and of a complete model consisting of a fuselage, a wing, and horizontal and vertical tail surfaces.

The paper presents the aerodynamic characteristics in pitch and in yaw of the different combinations tested. The data must be used in a strictly comparative manner, inasmuch as no corrections for wind-tunnel wall effects or support-strut interference have been applied.

## APPARATUS AND MODELS

The tests were made in the closed-throat NACA stability tunnel. This tunnel is of recent construction. At the outset of this investigation, the characteristics of neither the tunnel nor the balances had been explored very thoroughly; consequently, no attempt has been made to apply any corrections to the data obtained. In view of the comparative nature of the present investigation, however, this procedure should not qualify any of the conclusions reached.

The tunnel is equipped with a six-component balance especially designed to measure the forces and moments with respect to a system of axes commonly used in stability work. This system of axes is shown in figure 1. The orientation of the drag vector should be noted particularly.

The model used in this investigation is representa-

tive of a conventional single-engine military pursuit airplane equipped with a radial engine. A three-view drawing of the 0.1-scale airplane model showing the relative size and location of the streamline nose and the two cowlings is given in figure 2. Although no propeller was used, all nose sections were constructed in such a way that the plane of the propeller was at a fixed distance from the center of gravity.

The model was constructed of laminated mahogany. All exposed surfaces were sprayed with lacquer and a smooth finish was obtained. Both cowlings were modeled after the cowlings used in the investigation reported in reference 6. Details of the test arrangement for the NACA open-nose and high-speed cowlings are given in figures 3 and 4, respectively.

In the NACA high-speed cowling, the cooling air enters at the nose of the forward element of the cowling. This element acts as a spinner and normally rotates with the propeller; the rotating element is fitted internally with cuffs that serve as fairings for the propeller-blade shanks. In the present investigation, no propeller was used and the rotating element was held rigid. The cowling exit arrangement was unconventional in that the cooling air was exhausted through two ducts, one on each side of the fuselage, instead of through the conventional annular exit at the cowling skirt.

The resistance offered by the engine to the air flowing through the cowlings was simulated by perforated plates, the conductances of which could be varied. For both cowlings, the condition of maximum conductance approximated the full-scale conductance of a typical single-row radial-engine installation without cowling flaps.

Typical test set-ups are shown in figure 5.

## TESTS

The streamline nose and the two cowlings were each tested on the fuselage alone and on the complete model. Each combination was tested throughout the range of angle of attack from  $-5^\circ$  to  $18^\circ$  at angles of yaw of  $-5^\circ$ ,  $0^\circ$ , and  $5^\circ$ . Tests were made also in which the angle of yaw was varied from  $10^\circ$  to  $-40^\circ$  with the angle of attack fixed at  $0^\circ$  and  $10^\circ$ .

Most of the tests with the two cowlings were run in the condition representing maximum conductance. Some tests with the NACA open-nose cowling were, however, run with zero conductance: that is, the flow through the cowling was completely stopped.

The tests were made at a dynamic pressure of 65 pounds per square foot, which corresponds to a speed of about 160 miles per hour under standard conditions. The test Reynolds number based on the wing chord was about 888,000. Repeat tests were made at dynamic pressures of 100 and 125 pounds per square foot (corresponding to test Reynolds numbers of 1,190,000 and 1,338,000, respectively) but little scale effect was observed in this range.

### PRESENTATION OF RESULTS

All data are reduced to nondimensional coefficients and are uncorrected for initial asymmetry in the model or air stream, for support-strut interference, and for wind-tunnel wall interference. All coefficients are referred to the system of axes shown in figure 1. The coefficients for the fuselage are based on the wing dimensions. The symbols and coefficients used in this report are defined as follows:

|       |                                         |
|-------|-----------------------------------------|
| $C_D$ | drag coefficient ( $D/qS$ )             |
| $C_L$ | lift coefficient ( $lift/qS$ )          |
| $C_Y$ | lateral-force coefficient ( $Y/qS$ )    |
| $C_m$ | pitching-moment coefficient ( $M/qSc$ ) |
| $C_l$ | rolling-moment coefficient ( $L/qSb$ )  |
| $C_n$ | yawing-moment coefficient ( $N/qSb$ )   |
| $D$   | drag (See fig. 1.)                      |
| $Y$   | lateral force                           |
| $M$   | pitching moment                         |
| $L$   | rolling moment                          |

$N$  yawing moment  
 $q$  dynamic pressure ( $\frac{1}{2}\rho V^2$ )  
 $V$  tunnel air velocity  
 $S$  wing area  
 $c$  wing chord  
 $K$  conductance of cowling  
 $\alpha$  angle of attack (thrust line)  
 $\psi$  angle of yaw

$$C_{Y\psi} \quad \frac{\partial C_Y}{\partial \psi}$$

$$C_{l\psi} \quad \frac{\partial C_l}{\partial \psi}$$

$$C_{n\psi} \quad \frac{\partial C_n}{\partial \psi}$$

The variation of the measured lift, drag, and pitching-moment coefficients with angle of attack for the different combinations tested is given in figures 6 to 8. The effect of yaw on these coefficients is illustrated in figure 9. Typical variations of the lateral-force coefficient, the rolling-moment coefficient, and the yawing-moment coefficient with angle of yaw are presented in figures 10 to 12.

The variation of the stability parameters  $C_{Y\psi}$ ,  $C_{l\psi}$ , and  $C_{n\psi}$  with angle of attack is given in figures 13 and 14. The parameters  $C_{Y\psi}$ ,  $C_{l\psi}$ , and  $C_{n\psi}$  represent the slope, at zero yaw, of the curves of the coefficients against angle of yaw. In figures 13 and 14 the tailed points are the measured slopes. The plain points were evaluated from the data taken at angles of yaw of  $\pm 5^\circ$  by assuming a linear variation of the coefficients in this range. This method of computing the parameters yields results within the practical limits of accuracy for angles

of attack below the stall. For angles of attack above the stall, the parameters evaluated by this method have less significance.

## DISCUSSION

A change in the type or the shape of the cowling on an airplane could affect the stability and control characteristics by virtue of the following effects:

1. A change in the force and moment contribution of the fuselage resulting from a redistribution of area and the altered basic pressure distribution over the fuselage. This pressure distribution is affected also by the conductance of the cowling - at least, over the portion of the fuselage occupied by the cowling. (See reference 4.)

2. Interference effects on the wing and the tail surfaces resulting from the altered flow pattern around the fuselage. These effects may also vary with the conductance of the cowling.

The extent of the effects attributable to changes in the basic pressure distribution can be determined from the results for the fuselage alone. A comparison of the results for the fuselage alone and the results for the complete model should reveal some of the effects attributable to interference. It is appreciated that the comparisons should be made on a relative basis; that is, a given change of moment coefficient is of greater importance in yaw than in pitch because the total moment coefficient is usually less.

Fuselage alone.- The data in figure 6 indicated that the different shapes tested had little effect on the characteristics of the fuselage in pitch. The parameter most affected was the drag coefficient. At high angles of attack the more blunt NACA open-nose cowling had the highest drag, caused possibly by an earlier breakdown of the flow over the cowling. A similar increase in drag is noticeable at an angle of yaw of  $17^\circ$ . (See fig. 9.)

The effect of the different noses on the lateral-stability characteristics should be noted (fig. 13). The fuselage with the NACA high-speed cowling gives values of  $C_{Y\psi}$  about 0.0005 lower and values of  $C_{n\psi}$  as much as

0.0003 more positive, that is, more unstable, than the corresponding parameters for the fuselage with the NACA open-nose cowling. It will be observed also that at large angles of yaw the fuselage with the NACA high-speed cowling developed considerably smaller lateral forces than the fuselage with the NACA open-nose cowling. These results are in agreement with the trends indicated in reference 5. In figure 10 the scatter of the rolling-moment coefficients is so great that no attempt was made to fair a curve through the points.

Complete model.— Apart from slight irregularities in the lift curve at an angle of attack of about  $10^\circ$ , the type of cowling affixed to the fuselage did not appreciably affect the characteristics in pitch of the complete model (figs. 7(a) and 7(b)). The effect of conductance should be noted (figs. 8(a) and 8(b)). With zero conductance, a breakdown of the flow probably occurs at a lower angle of attack. This condition is associated possibly with the development of critical peak pressures on the cowling at lower angles of attack as the conductance of the cowling is decreased. (See reference 4.)

The effect of the three nose shapes on the lateral-stability characteristics of the complete model (fig. 14) is similar to the effects observed on the fuselage alone. The values fluctuate somewhat with angle of attack but it is seen that, in the normal range of angle of attack, the value of  $C_{Y_\psi}$  for the NACA high-speed cowling is always less than the corresponding value for the NACA open-nose cowling and that  $C_{n_\psi}$  is always more positive, that is, more unstable, than the corresponding value of  $C_{n_\psi}$  for the NACA open-nose cowling. The actual increments in these parameters were of the same order of magnitude as those observed for the fuselage alone; that is,  $\Delta C_{Y_\psi} = -0.0005$  and  $\Delta C_{n_\psi} = 0.0003$ . Check tests made of the fuselage in combination with the horizontal and vertical tail surfaces have verified these observations. The absolute differences, however, are small and, if plotted to the scale of figure 11, may be overlooked. In figure 14, fluctuations in the parameter  $C_l$  around  $11^\circ$  or  $12^\circ$  indicate that both wings do not stall simultaneously. The effect of conductance was not investigated very extensively but, at low angles of attack, it appeared to be small.

At large angles of yaw the results for the NACA open-nose cowling show larger lateral-force coefficients and larger yawing moments (fig. 11). The favorable effect of the flow through the cowling on the rolling characteristics at angles of yaw beyond  $15^\circ$  is interesting.

The effect of the cowlings cannot be rigorously separated into changes resulting from the altered basic pressure distribution over the fuselage and changes resulting from interference effects on the flow over the wing and tail surfaces without complete data on all combinations of the wing, tail surfaces, and fuselage. It would appear from a comparison of the results for the fuselage alone and for the complete model, however, that a large percentage of the changes in the stability parameters were primarily attributable to the altered basic pressure distribution over the fuselage.

### CONCLUSIONS

The results of this investigation admit the following conclusions concerning the effect of cowling shape on the stability characteristics of an airplane:

1. Ordinary changes in the type or the shape of a cowling affixed to an airplane are apt to produce appreciable changes in the lateral-stability characteristics of the airplane only if the initial stability is marginal. In the present investigation a radical change in the shape of the cowling resulted in only a small reduction in weathercock stability.

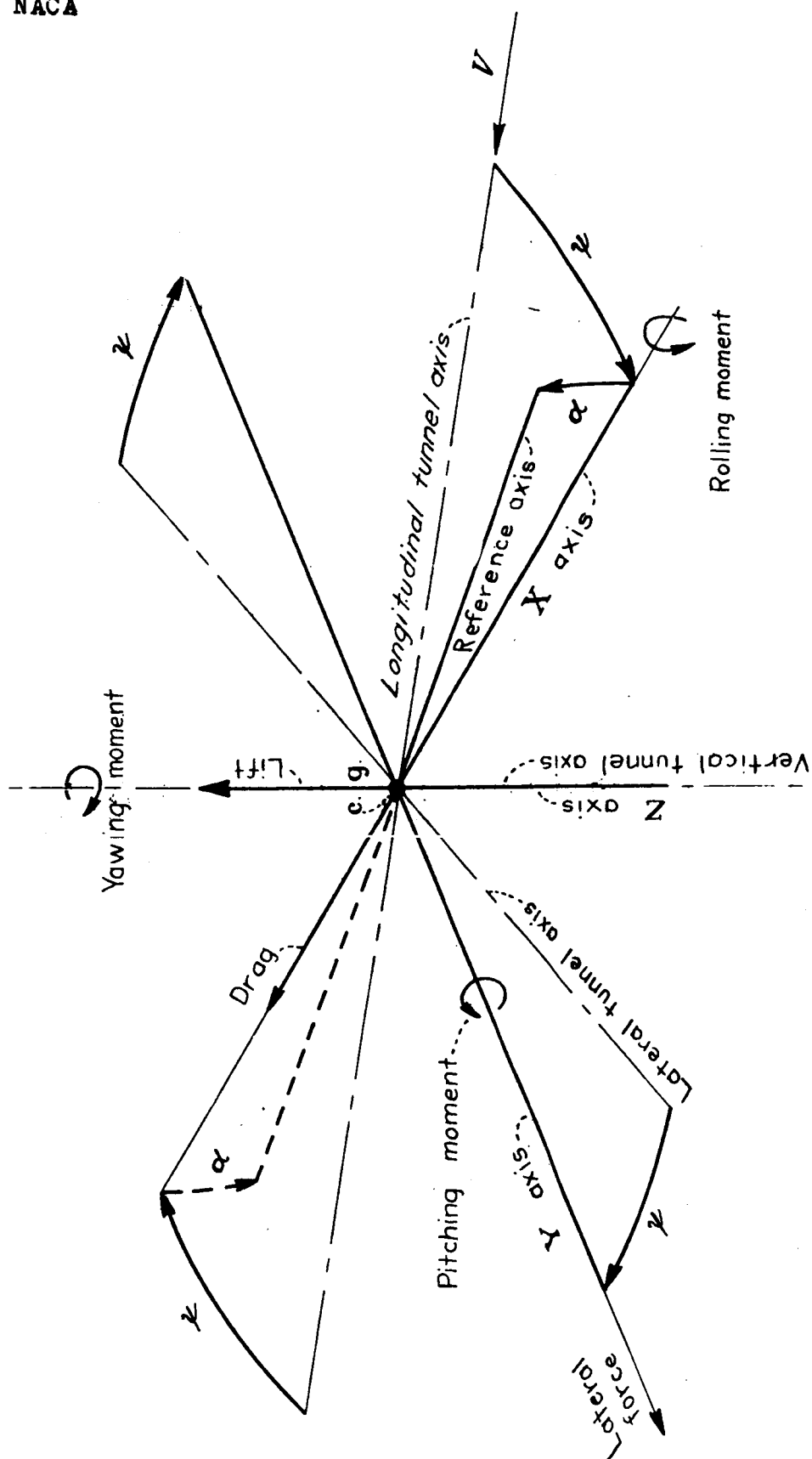
2. Below the critical speed of the cowling, practical variations of engine conductance will have minor effects on the stability characteristics of the airplane, except possibly in the neighborhood of the stall.

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Langley Field, Va.



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**Figure 1.- System of axes used for force and moment measurements in the stability tunnel.**

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Wing area 361 sq in.  
 C.g. location 0.23 mean chord  
 Vertical tail area 38.16 sq in.  
 Horizontal tail area 79.20 sq in.

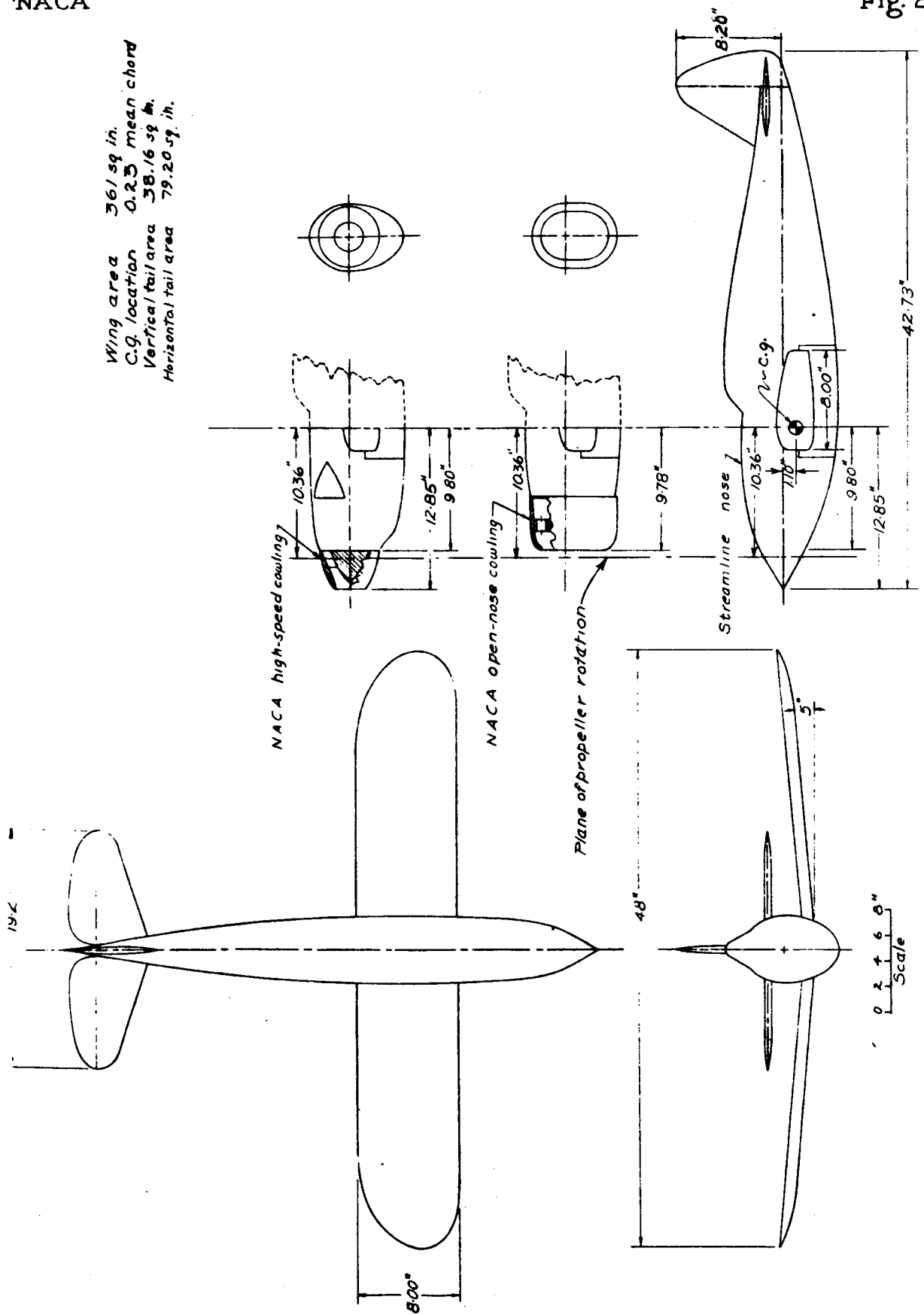


Fig. 2

Figure 2. - Three-view drawing of the 0.1-scale pursuit airplane model with three nose sections.

| Outer surface |       |
|---------------|-------|
| H"            | R"    |
| 0             | 1.976 |
| .026          | 2.114 |
| .054          | 2.164 |
| .108          | 2.234 |
| .214          | 2.330 |
| .322          | 2.412 |
| .430          | 2.472 |
| .540          | 2.532 |
| .646          | 2.574 |
| .754          | 2.616 |
| .862          | 2.654 |
| 1.076         | 2.718 |
| 1.292         | 2.762 |
| 1.508         | 2.788 |
| 1.724         | 2.800 |
| 2.094         | 2.800 |

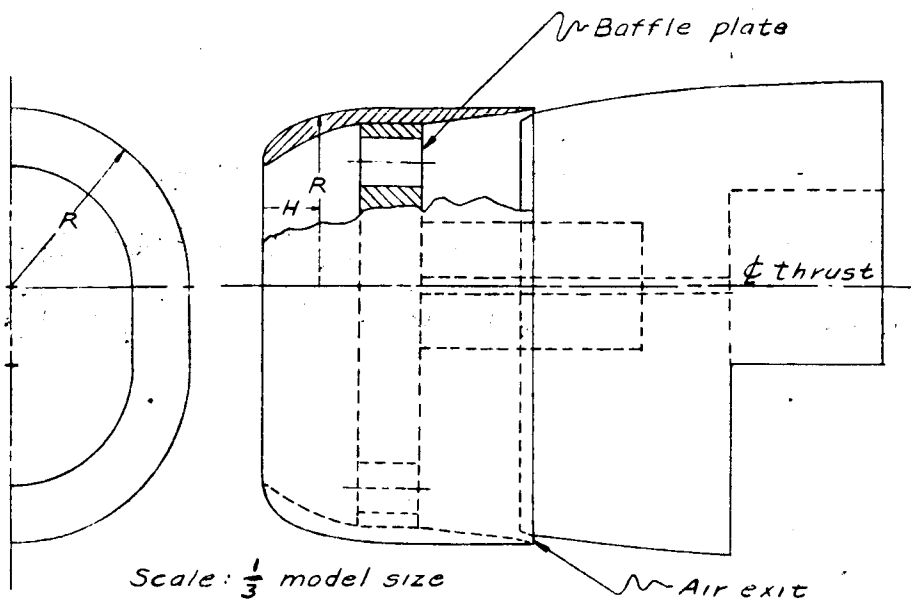


Figure 3.- Test arrangement for NACA open-nose cowling.

| Outer surface |       |
|---------------|-------|
| H"            | R"    |
| 0             | 2.402 |
| .548          | 2.280 |
| 1.096         | 2.142 |
| 1.754         | 1.946 |
| 2.412         | 1.676 |
| 2.630         | 1.562 |
| 2.850         | 1.442 |
| 2.960         | 1.326 |
| 3.014         | 1.264 |
| 3.046         | 1.184 |

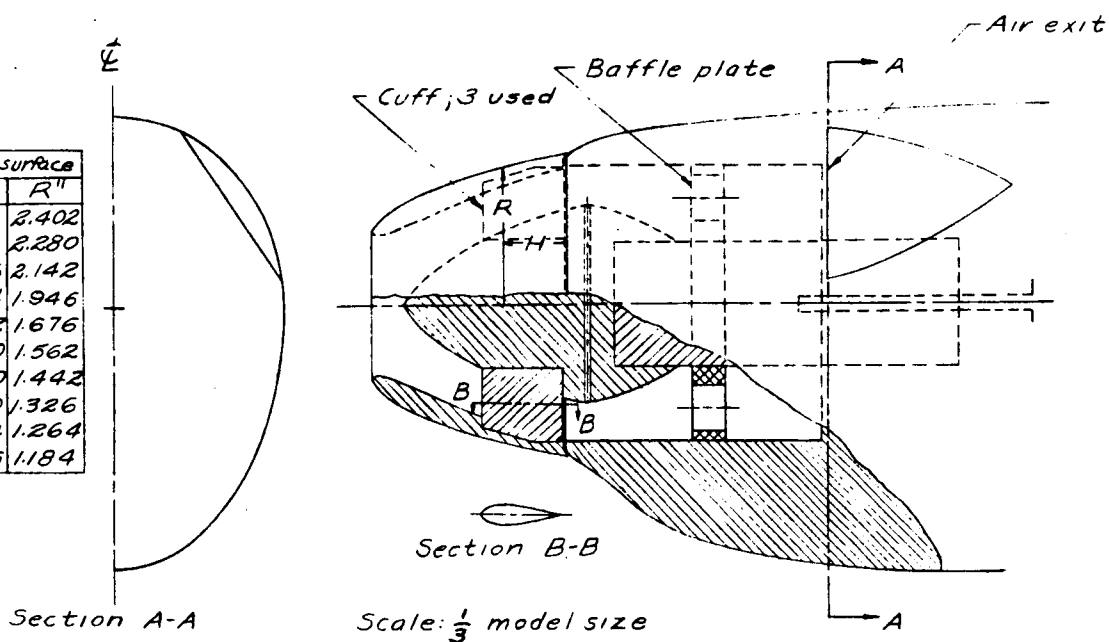
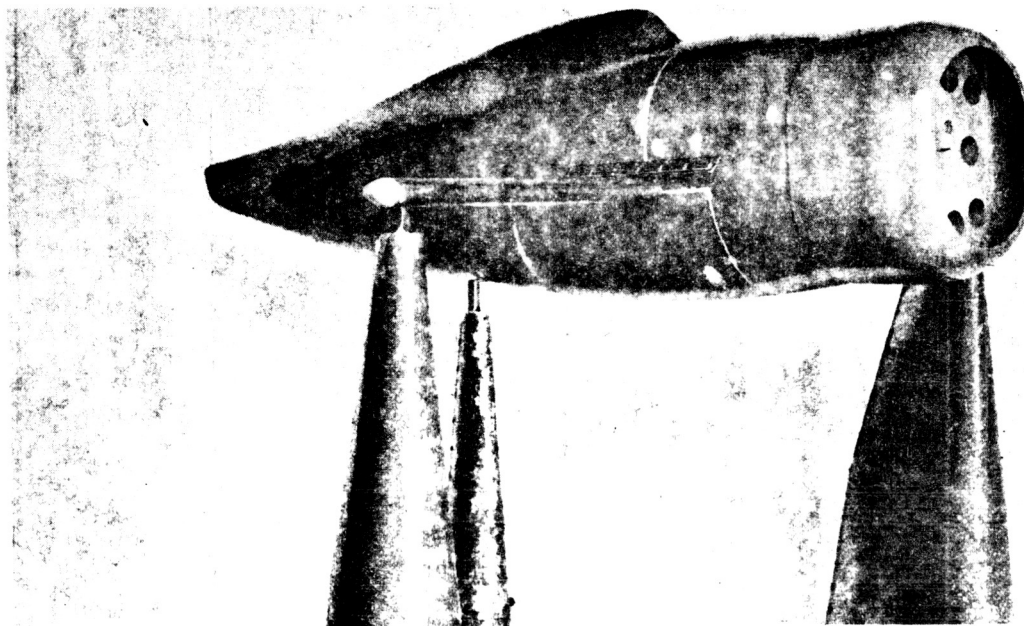
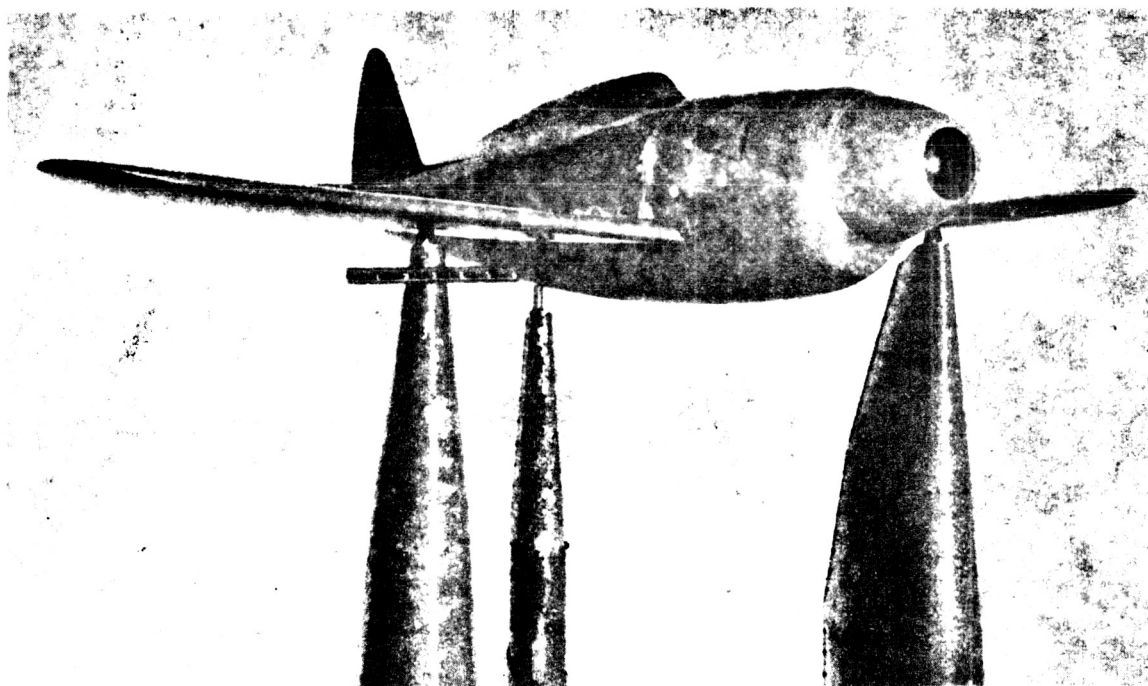


Figure 4.- Test arrangement of NACA high-speed cowling.



(B) FUSELAGE WITH NACA OPEN-NOSE COWLING.



(A) COMPLETE MODEL WITH NACA HIGH-SPEED COWLING.

FIGURE 5. - MODELS WITH TWO TYPES OF NACA COWLING MOUNTED ON SUPPORTS.

○ NACA open-nose cowl  
 + NACA high-speed cowl  
 x Streamline nose

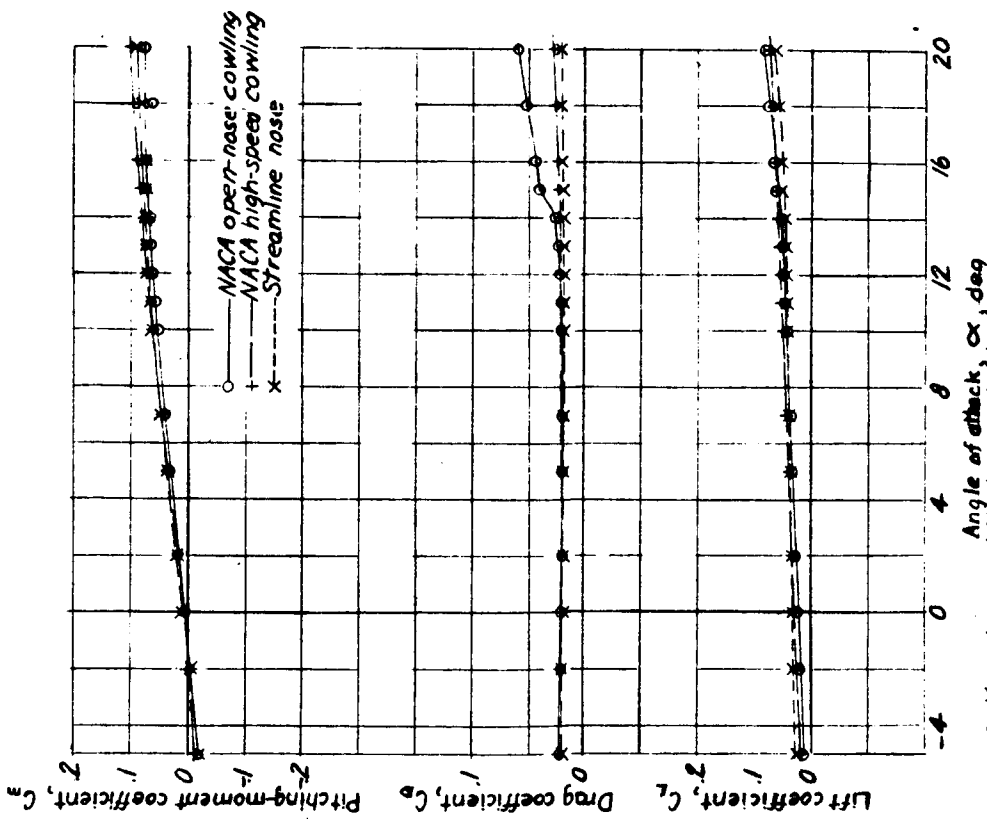
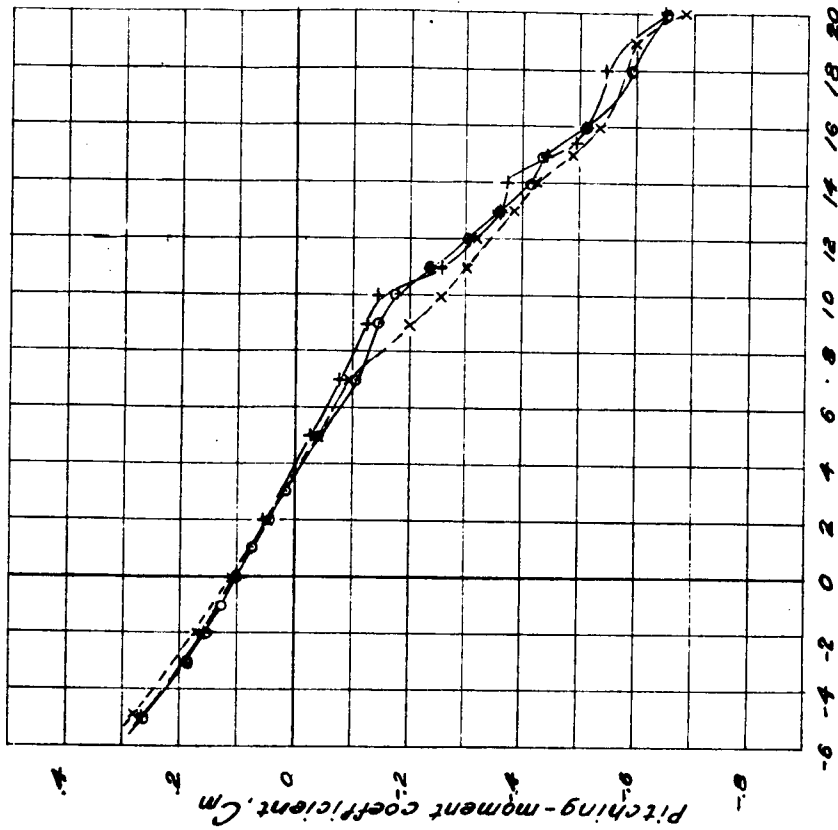
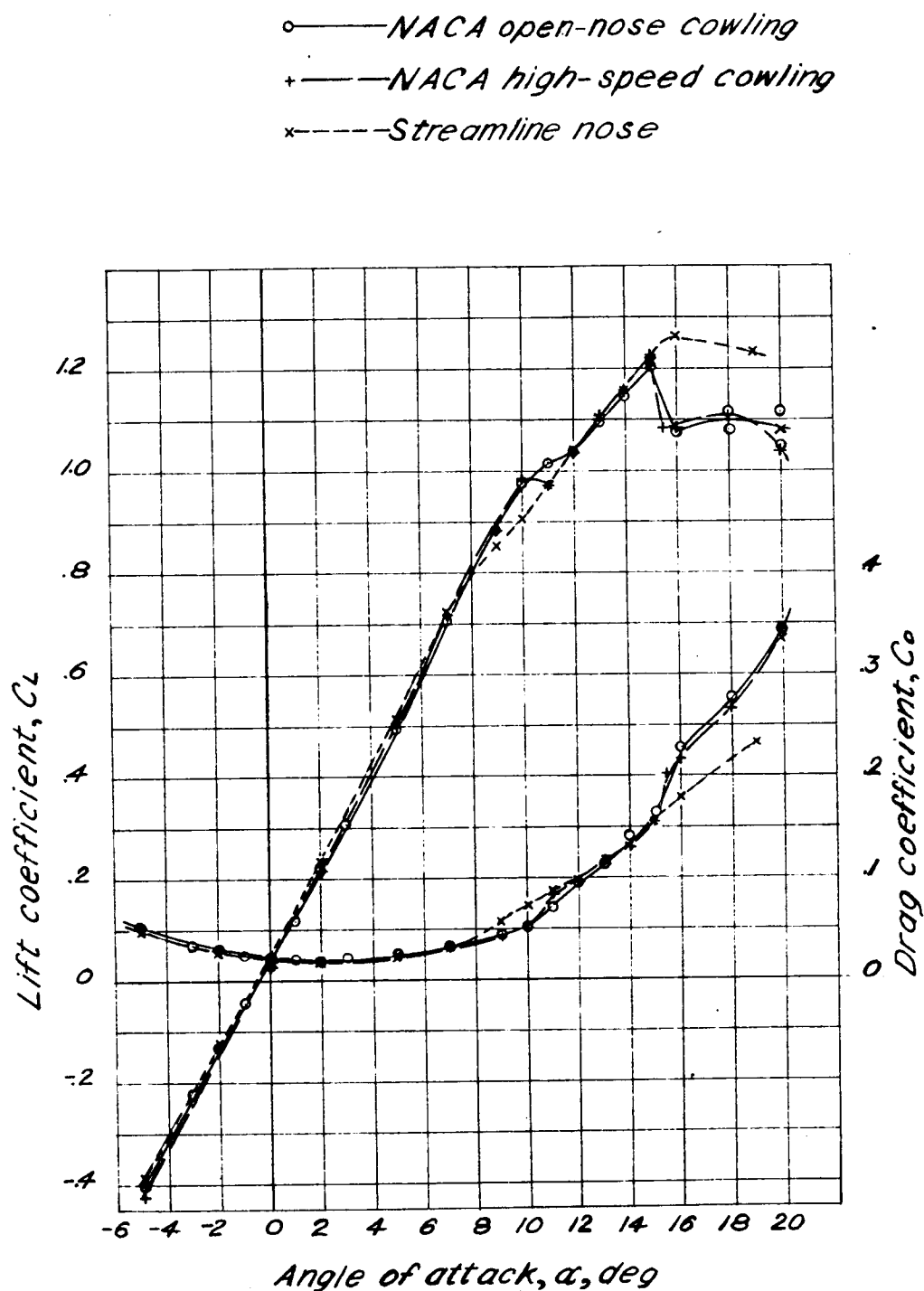


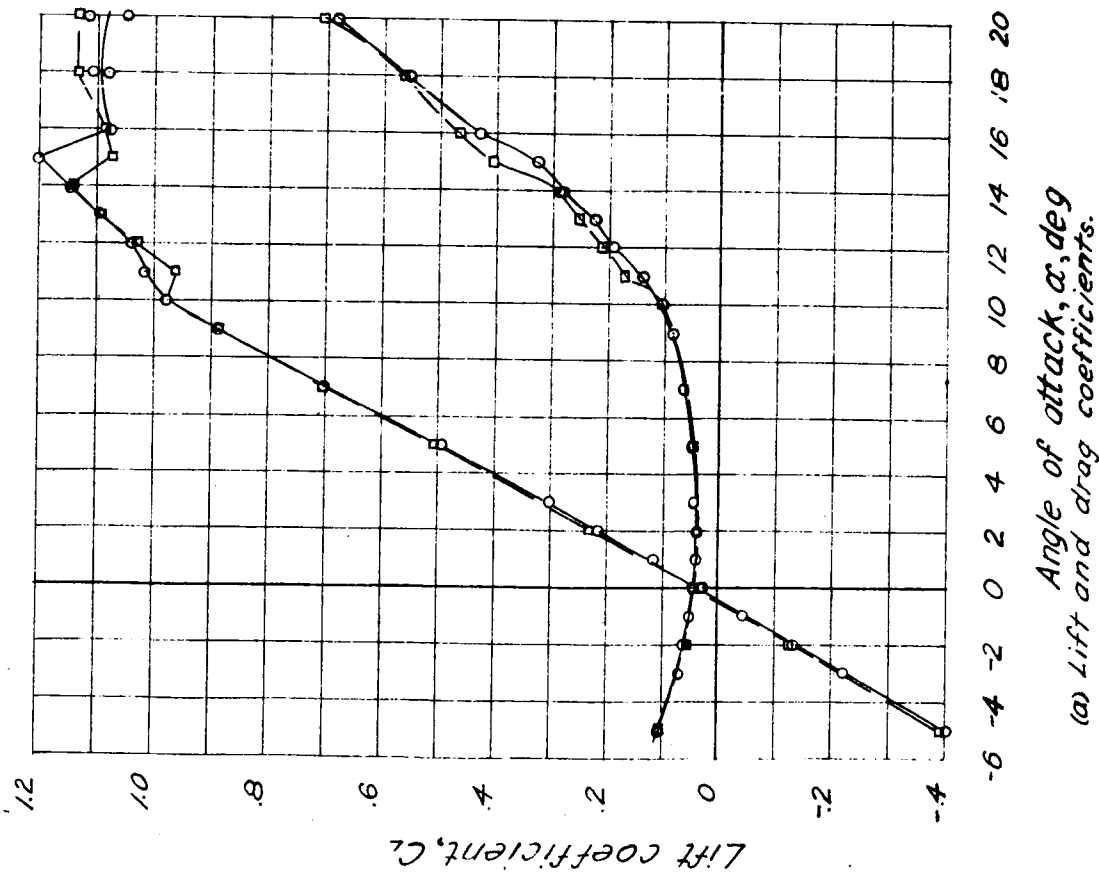
Figure 6.- Variation of lift, drag, and pitching-moment coefficients with angle of attack. Fuse-lage alone;  $K$ , maximum;  $y$ , 0.



(b) Pitching-moment coefficient  
Figure 7.- Concluded.



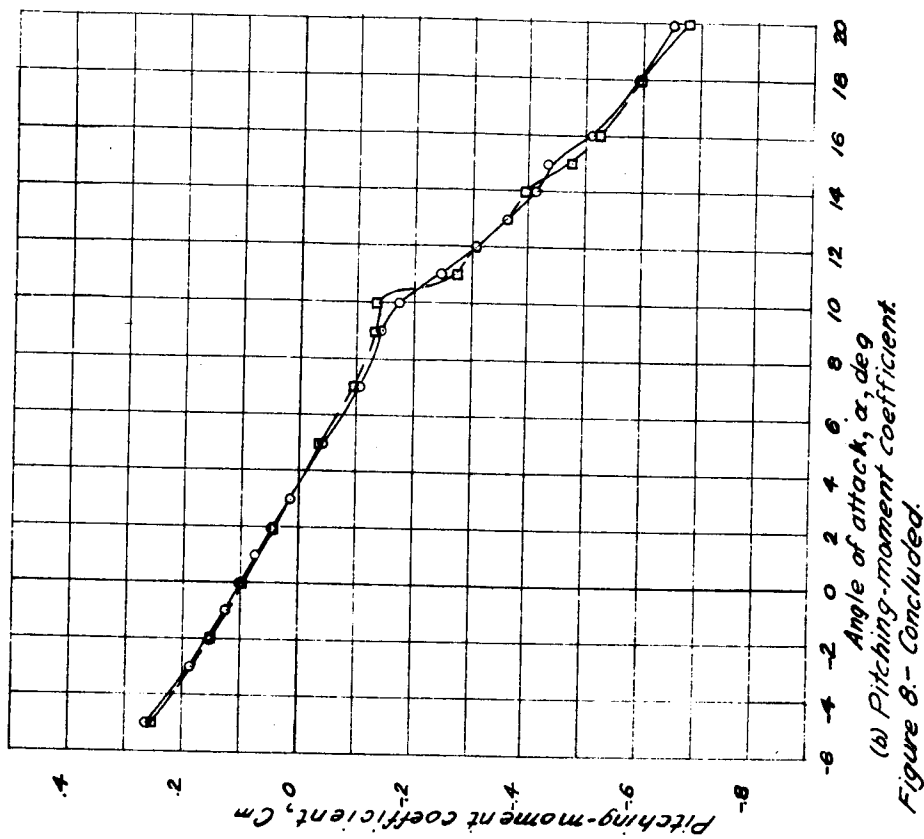
(a) Lift and drag coefficients.  
 Figure 7.—Aerodynamic characteristics of complete model,  $K$ ,  
 maximum;  $\mu$ , 0.



(a) Lift and drag coefficients.

Figure 8.—Effect of conductance on aerodynamic characteristics of complete model NACA open-nose cowl;  $\psi_1 = 0$ .

—○— Maximum conductance  
—□— Zero conductance



(b) Pitching-moment coefficient.

Figure 8.— Concluded.



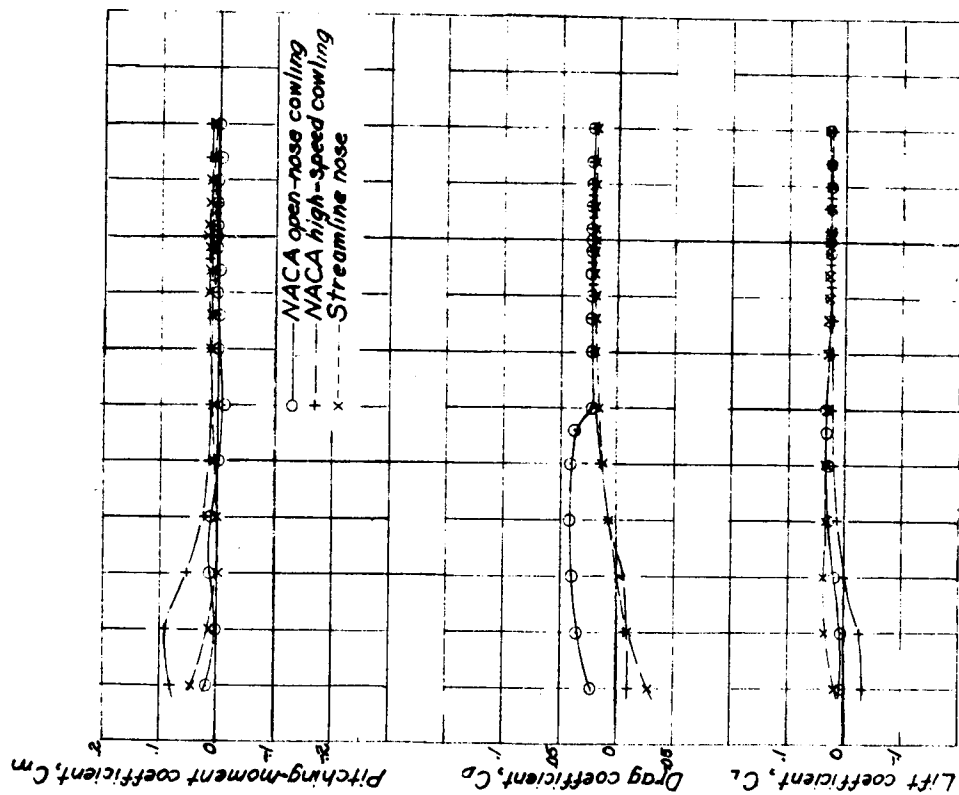
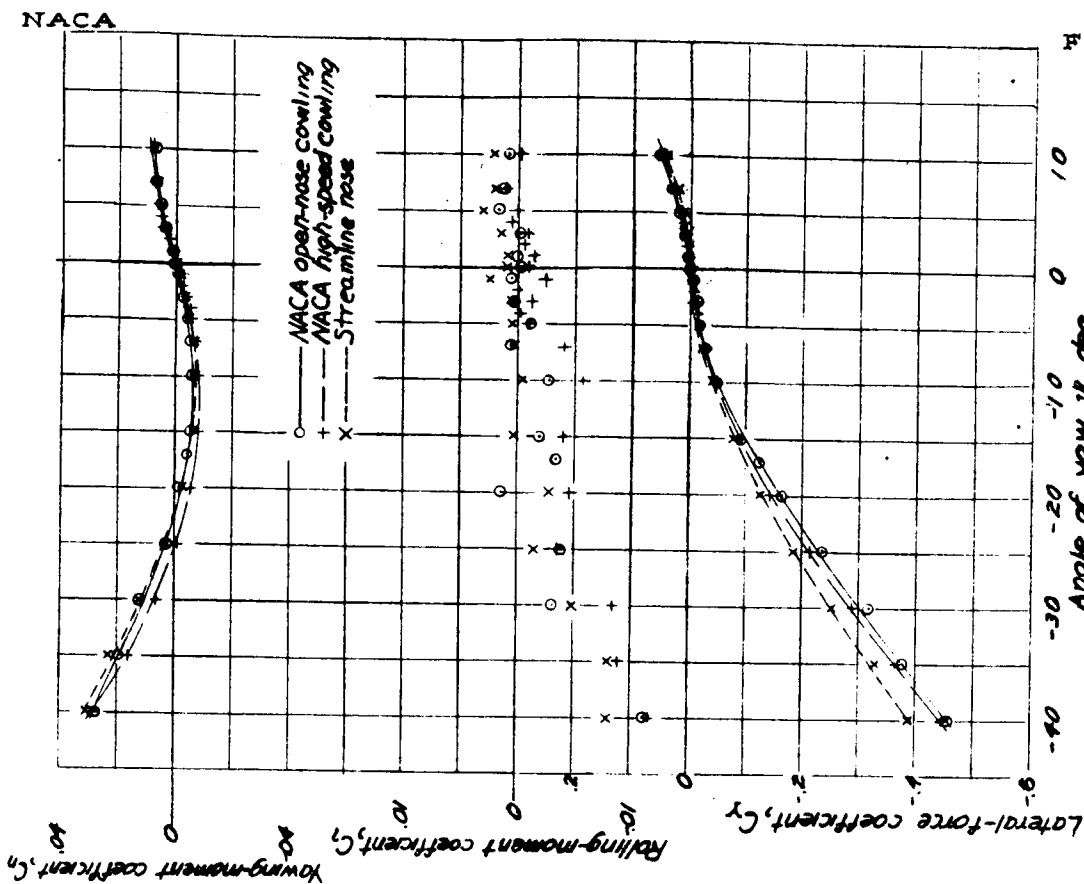


Fig. 9, 10  
Angle of yaw,  $\alpha$ , deg  
Variation of lateral-force, rolling-moment, and yawing-moment coefficients with angle of yaw. Fuselage alone;  $\alpha$ , maximum;  $\alpha$ , 0.

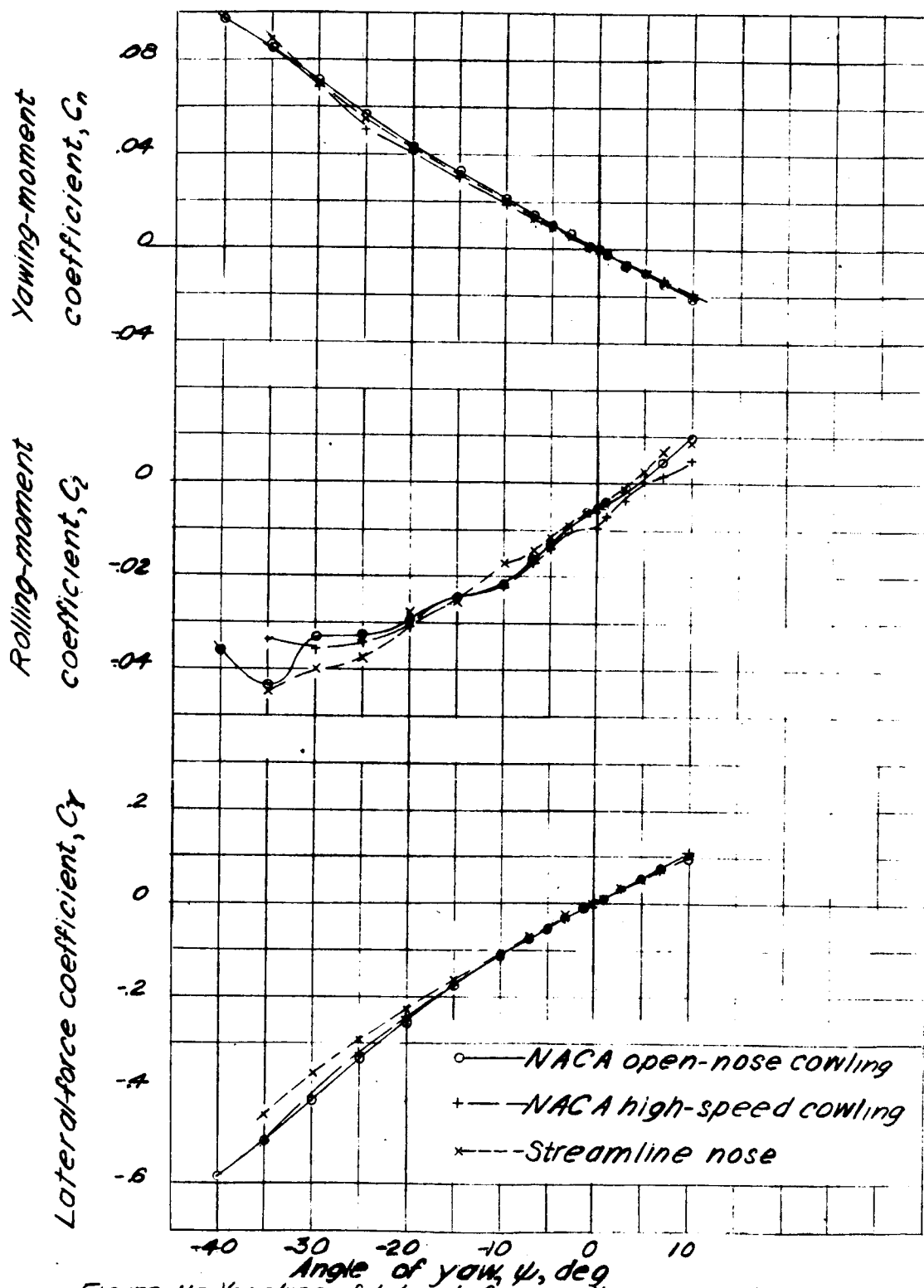


Figure 11.- Variation of lateral-force, rolling-moment, and yawing-moment coefficients with angle of yaw. Complete model;  $K$ , maximum;  $\alpha$ , 0.

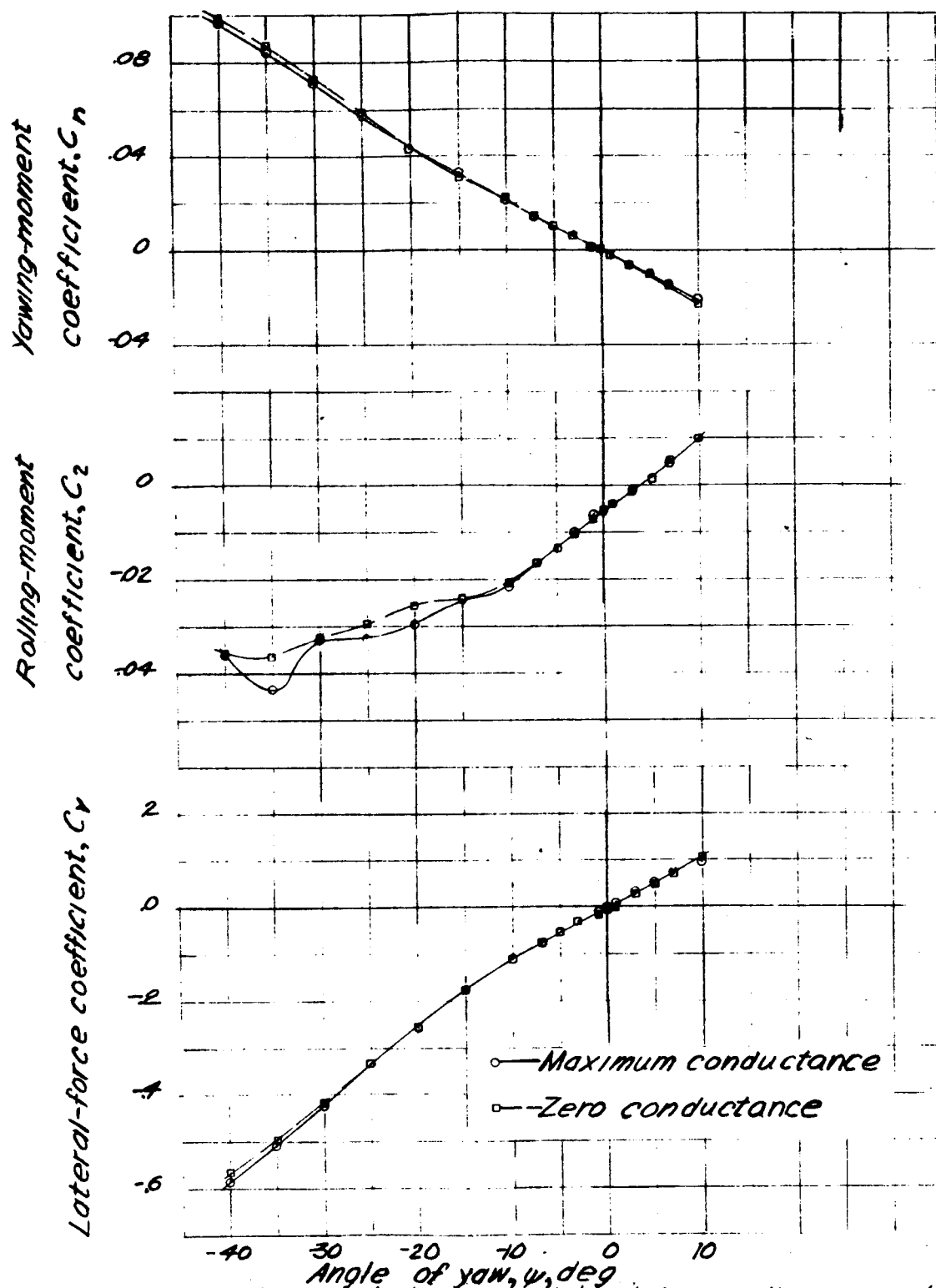


Figure 12.—Effect of conductance on lateral-force, rolling-moment, and yawing-moment coefficients. Complete model; NACA open-nose cowling;  $\alpha$ , 0.

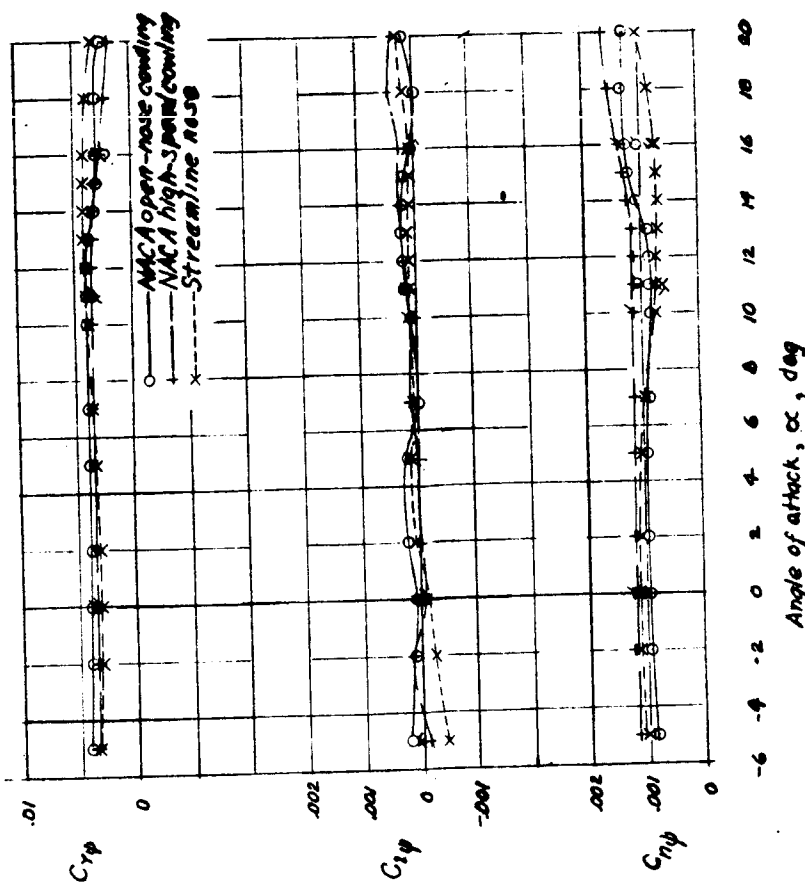


Figure 13.- Variation of  $C_{yp}$ ,  $C_{yp}'$ , and  $C_{np}$  with angle of attack. Fuselage alone;  $K_{\text{maximum}}$ .

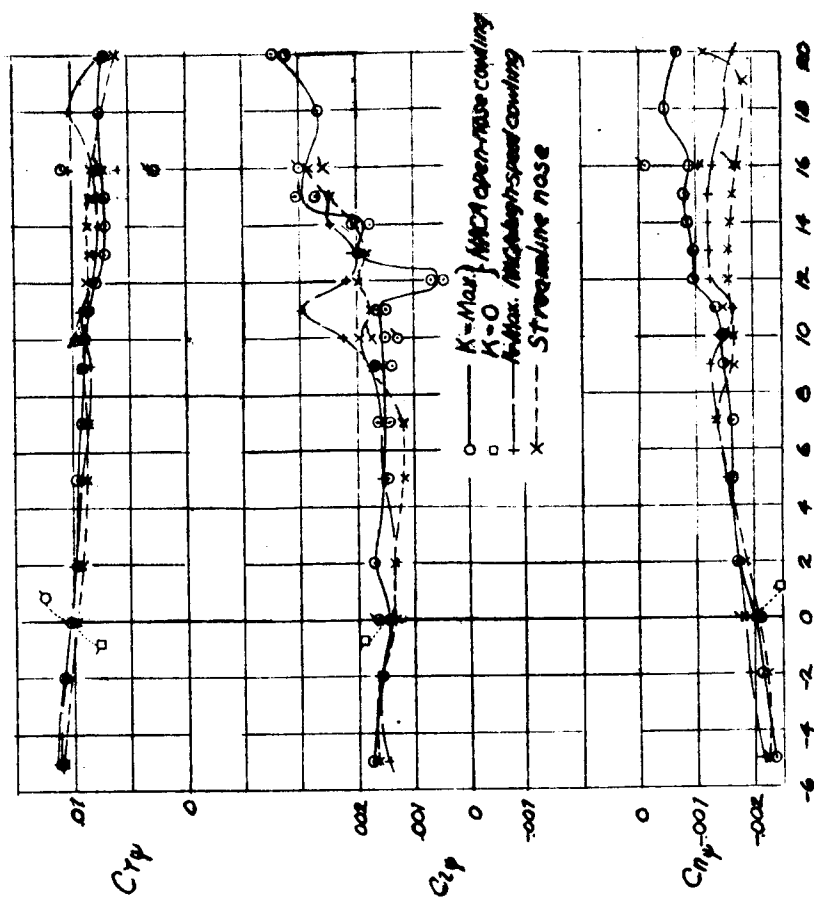


Figure 14.- Variation of  $C_{yp}$ ,  $C_{yp}'$ , and  $C_{np}$  with angle of attack. Complete model.

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

# WARTIME REPORT

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SOME THEORETICAL CONSIDERATIONS OF LONGITUDINAL  
STABILITY IN POWER-ON FLIGHT WITH SPECIAL  
REFERENCE TO WIND-TUNNEL TESTING

By Charles J. Donlan

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# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## ADVANCE RESTRICTED REPORT

### SOME THEORETICAL CONSIDERATIONS OF LONGITUDINAL STABILITY IN POWER-ON FLIGHT WITH SPECIAL REFERENCE TO WIND-TUNNEL TESTING

By Charles J. Donlan

#### SUMMARY

Some problems relating to longitudinal stability in power-on flight are considered. A derivation is included which shows that, under certain conditions, the rate of change of the pitching-moment coefficient with lift coefficient as obtained in wind-tunnel tests simulating constant-power operation is directly proportional to one of the indices of stability commonly associated with flight analysis, the slope of the curve relating the elevator angle for trim and lift coefficient (or velocity). The necessity of analyzing power-on wind-tunnel data for trim conditions is emphasized and a method is provided for converting data obtained from constant-thrust tests to simulated constant-throttle flight conditions. It is demonstrated how a downward tail load required to trim an airplane results in decreased stability in power-on flight and why a longitudinal center-of-gravity movement is likely to affect the stability characteristics less in power-on flight than in power-off flight.

#### INTRODUCTION

The effect of running propellers on the longitudinal-stability characteristics of airplanes has been appreciated for many years. The increased use of powered models for wind-tunnel testing has greatly increased the amount of empirical information on the subject. The evaluation of wind-tunnel data secured with a power model, however, demands a greater appreciation of trim conditions than the evaluation of conventional power-off data. Data obtained from tests made with the propeller thrust held constant (reference 1), for example, consequently require an interpretation different from data obtained either from tests in

which the propeller thrust is permitted to vary or from tests in which the propeller is absent.

The purpose of the present paper is to correlate the different test procedures used in testing a model equipped with running propellers and to establish the significance of the data obtained for the determination of the longitudinal-stability characteristics. The slopes of the wind-tunnel pitching-moment curves are correlated with the index of stick-fixed stability commonly used in free-flight test work - the variation of elevator angle for trim with speed or lift coefficient. It is believed that a demonstration of the quantitative relationships existing between these different indices of stability would aid in the correlation of flight and wind-tunnel tests. The paper also considers the magnitude of the changes in the stick-fixed power-on stability of an airplane resulting from a change in center-of-gravity position and the different tail loads necessary for trim.

#### SYMBOLS AND FORMULAS

|           |                                                                                                                       |
|-----------|-----------------------------------------------------------------------------------------------------------------------|
| $C_L$     | lift coefficient                                                                                                      |
| $C_{L_t}$ | lift coefficient of horizontal tail                                                                                   |
| $C_D$     | drag coefficient                                                                                                      |
| $C_{X_W}$ | longitudinal force coefficient of wing                                                                                |
| $C_{m_o}$ | pitching-moment coefficient of wing-fuselage<br>combination about aerodynamic center                                  |
| $C_m$     | pitching-moment coefficient of airplane excluding<br>propeller-thrust component                                       |
| $C_M$     | resultant pitching-moment coefficient (includes<br>propeller-thrust component)                                        |
| $c$       | mean aerodynamic chord (M.A.C.)                                                                                       |
| $c_e$     | mean elevator chord                                                                                                   |
| $C_g$     | ratio of distance of center of gravity back of<br>leading edge of mean aerodynamic chord to mean<br>aerodynamic chord |

|              |                                                                                                                          |
|--------------|--------------------------------------------------------------------------------------------------------------------------|
| $C_a$        | ratio of distance of aerodynamic center of wing back of leading edge of mean aerodynamic chord to mean aerodynamic chord |
| $z$          | distance of mean aerodynamic chord below center of gravity                                                               |
| $h$          | distance of thrust axis below center of gravity                                                                          |
| $l_1$        | distance from center of gravity to hinge line of horizontal tail                                                         |
| $l_2$        | distance from center of gravity to plane of propeller disk                                                               |
| $S_w$        | wing area                                                                                                                |
| $S_t$        | horizontal tail area                                                                                                     |
| $S_e$        | elevator area                                                                                                            |
| $S_p$        | propeller-disk area ( $\pi D^2/4$ )                                                                                      |
| $D$          | propeller diameter                                                                                                       |
| $n$          | propeller rotational speed                                                                                               |
| $V$          | velocity of flight                                                                                                       |
| $V_s$        | slipstream velocity                                                                                                      |
| $J$          | advance-diameter ratio ( $V/nD$ )                                                                                        |
| $K$          | function of $J$ and propeller-blade angle for an inclined propeller (reference 2)                                        |
| $T$          | thrust                                                                                                                   |
| $\alpha$     | angle of attack                                                                                                          |
| $\gamma$     | flight-path angle                                                                                                        |
| $\theta$     | angle of airplane to horizontal                                                                                          |
| $\alpha_t$   | angle of attack of horizontal tail                                                                                       |
| $\epsilon_w$ | downwash angle at tail due to wing                                                                                       |



- $\epsilon_p$  downwash angle at tail due to propeller  
 $i_t$  initial horizontal tail setting  
 $\delta_e$  elevator angle  
 $W$  weight (mg)  
 $m$  mass of the airplane  
 $g$  gravity  
 $k$  mechanical advantage of elevator-control system  
 $k_Y$  radius of gyration about Y axis  
 $P_D$  component of thrust coefficient along X axis  
     during motion  $\left[ (T_c' + \frac{1}{2} T_c' u) \cos \alpha \right]$   
 $P_L$  component of thrust coefficient along Z axis  
     during motion  $\left[ (T_c' + \frac{1}{2} T_c' u) \sin \alpha \right]$   
 $C_h$  elevator hinge-moment coefficient

$$R = (V_s/V)^2$$

$$T_c' = \frac{T}{\frac{1}{2} \rho S_w V^2}$$

$$T_{c'u}' = \frac{dT_c'}{du'}$$

$$k_1 = \frac{l_e}{c_e} \frac{S_w}{S_e}$$

$$R_Y = k_Y/C$$

$$u' = \Delta V/V$$

$$C_{D_1} = \frac{W}{\frac{1}{2} \rho S_w V_o^2} \sin \gamma_o$$

$$C_{L_1} = \frac{W}{\frac{1}{2} \rho S_w V_o^2} \cos \gamma_o$$

$$C_{m_{u^1}} = \frac{\partial C_m}{\partial u^1}$$

$$C_{M_{u^1}} = \frac{\partial C_M}{\partial u^1}$$

$$C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha}$$

$$C_{m_\delta} = \frac{\partial C_m}{\partial \delta_o}$$

$$C_{h_{\delta_o}} = \frac{\partial C_h}{\partial \delta_o}$$

$$C_{h_{\alpha_t}} = \frac{\partial C_h}{\partial \alpha_t}$$

$$C_{L_\alpha} = \frac{dC_L}{d\alpha} \text{ per radian}$$

$$C_{D_\alpha} = \frac{dC_D}{d\alpha} \text{ per radian}$$

$$f = \frac{\text{stick force}}{\frac{1}{2} \rho S_w V_o^2}$$

$$\mu = \frac{m}{\rho S_w c}$$

$$K_1 = \frac{R_y^2 \rho V_o^2}{8\mu \left( \frac{W}{S_w} \right)}$$

Subscripts:

o equilibrium condition

nt tail removed

THE DETERMINATION OF LONGITUDINAL STABILITY  
CHARACTERISTICS FROM WIND-TUNNEL TESTS OF  
A MODEL EQUIPPED WITH RUNNING PROPELLERS

A significance of wind-tunnel data obtained with running propellers for the determination of longitudinal-stability characteristics can be evaluated by comparison with existing criteria for longitudinal stability commonly used in flight-test work. One of the simpler experiments to perform in flight consists in determining the position of the elevator required for trim over a range of flying speeds with the throttle setting fixed. When the elevator positions thus obtained are plotted against lift coefficient (or airspeed), an index of the stick-fixed stability, which is commonly used in flight work, results.

This index of stability is the slope  $(d\delta_e/dC_L)_{\text{trim}}$  [or  $(d\delta_e/dV)_{\text{trim}}$ ]. This criterion of stability is also asso-

ciated with the slope of the resultant pitching-moment curve - a curve readily obtained from wind-tunnel data. It is of interest then to know specific quantitative relationships involving these quantities in power-on flight. These relationships are subsequently developed.

The reader who wishes to acquaint himself with the desired fundamental relationships without familiarizing himself with the details of proof may omit the following development and turn immediately to equation (10).

### Theory

The equations of equilibrium for an airplane in power-on flight subjected to the force system illustrated in figure 1 may be written as follows:

$$T \cos \alpha - C_D \frac{1}{2} \rho S_w V^2 - mg \sin \gamma = 0 \quad (1a)$$

$$-T \sin \alpha - C_L \frac{1}{2} \rho S_w V^2 + mg \cos \gamma = 0 \quad (1b)$$

$$Th + C_m \frac{1}{2} \rho S_w c V^2 = 0 \quad (= C_m \frac{1}{2} \rho S_w c V^2) \quad (1c)$$

$$\frac{1}{2} \rho S_e c_e V^2 C_h = 0 \quad (1d)$$

If it is assumed in equation 1 that the throttle setting is fixed, any small increments imposed on any of the variables must be such as to maintain equilibrium. After a small change in attitude, equation (1) may be transformed into the following set of equations:

$$(2P_D - 2C_D) \frac{\Delta V}{V} + [(C_L - C_{L_1}) - C_{D_\alpha}] \Delta \alpha - C_{L_1} \Delta \gamma = 0 \quad (2a)$$

$$(2P_L + 2C_L) \frac{\Delta V}{V} + [(C_D + C_{D_1}) + C_{L_\alpha}] \Delta \alpha + C_{D_1} \Delta \gamma = 0 \quad (2b)$$

$$\frac{2\mu}{R_Y^2} C_{H_u} \frac{\Delta V}{V} + \frac{2\mu}{R_Y^2} C_{m_\alpha} \Delta \alpha + \frac{2\mu}{R_Y^2} C_{m_\delta} \Delta \delta_e = 0 \quad (2c)$$

$$2C_h \frac{\Delta V}{V} + C_{h_\alpha} \Delta \alpha + C_{h_\delta} \Delta \delta_e = k_1 f \quad (2d)$$

From the foregoing equations the following relationships between the increments  $\delta_e$ ,  $\alpha$ , and  $f$  may be established:

$$\frac{d\delta_e}{d\alpha} = \frac{\Delta\delta_e}{\Delta\alpha} = -\frac{E}{F} = \frac{k_1 f}{G} \quad (3)$$

where

$$E = -\frac{4\mu C_{m_\alpha}}{R_Y^2} [C_{L_1} (C_L + P_L) - C_{D_1} (C_D - P_D)] \\ + \frac{2\mu C_{H_u}}{R_Y^2} [C_{L_1} (C_{L_\alpha} + C_D) + C_{D_1} (C_L - C_{D_\alpha})]$$

$$F = \frac{4\mu C_{m_\delta}}{R_Y^2} [C_L (C_L + P_L) - C_{D_1} (C_D - P_D)]$$

$$G = -\frac{4\mu}{R_Y^2} (C_{h_\delta} C_{m_\alpha} - C_{h_\alpha} C_{m_\delta}) [C_{L_1} (C_L + P_L) - C_{D_1} (C_D - P_D)] \\ + \frac{2\mu}{R_Y^2} C_{h_\delta} (C_{H_u} - 2C_{m_\delta}) [C_{L_1} (C_L + P_L) - C_{D_1} (C_D - P_D)]$$

The term  $E$  is the familiar constant term associated with the biquadratic equation for power-on stick-fixed stability. If the airplane is suddenly displaced from its equilibrium condition, it will continue to diverge from this position if  $E$  is negative. Hence, for stability,  $E$  must remain positive. If the substitutions

$$C_{L_1} = \frac{W}{\frac{1}{2}\rho S_w V_o^2} (\cos \gamma_o)$$

and

$$C_{D_1} = \frac{W}{\frac{1}{2}\rho S_w V_o^2} (\sin \gamma_o)$$

are made,  $E$  may be rewritten as

$$E = - \frac{4\mu}{R_Y^2} \frac{W}{S_w} \frac{2}{\rho V_o^2} \left\{ C_{m_\alpha} \left[ \cos \gamma_o (C_L + P_L) - \sin \gamma_o (C_D - P_D) \right] \right. \\ \left. - \frac{1}{2} C_{M_{u_1}} \left[ \cos \gamma_o (C_{L_\alpha} + C_D) + \sin \gamma_o (C_D + C_{D_\alpha}) \right] \right\} \quad (4)$$

Equation (4) is the general expression for  $E$  for power-on flight. For small values of  $\gamma_o$ ,

$\sin \gamma_o (C_D - P_D) \ll \cos \gamma_o (C_L + P_L)$  and

$\sin \gamma_o (C_D + C_{D_\alpha}) \ll \cos \gamma_o (C_{L_\alpha} + C_D)$ ; also,  $P_L \ll C_L$ ,  $C_D \ll C_{L_\alpha}$ ,

and  $\cos \gamma_o = 1$ . With these simplifications, equation (4)

reduces to

$$E = - \frac{4\mu}{R_Y^2} \frac{W}{S_w} \left( C_{m_\alpha} C_L - \frac{C_{M_{u_1}}}{2} C_{L_\alpha} \right) \left( \frac{2}{\rho V_o^2} \right)$$

or

$$\left(\frac{K_1}{C_L}\right) E = \left(C_{m\alpha} - \frac{1}{2} \frac{C_L \alpha}{C_L} C_{M_u'}\right) \quad (4a)$$

where

$$K_1 = - \frac{R_Y^2 \rho V_o^2}{8\mu \frac{W}{S_w}}$$

Further, if the notation is changed

$$\left(\frac{K_1}{C_L}\right) E = \left(\frac{\partial C_m}{\partial \alpha}\right) - \frac{1}{2} \frac{dC_L}{d\alpha} \frac{1}{C_L} \left(v \frac{\partial C_m}{\partial v} + v \frac{dT_{c'}}{dv} \frac{h}{c}\right)$$

dividing by  $dC_L/d\alpha$  yields

$$\frac{K_1}{C_L} \frac{d\alpha}{dC_L} E = \frac{\partial C_m}{\partial C_L} - \frac{1}{2C_L} \left(v \frac{\partial C_m}{\partial v} + v \frac{dT_{c'}}{dv} \frac{h}{c}\right) \quad (5)$$

Now, from equation (1c)

$$C_M = C_m + T_{c'} \frac{h}{c} = 0$$

and on differentiation

$$\left(\frac{dC_M}{dC_L}\right)_{C_M=0} = \frac{dC_m}{dC_L} + \frac{dT_{c'}}{dC_L} \frac{h}{c} \quad (6)$$

For the small values of  $\theta$

$$W = \frac{1}{2} \rho S_w V^2 C_L$$

or

$$V^2 C_L = \text{constant (approx.)}$$

Hence

$$2VC_L dV + V^2 dC_L = 0$$

and

$$\frac{dV}{dC_L} = - \frac{V}{2C_L}$$

Under the conditions  $V^2 C_L = \text{constant}$  and constant throttle operation

$$C_m = C_m(C_L, V)$$

or

$$\frac{dC_m}{dC_L} = \left( \frac{\partial C_m}{\partial C_L} \right)_V + \left( \frac{\partial C_m}{\partial V} \right)_{C_L} \left( \frac{dV}{dC_L} \right)_{V^2 C_L = \text{constant and constant throttle}}$$

Further

$$\frac{dC_m}{dC_L} = \frac{\partial C_m}{\partial C_L} - \frac{V}{2C_L} \left( \frac{\partial C_m}{\partial V} \right) \quad (6a)$$

and

$$\frac{dT_c'}{dC_L} = \frac{dT_c'}{dV} \frac{dV}{dC_L} = - \frac{V}{2C_L} \left( \frac{dT_c'}{dV} \right) \quad (6b)$$

The substitution of equations (6a) and (6b) in equation (6) gives

$$\left( \frac{dC_m}{dC_L} \right)_{C_H=0} = \left( \frac{\partial C_m}{\partial C_L} \right) - \frac{1}{2C_L} \left[ V \left( \frac{\partial C_m}{\partial V} \right) + V \left( \frac{dT_c'}{dV} \right) \left( \frac{h}{c} \right) \right] \quad (7)$$

and now equation (5) may be rewritten as

$$\frac{K_1}{C_L} \frac{1}{\frac{dC_L}{d\alpha}} E = \left( \frac{dC_M}{dC_L} \right)_{C_M=0} \quad (8)$$

Equation (7) establishes the relation between the constant term  $E$  of the stability biquadratic for power-on flight and the slope of the resultant pitching-moment curve for power-on flight. It should be noted, however, that in equations (6) and (7), the terms  $C_L$  and  $dC_M/dC_L$  must include any effects of the slipstream; that is,

$$C_L = (C_L)_{T_{c'}=0} + (\Delta C_L)_{\text{slipstream}} \quad (9a)$$

and

$$\frac{dC_L}{d\alpha} = \left( \frac{\partial C_L}{\partial \alpha} \right)_{T_{c'}=0} + \left( \frac{\partial C_L}{\partial V} \right)_{\alpha} \frac{dV}{dT_{c'}} \left( \frac{dT_{c'}}{d\alpha} \right)_{V^2 C_L = \text{constant and constant throttle}} \quad (9b)$$

The relationship  $d\delta_e/d\alpha = -E/F$  can now be developed in terms of the quantity  $\left( \frac{dC_M}{dC_L} \right)_{C_M=0}$

From equation (3)

$$\left( \frac{d\delta_e}{d\alpha} \right)_{C_M=0} = -\frac{E}{F} = - \left\{ \frac{\frac{W}{S_w} \frac{2}{\rho V_0^2} C_L C_{L\alpha} \left( \frac{dC_M}{dC_L} \right)_{C_M=0}}{C_{m_0} [C_{L_1} (C_L + P_L) - C_{D_1} (C_D - P_D)]} \right\}$$

Now

$$\begin{aligned} C_{L_1} (C_L + P_L) - C_{D_1} (C_D - P_D) &= \frac{W}{S_w} \frac{2}{\rho V_0^2} \left( \frac{W}{S_w} \frac{2}{\rho V_0^2} - C_L \frac{dT_{c'}}{dC_L} \sin \theta_0 \right) \\ &= \left( \frac{W}{S_w} \frac{2}{\rho V_0^2} \right)^2 = C_L^2 \end{aligned}$$



for small values of  $\theta$ . Hence

$$\left(\frac{d\delta_e}{d\alpha}\right)_{C_M=0} = \frac{C_L^2 C_{L\alpha} \left(\frac{dC_M}{dC_L}\right)_{C_M=0}}{-C_{m\delta} C_L^2} = \frac{C_{L\alpha} \left(\frac{dC_M}{dC_L}\right)_{C_M=0}}{-C_{m\delta}}$$

and

$$\left(\frac{d\delta_e}{dC_L}\right)_{C_M=0} = \frac{\left(\frac{dC_M}{dC_L}\right)_{C_M=0}}{-\frac{\partial C_M}{\partial \delta_e}} \quad (10)$$

Equation (10) states precisely that for power-on flight with a fixed throttle setting, the flight index of stick-

fixed stability  $\left(\frac{d\delta_e}{dC_L}\right)_{C_M=0}$  is proportional to the slope

of the resultant pitching-moment curve at only the point  $C_M = 0$ . The proportionality factor is the negative reciprocal of the elevator effectiveness parameter,  $C_{m\delta}$ .

In equation (10) the slope  $dC_M/dC_L$  may be evaluated from the wind-tunnel test data. The slope of the wind-tunnel resultant pitching-moment curve, however, depends on the test procedure adopted for the investigation. In the present analysis two procedures for testing a model equipped with running propellers will be considered. In one method, the model propeller thrust is held constant as the angle of attack is varied, the process being repeated for different amounts of thrust. The expression "constant thrust" will be associated with this test procedure. In the other method, the thrust is varied with lift coefficient in a predetermined manner such as to simulate the thrust condition that exists on the full-scale airplane when flown at a fixed manifold pressure (constant throttle setting). (For constant-speed propeller operation the propeller speed is also constant.) This type of testing will be referred to as the "constant-power" procedure. The constant-thrust procedure will be considered first.

### Constant-Thrust Procedure

For the purposes of analysis, an airplane of the single-engine tractor type with conventional tail arrangement will be considered. If the entire tail surface is assumed to be subjected to slipstream action, the resultant pitching-moment coefficient may be written as follows:

$$C_M = C_{m_0} + (C_g - C_a) C_L - \left(\frac{z}{c}\right) C_{X_w} - R C_{L_t} \frac{S_t}{S_w} \frac{l_1}{c} \\ + \frac{l_2}{c} \frac{8}{\pi} \frac{S_p}{S_w} \frac{K}{J^2} \alpha + \frac{h}{c} T_c' \quad (11)$$

The factors  $C_{m_0}$ ,  $C_L$ , and  $C_{X_w}$  exclude direct thrust effects but include interference effects due to the slipstream. (See, for example, equation (9a).) When only part of the tail is included in the slipstream, the fourth term on the right side of equation (11) will be lower but will depend on the identical parameters. The fifth term represents the contribution of the aerodynamic side force developed by the inclined propeller (reference 2). In this analysis, this contribution will be grouped with the aerodynamic terms rather than with the direct contribution of the thrust.

If

$$R = 1 + T_c' \frac{S_w}{S_p}$$

and it is assumed

$$\frac{dR}{dC_L} = \frac{dR}{dT_c'} \frac{dT_c'}{dC_L}$$

differentiation with respect to the over-all lift coefficient yields

$$\begin{aligned} \frac{dC_M}{dC_L} = (C_g - C_a) - \frac{z}{c} \frac{dC_{X_W}}{dC_L} - R \frac{l_1}{c} \frac{S_t}{S_w} \frac{dC_{L_t}}{dC_L} \\ + \frac{l_2}{c} \frac{8}{\pi} \frac{S_p}{S_w} \frac{K}{J^2} \frac{d\alpha}{dC_L} + \frac{h}{c} \frac{dT_{c'}}{dC_L} - C_{L_t} \frac{l_1}{c} \frac{S_t}{S_p} \frac{dT_{c'}}{dC_L} \quad (12) \end{aligned}$$

Equation (12) represents the general variation of the resultant pitching-moment coefficient with lift coefficient as it includes terms involving the variation of propeller thrust. If the constant-thrust test procedure is employed ( $T_{c'} = \text{constant}$ ), equation (12) reduces to

$$\begin{aligned} \left( \frac{dC_M}{dC_L} \right)_{T_{c'}} = \left( \frac{\partial C_M}{\partial C_L} \right) = (C_g - C_a) - \frac{z}{c} \frac{dC_{X_W}}{dC_L} \\ - R \frac{l_1}{c} \frac{S_t}{S_w} \frac{dC_{L_t}}{dC_L} + \frac{l_2}{c} \frac{8}{\pi} \frac{S_p}{S_w} \frac{K}{J^2} \frac{d\alpha}{dC_L} = \frac{\partial C_M}{\partial C_L} \quad (12a) \end{aligned}$$

The term  $dC_{L_t}/dC_L$  is assumed to be independent of  $T_{c'}$  because

$$\frac{dC_{L_t}}{dC_L} = \frac{dC_{L_t}}{d\alpha_t} \left( \frac{d\alpha}{dC_L} - \frac{d\epsilon_w}{dC_L} - \frac{d\epsilon_p}{dC_L} \right)$$

and experiments indicate that  $d\epsilon_p/dC_L$  is essentially constant, at least for the flap-up condition. It is seen that the value of  $\left( \frac{dC_M}{dC_L} \right)_{T_{c'}}$  varies with the magnitude of

$T_{c'}$  but that it is essentially independent of the tail load and hence of the value of the pitching moment at which the slope is measured. Accordingly, it makes little difference whether the model is trimmed or not and the slope

$\left( \frac{dC_M}{dC_L} \right)_{T_{c'}}$  may be evaluated with any tail setting, although

it should be appreciated that the slope of the pitching-moment curve obtained from the constant-thrust procedure is, by itself, meaningless from the consideration of stability in steady flight.

In order to estimate the stability parameter

$\left(\frac{dC_M}{dC_L}\right)_{C_M=0}$  from wind-tunnel data obtained by the constant-

thrust procedure, it is now necessary to evaluate the terms

$$\frac{1}{2C_L} \left[ v \left( \frac{\partial C_M}{\partial v} \right) + v \frac{dT_c'}{dv} \frac{h}{c} \right] = - C_{Lt} \frac{l_1}{c} \frac{S_t}{S_p} \frac{dT_c'}{dC_L} + \frac{h}{c} \frac{dT_c'}{dC_L}$$

where

$$C_{Lt} = - \frac{(C_{Mnt})_{T_c'}}{\left(1 + T_c' \frac{S_w}{S_p}\right) \frac{S_t}{S_w} \frac{l_1}{c}}$$

Thus

$$\begin{aligned} \left(\frac{dC_M}{dC_L}\right)_{C_M=0} &= \left(\frac{\partial C_M}{\partial C_L}\right)_{T_c'} + \frac{(C_{Mnt})_{T_c'}}{\left(1 + T_c' \frac{S_w}{S_p}\right) \frac{S_t}{S_w} \frac{l_1}{c}} \frac{l_1}{c} \frac{S_t}{S_p} \frac{dT_c'}{dC_L} + \frac{h}{c} \frac{dT_c'}{dC_L} \\ &= \left(\frac{\partial C_M}{\partial C_L}\right)_{T_c'} + \frac{dT_c'}{dC_L} \left[ \frac{(C_{Mnt})_{T_c'}}{\left(1 + T_c' \frac{S_w}{S_p}\right) \frac{S_t}{S_w} \frac{l_1}{c}} \frac{l_1}{c} \frac{S_t}{S_p} + \left(\frac{h}{c}\right) \right] \quad (13) \end{aligned}$$

The foregoing relationships can be used to estimate  $\left(\frac{dC_M}{dC_L}\right)_{C_M=0}$  when tunnel data are available in the form  $\left(\frac{dC_M}{dC_L}\right)_{T_c'}$  for

both tail-on and tail-off conditions and when propeller data are available to evaluate the quantity,  $dT_c'/dC_L$ .

From equation (6b)

$$\frac{dT_c'}{dC_L} = - \frac{V}{2C_L} \left( \frac{dT_c'}{dV} \right)$$

The slope  $dT_c'/dV$  is best determined graphically for the particular condition under consideration. A method for finding the variation of propeller thrust with forward velocity for either fixed-pitch or controllable-pitch constant-speed propellers is outlined in reference 3. Figure 2 typifies the variation of thrust coefficient and blade angle with velocity for constant-speed propeller operation.

#### Constant-Power Procedure

If in equation (12) the thrust is varied in accordance with the relation  $\frac{dT_c'}{dC_L} = - \frac{V}{2C_L} \left( \frac{dT_c'}{dV} \right)$ , equation (12) is

representative of the constant-power test procedure. The direct relationship between the resultant slope of the pitching-moment curve for the constant-power test procedure and the constant term  $E$  of the stability biquadratic for power-on flight has already been established.

It is obvious from equation (12) that, in the constant-power test procedure, the tail setting directly affects the measured wind-tunnel pitching-moment slope,  $dC_M/dC_L$ . In evaluating  $dC_M/dC_L$  it is consequently important to use that tail setting for which the model is trimmed (that is,  $C_M = 0$ ). It will be observed from equation (12) that, in power-on flight, the slope of the resultant pitching-moment curve (and consequently the stability characteristics) is affected by both the center-of-gravity location and the associated tail load necessary to produce trim. The manner in which these associated variables affect the stability characteristics forms the subject matter for the remainder of this paper.

# EFFECT OF TAIL LOAD AND CENTER-OF-GRAVITY POSITION ON THE SLOPE OF THE PITCHING-MOMENT CURVE

In general, the contribution of the tail load to the resultant pitching-moment slope,  $dC_M/dC_L$ , is expressed by final term in equation (12). Thus

$$\Delta \left( \frac{dC_M}{dC_L} \right)_{\text{tail}} = - C_{L_t} \frac{l_1}{c} \frac{S_t}{S_p} \frac{dT_c'}{dC_L}$$

where

$$C_{L_t} = \left( \frac{dC_{L_t}}{d\alpha_t} \right) \left[ \alpha - \epsilon_w - \epsilon_p + \left( \frac{\partial \alpha_t}{\partial \delta_e} \right) \delta_e + i_t \right]$$

The effect of this term on  $dC_M/dC_L$  is demonstrated in figure 3. The theoretical curve was evaluated by use of the propeller-operating characteristics that were used in securing the experimental results. The experimental points were obtained from data of unpublished tests. Both the theoretical and the experimental results indicate that increased down loads on the tail result in more positive values of  $dC_M/dC_L$  and thus are detrimental to stability.

In accordance with equation (11), the downward tail load must be increased to preserve the trim condition when the center of gravity of the airplane is moved forward. In power-off flight, a forward movement of the center of gravity is normally stabilizing, as it results in more negative values of  $dC_M/dC_L$ . It has just been shown, however, that an increasing down load on the tail is detrimental to stability in power-on flight. Thus, the two effects oppose one another. The results of a theoretical examination of these antithetical effects of a center-of-gravity movement are presented in figures 4 and 5. In the computations for figure 4, the elevator angles necessary to trim the airplane with the various center-of-gravity positions were computed and the associated values of  $dC_M/dC_L$

and  $\left( \frac{\partial C_M}{\partial C_L} \right)_{T_c'}$  were calculated. The value of  $dC_M/dC_L$  is

the slope of the pitching-moment curve associated with constant-power tests:  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$  is the slope of the pitching-moment curve associated with constant-thrust tests. It will be noted that the variations of both  $dC_M/dC_L$  and  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$  with forward center-of-gravity movement indicate a net increase in stability, but that the increased negative values of  $dC_M/dC_L$  are less than the increased negative values of  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$ . A comparison of these two quantities reveals directly the effect of the increased down load on the tail required for trim with the forward center-of-gravity position, for  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$  includes the effect of the shift in center-of-gravity location but not the change in tail load, whereas  $dC_M/dC_L$  includes both of these variations.

Computations that show the variations of  $dC_M/dC_L$ ,  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$ , and the elevator angle for trim with lift coefficient for two center-of-gravity positions are presented in figure 5. The variation of the thrust coefficient,  $T_c$ , with lift coefficient is also shown. The thrust coefficients were used in evaluating  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$ . The more negative values of  $dC_M/dC_L$  and the steeper slope to the elevator angle for trim curve are associated with the most forward center-of-gravity location. It will be noted that for values of  $C_L$  greater than 1, the slope  $dC_M/dC_L$  for the 26-percent mean aerodynamic chord center-of-gravity position is greater than the associated slope  $\left(\frac{\partial C_M}{\partial C_L}\right)_{T_c}$ . This behavior results from the increased positive (upward) tail loads required for trim at the higher lift coefficients. Thus, it is seen that the effect of power on the contribution of the tail to the stability characteristics is adverse only when the tail is carrying an initial down load.

## CONCLUSIONS

On the basis of this analysis, the following conclusions may be reached:

1. For small angles of climb, the slope of the curve of elevator angle for trim against lift coefficient secured from flight tests is directly proportional to the slope of the curve of pitching-moment coefficient against lift coefficient secured from wind-tunnel tests simulating flight with constant power only when the model is trimmed for zero pitching-moment.

2. The destabilizing effects of power are more pronounced when the horizontal tail is required to carry a down load to maintain flight equilibrium.

3. A longitudinal movement of the center of gravity affects the longitudinal stability characteristics less in power-on flight than in power-off flight.

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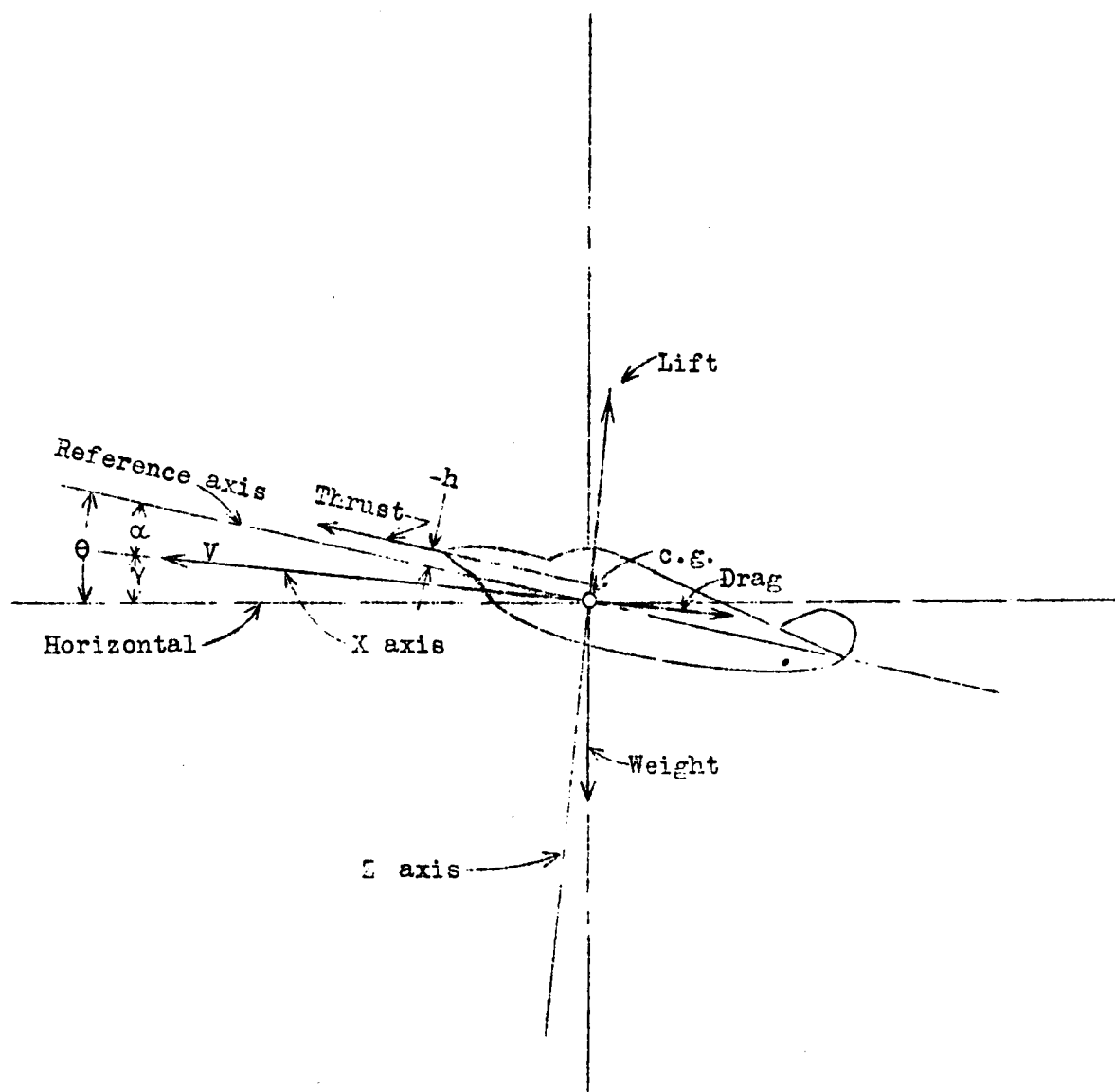


Figure 1.- Angular and vectorial relationships in power-on flight. Flight-path axis.

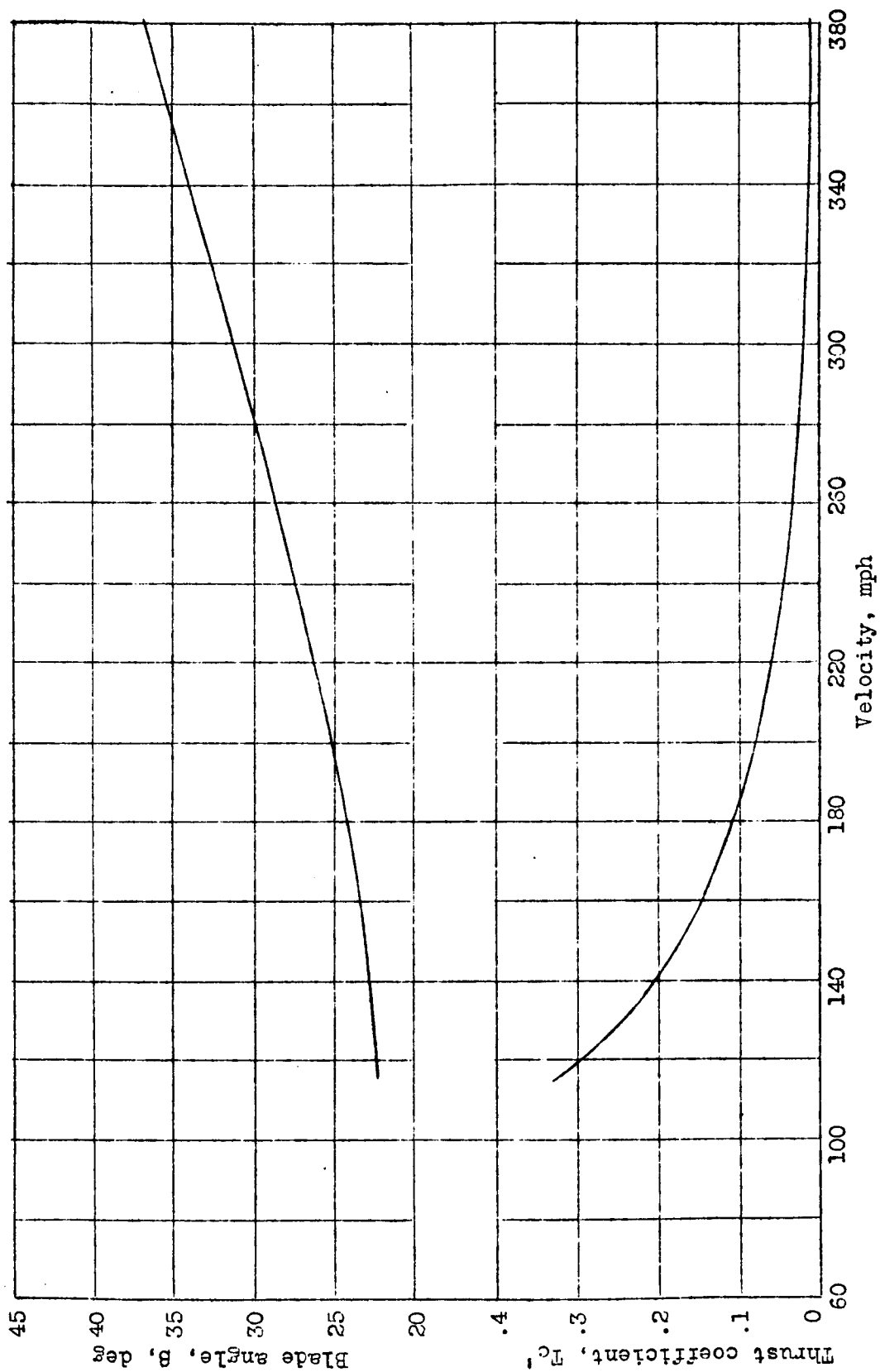


Figure 2.- Typical variation of thrust coefficient and blade angle with velocity for constant-speed propeller.

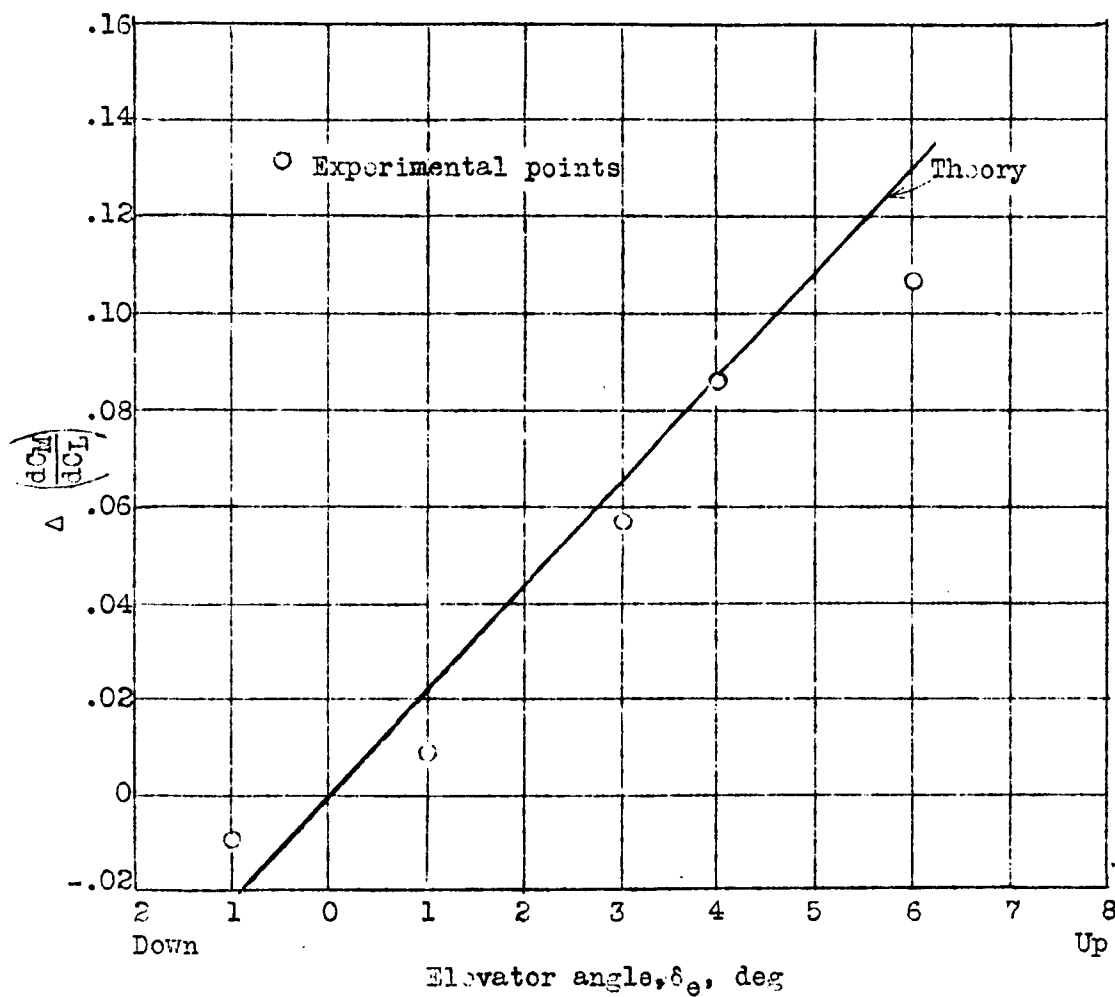


Figure 3.- Effect of tail load on slope of resultant pitching-moment curve.  
 $T_c'$ , 0.195,  $C_L$ , 0.6.

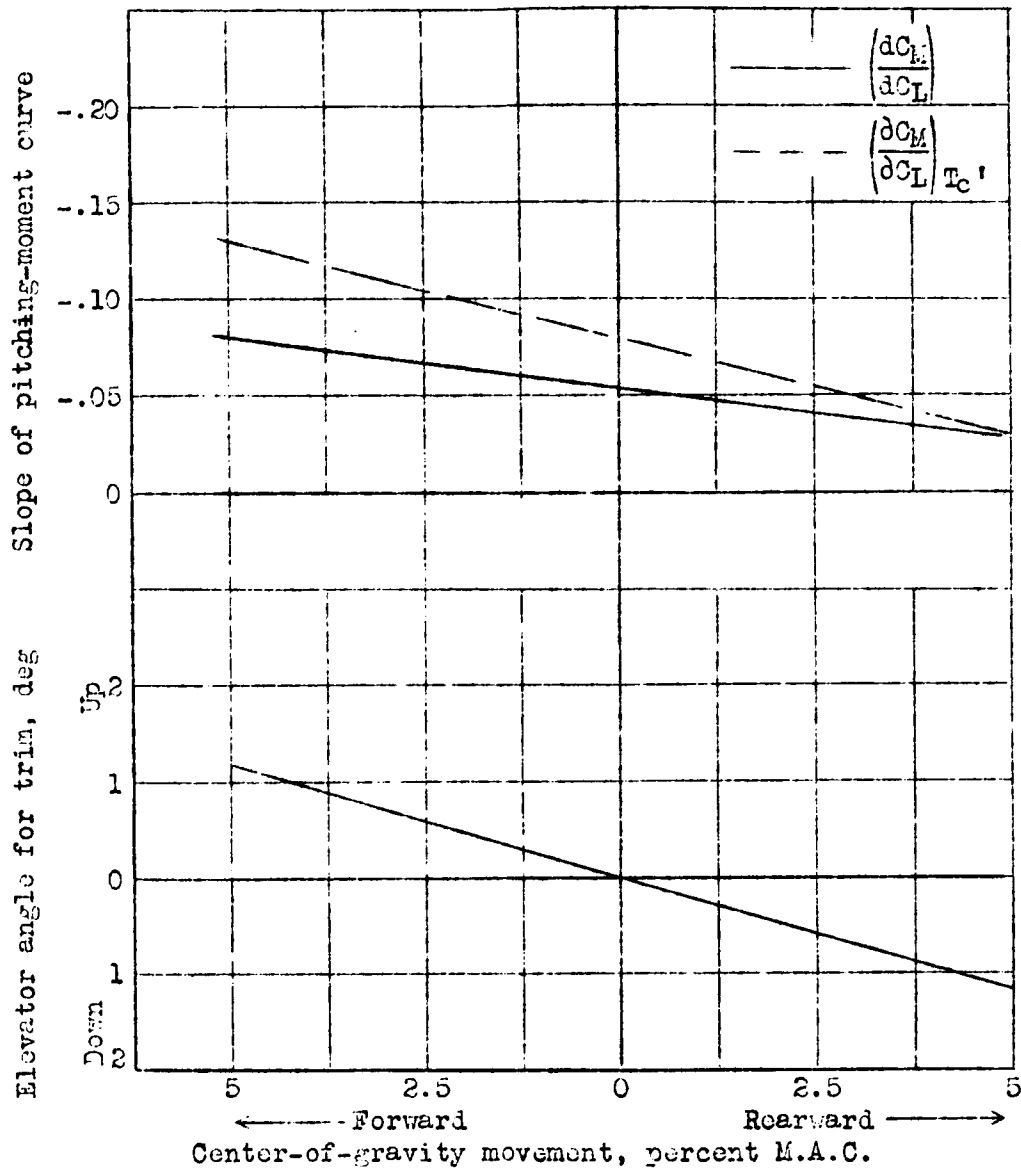


Figure 4.- Effect of center-of-gravity movement on slope of pitching-moment curve and on elevator angle for trim.  $T_C'$ , 0.195,  $C_L$ , 0.5.

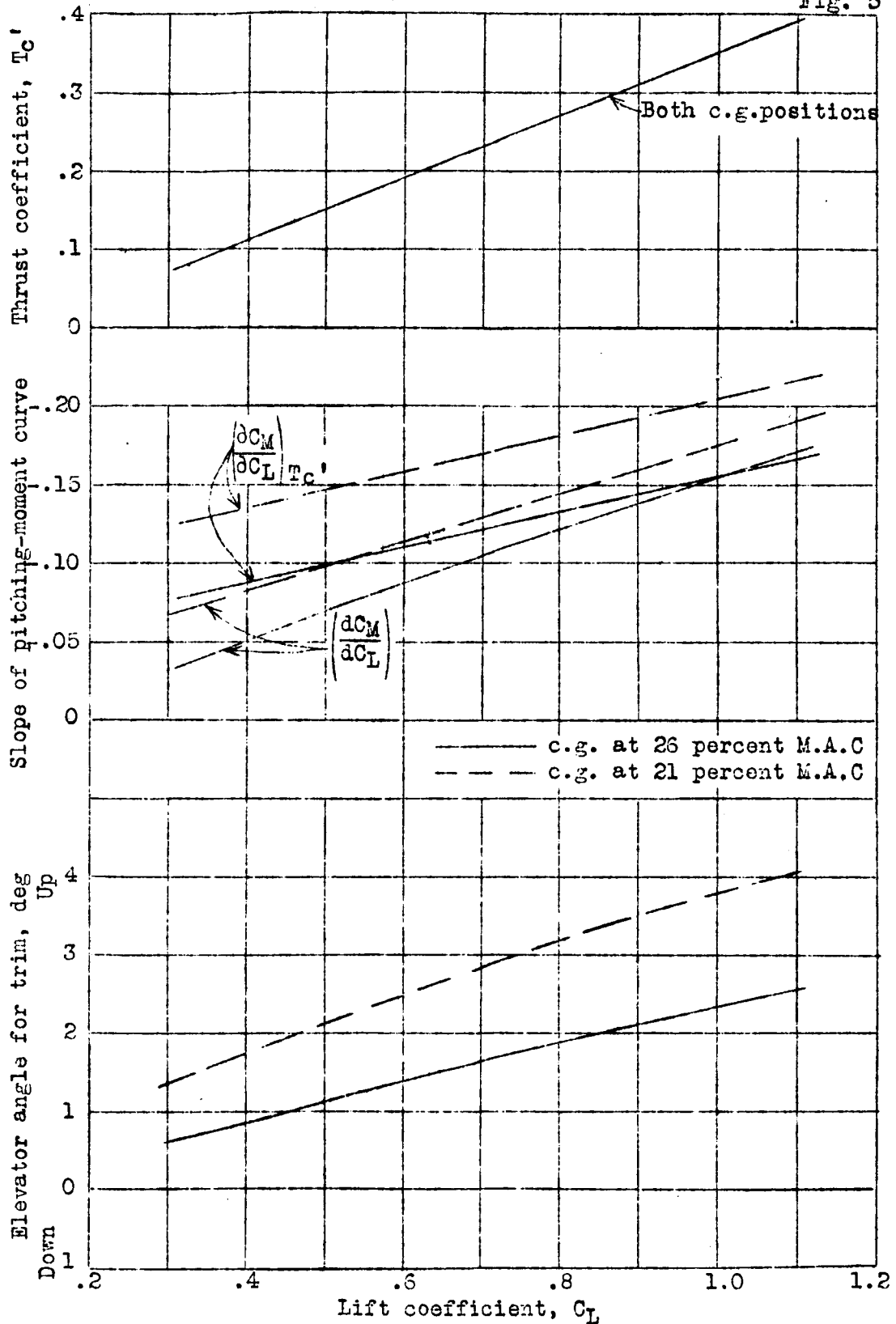


Figure 5.- Variation of elevator angle for trim, slope of pitching-moment curve, and thrust coefficient with lift coefficient.

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MEMORANDUM REPORT

for the

Army Air Forces, Materiel Command

LATERAL STABILITY AND CONTROL TESTS OF THE  
XP-77 AIRPLANE IN THE NACA FULL-SCALE TUNNEL

By K. R. Czarnecki and C. J. Donlan

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# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## MEMORANDUM REPORT

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Army Air Forces, Materiel Command

### LATERAL STABILITY AND CONTROL TESTS OF THE XP-77 AIRPLANE IN THE NACA FULL-SCALE TUNNEL

By K. R. Czarnecki and C. J. Donlan

#### INTRODUCTION

At the request of the Army Air Forces, Materiel Command, tests have been made in the NACA full-scale tunnel to determine the lateral stability and control characteristics of the XP-77 airplane. Measurements were made of the forces and moments on the airplane at various angles of attack and angles of yaw. The measurements were made with the propeller removed and with the propeller installed and operating at various thrust coefficients, and with the landing flaps retracted and deflected. The effects of aileron, elevator, and rudder deflection on control-surface effectiveness and hinge moments were determined. The tests were planned to obtain the data required to evaluate as completely as possible the Army Air Force requirements on lateral stability and control for pursuit-type airplanes. In this report are presented the results of the tunnel tests while in reference 1 are reported the estimated flying qualities of the XP-77 airplane as evaluated from these data.

#### DESCRIPTION OF TEST AIRPLANE

The XP-77 is a single-engine, low-wing pursuit airplane constructed almost entirely of wood and of magnesium alloy. Two versions of the airplane are under construction, each of which is equipped with a Ranger SGV-770 inverted-vee air-cooled engine. The first version of the airplane is a low-altitude fighter with a military rating of 515 horsepower and a normal rating of 450 horsepower at a critical altitude of 12,000 feet. The second version

of the airplane is a high-altitude fighter with approximately the same power ratings but with a critical altitude of 27,000 feet.

The airplane used in the full-scale tunnel tests (three-view in fig. 1) differed from either the low- or high-altitude versions of the XP-77 in that:

1. The test airplane had the thrust line location associated with the low-altitude version of the XP-77 airplane.

2. The propeller available for the tests was the  $10\frac{1}{2}$ -foot propeller associated with the high-altitude version of the XP-77 airplane.

3. The landing gear was removed for all tests.

These circumstances precluded the simultaneous duplication of the thrust and slipstream characteristics associated with the specific flight operation of either the low- or high-altitude version of the airplane. The quantitative results included in this report therefore cannot be associated specifically with either the low- or high-altitude version of the XP-77 airplane but may be taken as characteristic of the airplane in general.

Prior to the tests, some modifications were also made to the ailerons and landing flaps, both of which were of the internally balanced sealed-hinge type. Vertical end seals were provided at the edges of the fabric adjacent to the aileron and flap hinges in order to prevent air leakage from reducing the pressure differential across the seals, which, in the original condition, did not extend across the hinges. In addition, extremely large and uneven gaps and discontinuities in contour existed between the wing and the aileron and flap surfaces. To remedy this condition, the skin on both the top and bottom of the wing aft of the rear beam was faired smoothly into the contour of the ailerons and flaps, and the gaps between the wing and the movable surfaces were decreased (fig. 2).

The airplane was equipped with the modified NACA cowling described in the recommendations of reference 2. Throughout the tests the engine- and oil-cooler cooling-air exit doors were locked in the full-open position.



## SYMBOLS

The so-called "stability axes," which are used in presenting the force and moment coefficients, form an orthogonal coordinate system with the origin at the center of gravity of the airplane. The X axis always lies in the plane of symmetry of the airplane and is oriented such that it is coincident with the relative wind axis at zero yaw and with the projection of the relative wind axis on the plane of symmetry at other angles of yaw.

The moments are computed about a center of gravity located at 21.66 percent of the mean aerodynamic chord and 1.80 percent of the mean aerodynamic chord below the airplane thrust axis.

$C_{DR}$  resultant drag coefficient  $\left(\frac{D}{qS}\right)$

$C_Y, C_y$  lateral-force coefficient  $\left(\frac{Y}{qS}\right)$

$C_L$  lift coefficient  $\left(\frac{Z}{qS}\right)$

$C_l$  rolling-moment coefficient  $\left(\frac{L}{qSb}\right)$

$C_m$  pitching-moment coefficient  $\left(\frac{M}{qSc}\right)$

$C_n$  yawing-moment coefficient  $\left(\frac{N}{qSb}\right)$

$C_{h_a}$  aileron hinge-moment coefficient  $\left(\frac{H_a}{q\bar{c}_a^2 b_a}\right)$

$C_{h_e}$  elevator hinge-moment coefficient  $\left(\frac{H_e}{q\bar{c}_e^2 b_e}\right)$

$C_{h_r}$  rudder hinge-moment coefficient  $\left(\frac{H_r}{q\bar{c}_r^2 b_r}\right)$

|                                              |                                                                              |
|----------------------------------------------|------------------------------------------------------------------------------|
| $T_c$                                        | effective thrust coefficient $\left(\frac{T}{\rho V^2 D^2}\right)$           |
| $V/nD$                                       | propeller advance-diameter ratio                                             |
| $\alpha$                                     | angle of attack of thrust axis relative to free-stream direction             |
| $\beta$                                      | propeller blade setting at 0.75 radius                                       |
| $\psi$                                       | angle of yaw, positive when the left wing is forward                         |
| $\delta_a$                                   | aileron deflection, positive when trailing edge of aileron is down           |
| $\delta_e$                                   | elevator deflection, positive when trailing edge of elevator is down         |
| $\delta_r$                                   | rudder deflection, positive when trailing edge of rudder is to left          |
| $\delta_f$                                   | flap deflection                                                              |
| $V_i$                                        | correct indicated airspeed                                                   |
| $\frac{\partial C_l}{\partial \delta_a}$     | rate of change of rolling-moment coefficient with aileron deflection         |
| $\frac{\partial C_{h_a}}{\partial \delta_a}$ | rate of change of aileron hinge-moment coefficient with aileron deflection   |
| $\frac{\partial C_m}{\partial \delta_e}$     | rate of change of pitching-moment coefficient with elevator deflection       |
| $\frac{\partial C_{h_e}}{\partial \delta_e}$ | rate of change of elevator hinge-moment coefficient with elevator deflection |
| $\frac{\partial C_n}{\partial \delta_r}$     | rate of change of yawing-moment coefficient with rudder deflection           |
| $\frac{\partial C_y}{\partial \delta_r}$     | rate of change of lateral-force coefficient with rudder deflection           |

$\frac{\delta C_{h_r}}{\delta \delta_r}$  rate of change of rudder hinge-moment coefficient  
with rudder deflection

where

- D force in direction of relative wind
- X force along X axis, positive when directed  
backward
- Y force along Y axis, positive when directed to  
right
- Z force along Z axis, positive when directed  
upward
- L rolling moment about X axis, positive when it  
tends to depress the right wing
- M pitching moment about Y axis, positive when it  
tends to depress the tail
- N yawing moment about Z axis, positive when it  
tends to retard the right wing
- H<sub>a</sub> aileron hinge moment, positive when it tends to  
depress the trailing edge
- H<sub>e</sub> elevator hinge moment, positive when it tends to  
depress the trailing edge
- H<sub>r</sub> rudder hinge moment, positive when it tends to  
move trailing edge to left
- q dynamic pressure  $\left(\frac{1}{2}\rho v^2\right)$
- S wing area (100 square feet)
- b wing span (27.5 feet)
- b<sub>a</sub> aileron span (5.64 feet)
- b<sub>e</sub> elevator span (8.93 feet)
- b<sub>r</sub> rudder span (4.15 feet)
- c mean aerodynamic chord (3.989 feet)

|             |                                              |
|-------------|----------------------------------------------|
| $\bar{c}_a$ | root-mean-square aileron chord (0.477 foot)  |
| $\bar{c}_e$ | root-mean-square elevator chord (0.807 foot) |
| $\bar{c}_r$ | root-mean-square rudder chord (1.175 feet)   |
| T           | effective thrust                             |
| $\rho$      | mass density of air                          |
| V           | free-stream velocity                         |
| D           | propeller diameter                           |
| n           | propeller rotational speed                   |

#### METHODS AND TESTS

The NACA full-scale tunnel and balance equipment used for the tests are described in reference 3. Figures 3 and 4 show the XP-77 airplane mounted on the balance in unyawed and yawed conditions, respectively. All data presented in the report have been corrected for jet boundary and blocking effects by the methods of references 4 and 5.

Measurements were made of the effect of rudder deflection on the aerodynamic characteristics and on the rudder hinge-moments of the airplane for various flight attitudes, with the flaps retracted and with the flaps deflected  $20^\circ$  and  $55^\circ$ . The flight attitudes associated with specific combinations of angle of attack and thrust coefficient are given in table I. These tests were made at the following angles of yaw:  $\psi = -14.6^\circ, -9.8^\circ, -5.3^\circ, -3.0^\circ, 0^\circ, 3.7^\circ, 6.2^\circ, 10.0^\circ$ , and  $14.7^\circ$ . For all tests the rudder tab was deflected a small negative amount (trailing edge to right). No tab-effectiveness tests are included in this report.

Elevator effectiveness and hinge moments were determined for the flight attitudes listed in table II. These tests were made at angles of yaw of  $-9.8^\circ, 0^\circ$ , and  $10.0^\circ$ .

Both the rudder and elevator tests were made with the propeller operating. The propeller blade angle was

set at  $22^\circ$  at the 0.75 radius, and the propeller speeds, for the power conditions to be simulated, were determined from a propeller-calibration test made with the airplane at zero yaw in an attitude near zero lift. The results of the propeller-calibration test ( $T_c$  against  $V/nD$ ) and the estimated variation of thrust coefficient with lift coefficient for various engine powers are shown in figure 5. For the simulated level-flight attitudes the propeller rpm was adjusted so that the thrust would be equal to the airplane drag at zero yaw. At angles of yaw other than zero, the airplane was set for the specific flight attitude and the propeller rpm manipulated to yield the same advance-diameter ratio  $V/nD$  that existed for the same attitude at zero yaw.

Because the ailerons are well out of the region covered by the propeller slipstream, all of the aileron-effectiveness and hinge-moment tests were made with the propeller removed. These tests were made with the airplane at zero yaw for deflections of only the left aileron. The angles of attack and flap settings used during the tests are given in table III.

All hinge moments were measured by means of cantilever springs equipped with electrical strain gages to indicate the beam deflections under load.

The tests simulating take-off power and the propeller-calibration tests at high thrust coefficients were made at a tunnel airspeed of approximately 62 miles per hour. All other tests were made at a tunnel airspeed of approximately 85 miles per hour, corresponding to a Reynolds number of 3,400,000.

## RESULTS AND DISCUSSION

### Rudder Effectiveness and Hinge Moments

The effects of rudder deflection on the aerodynamic characteristics of the XP-77 airplane are shown in figures 6 to 9 for the various conditions listed in table I with the airplane at zero yaw. The measured rudder-effectiveness and hinge-moment parameters are also given in table I. For the attitudes with flap retracted the parameters  $\frac{\partial C_n}{\partial \delta_r}$ ,  $\frac{\partial C_y}{\partial \delta_r}$ , and  $\frac{\partial C_{h_r}}{\partial \delta_r}$

averaged about -0.0012, 0.0025, and -0.0053 per degree, respectively. With the flaps deflected  $55^\circ$  respective values of  $\frac{\partial C_n}{\partial \delta_r}$ ,  $\frac{\partial C_y}{\partial \delta_r}$ , and  $\frac{\partial C_{hr}}{\partial \delta_r}$  of -0.0015, 0.0032, and -0.0053 were obtained for the airplane in the approach attitude ( $\alpha = 4.4^\circ$ ) with power for level flight ( $T_c = 0.089$ ). Comparison of the runs made with the airplane in the intermediate and low-speed attitudes with propeller idling indicates some loss in rudder effectiveness with increase in angle of attack when flaps are deflected for landing.

Results of tests made to determine the effect of rudder deflection on the aerodynamic characteristics of the airplane in yaw for some of the conditions listed in table I are given in figures 10 to 12 for  $\psi = -9.8^\circ$  and in figures 13 to 15 for  $\psi = 10.0^\circ$ . The variations of  $\frac{\partial C_n}{\partial \delta_r}$  and  $\frac{\partial C_{hr}}{\partial \delta_r}$  with angles of yaw for these conditions are presented in figures 16 to 18. In general, for the range of yaw angles tested, both  $\frac{\partial C_{hr}}{\partial \delta_r}$  and  $\frac{\partial C_n}{\partial \delta_r}$  did not change appreciably with angle of yaw.

The variation of rudder hinge-moment coefficient at zero rudder deflection with angle of yaw is plotted in figure 19. Some irregularities in the variation of  $(C_{hr})_{\delta_r=0}$  with  $\psi$  are evident at low angles of yaw. These irregularities are generally small and may be within the experimental accuracy of the tests.

The rudder angles necessary to hold a given angle of yaw are given in figures 20 to 22 for some of the attitudes listed in table I. Attention is called to the irregularities in the slope of the curves in the vicinity of zero yaw for most of the conditions tested.

#### Elevator Effectiveness and Hinge Moments

The variation of  $C_m$ ,  $C_L$ , and  $C_{he}$  with  $\delta_e$  is shown in figures 23 to 25 for the attitudes listed in table II. The values for the elevator parameters,

$\frac{\partial C_m}{\partial \delta_e}$  and  $\frac{\partial C_{h_e}}{\partial \delta_e}$ , for these attitudes are also listed in table II. No tests were made to determine the elevator tab effectiveness.

At zero yaw, with the airplane in the low-speed level-flight condition ( $\alpha = 8.8^\circ$  and  $T_c = 0.049$ ) with flaps retracted, a value of  $\frac{\partial C_m}{\partial \delta_e}$  of -0.0208 per degree was obtained. With flaps deflected  $20^\circ$ , and with the airplane in the intermediate-speed attitude ( $\alpha = 2.2^\circ$ ) with level-flight power ( $T_c = 0.040$ ), the value of  $\frac{\partial C_m}{\partial \delta_e}$  was decreased to -0.0189 per degree. In the approach attitude with flaps deflected  $55^\circ$ , the value of elevator effectiveness was -0.0215 per degree. The effect of yaw was to decrease  $\frac{\partial C_m}{\partial \delta_e}$  slightly for all the conditions tested.

No values of  $\frac{\partial C_{h_e}}{\partial \delta_e}$  near zero elevator hinge-moment coefficient are available for  $\psi = 0^\circ$ , but values ranging from -0.0060 to -0.0088 were measured at yaw angles of  $-9.8^\circ$  and  $10.0^\circ$ . At zero yaw (fig. 23) there was a pronounced increase in elevator hinge moments for elevator deflections greater than  $-16^\circ$  accompanied by decreases in the slope  $\frac{\partial C_m}{\partial \delta_e}$ .

#### Aileron Effectiveness and Hinge Moments

The variations of  $C_{h_a}$ ,  $C_n$ , and  $C_l$  with  $\delta_a$  are shown in figures 26 to 28 for the airplane with flaps retracted and with flaps deflected  $20^\circ$  and  $55^\circ$ , respectively. In table III are given the corresponding values of  $\frac{\partial C_l}{\partial \delta_a}$  and  $\frac{\partial C_{h_a}}{\partial \delta_a}$ . With the flaps retracted, the value of  $\frac{\partial C_l}{\partial \delta_a}$  decreased from 0.0016 per degree at  $\alpha = -1.2^\circ$

to 0.0013 per degree at  $\alpha = 11.7^\circ$ . Similar decreases occurred with the flaps deflected  $20^\circ$  and  $55^\circ$ . For a constant airplane angle of attack the effect of deflecting the landing flaps was to decrease the value of  $\frac{\partial C_L}{\partial \delta_a}$  slightly. The values of  $\frac{\partial C_{h_a}}{\partial \delta_a}$  increased with airplane angle of attack up to about  $6.5^\circ$  then decreased at the highest angle of attack of about  $11.5^\circ$ .

### Aerodynamic Characteristics in Yaw

The aerodynamic characteristics of the XP-77 airplane in yaw were determined by cross-plotting the results of the rudder-effectiveness and hinge-moment tests at several angles of yaw. The cross plots showing the variation of  $C_m$ ,  $C_L$ , and  $C_{D_R}$  with  $\psi$  for  $\delta_r = 0$  are given in figures 29 to 31 for the attitudes listed in table I. For the conditions with high thrust coefficients, increases in lift at high angles of positive yaw and decreases in lift at high angles of negative yaw were measured. These changes probably occur as a result of the effects on the tail lift of the propeller slipstream and the direction of yaw inasmuch as the increases in lift are accompanied by increases in negative pitching moment and vice versa.

The variations of  $C_n$ ,  $C_L$ , and  $C_y$ , with  $\psi$  for  $\delta_r = 0$  are presented in figures 32 to 34 for all the attitudes listed in table I. A sharp change in the slope  $\frac{\partial C_n}{\partial \psi}$  occurs around zero yaw in the flaps-retracted, intermediate-speed condition ( $\alpha = 4.8^\circ$ ) with level-flight power ( $T_c = 0.038$ ). This change in slope with yaw angle, although less pronounced, is also apparent for the other test conditions. These conditions are also reflected in the curves showing the variation of rudder angle for trim with angle of yaw (figs. 20 and 22). The reversal in the slope of the yawing-moment curve at high angles of positive yaw for the airplane in the take-off condition with take-off power (fig. 33(a)) may be due either to the stalling of the vertical tail or to the emergence of the tail from the propeller slipstream.



Cross plots of  $C_n$  against angle of yaw for  $C_{h_r} = 0$  are shown in figures 35 to 37. The slopes of these curves are lower than for the corresponding curves with  $\delta_r = 0$ .

The parameters  $\left(\frac{\partial C_n}{\partial \psi}\right)_{\delta_r=0}$ ,  $\left(\frac{\partial C_n}{\partial \psi}\right)_{C_{h_r}=0}$ ,  $\frac{\partial C_l}{\partial \psi}$  and  $\frac{\partial C_y}{\partial \psi}$  are given in table IV. With the rudder locked in neutral the values of  $\frac{\partial C_n}{\partial \psi}$  ranged from -0.0009 to -0.0016 except for the take-off condition with idling power when a value of -0.0006 was obtained. The values of  $\frac{\partial C_n}{\partial \psi}$  for  $C_{h_r} = 0$  were somewhat lower, ranging from -0.0004 to -0.0010. The slope of the curves of the lateral-force coefficient plotted as a function of angle of yaw varied between 0.0090 and 0.0176.

With the rudder locked in neutral the airplane exhibits an unstable dihedral effect  $\frac{\partial C_l}{\partial \psi}$  in the approach attitude with level-flight power and in the take-off attitude with take-off power. For the approach attitude the instability amounts to about  $3^\circ$  negative dihedral effect; for the take-off attitude the instability amounts to nearly  $5^\circ$  negative dihedral effect. For the intermediate-speed attitude with the flaps deflected  $20^\circ$  and with level-flight power the dihedral effect is only slightly positive. It appears likely that the use of full-rated power in this condition will also result in a negative dihedral effect.

## SUMMARY OF RESULTS

The following results were obtained from lateral stability and control tests made in the NACA full-scale tunnel on a full-scale model of the Bell XP-77 airplane.

1. For flight attitudes with flaps retracted the parameter of rudder effectiveness,  $\frac{\partial C_n}{\partial \delta_r}$ , averaged about -0.0012 per degree. The associated value of the hinge-

moment parameter  $\frac{\partial C_n}{\partial \delta_r}$  was -0.0053 per degree. The effect of power was to increase the magnitude of these parameters slightly; whereas flap deflection had little effect except at high angles of attack with the flaps deflected  $55^\circ$ , where some loss in rudder effectiveness was measured.

2. For a low-speed, level-flight attitude with flaps retracted  $\alpha = 8.8^\circ$  the parameter of elevator effectiveness  $\frac{\partial C_m}{\partial \delta_e}$  was about -0.0208 per degree. With the flap deflected this value was decreased slightly.

3. With the flaps retracted, the value of the parameter of aileron effectiveness  $\frac{\partial C_l}{\partial \delta_a}$  was about 0.0016 per degree at an angle of attack of  $-1.2^\circ$ . The value of  $\frac{\partial C_l}{\partial \delta_a}$  decreased slightly with increasing angles of attack and with flap deflection.

4. The parameter of weathercock stability  $\frac{\partial C_n}{\partial \psi}$  ranged from -0.0009 per degree to -0.0016 per degree except for the take-off condition with idling power where a value of -0.0006 was obtained. With the rudder freed, the parameter  $\frac{\partial C_n}{\partial \psi}$  ranged from -0.0004 per degree to -0.0010 per degree.

5. The airplane displayed a stable dihedral effect  $\frac{\partial C_l}{\partial \psi}$  in all attitudes tested except in the approach attitude with level-flight power and in the take-off attitude with take-off power. For these two conditions

the amount of instability displayed was equivalent to about  $3^{\circ}$  and  $5^{\circ}$  negative dihedral, respectively.

Langley Memorial Aeronautical Laboratory  
National Advisory Committee for Aeronautics  
Langley Field, Va., June 16, 1944

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TABLE I

RUDDER-EFFECTIVENESS AND HINGE-MOMENT PARAMETERS AT ZERO ANGLES OF YAW

| Attitude                  | $\sigma$<br>(deg) | $T_c$ | $\left(\frac{\partial C_n}{\partial \delta_r}\right)_{C_n=0}$ | $\left(\frac{\partial C_y}{\partial \delta_r}\right)_{C_y=0}$ | $\left(\frac{\partial C_{h_r}}{\partial \delta_r}\right)_{C_{h_r}=0}$ |
|---------------------------|-------------------|-------|---------------------------------------------------------------|---------------------------------------------------------------|-----------------------------------------------------------------------|
| $\delta_f = 0$            |                   |       |                                                               |                                                               |                                                                       |
| <u>Dive</u>               |                   |       |                                                               |                                                               |                                                                       |
| <u>Idling power</u>       | -2.4              | 0     | -0.0012                                                       | 0.0022                                                        | -0.0053                                                               |
| <u>High speed</u>         |                   |       |                                                               |                                                               |                                                                       |
| Level-flight power        | 1.2               | .021  | -.0012                                                        | .0025                                                         | -.0055                                                                |
| Intermediate speed        |                   |       |                                                               |                                                               |                                                                       |
| Level-flight power        | 4.8               | .038  | -.0012                                                        | .0028                                                         | -.0050                                                                |
| <u>Low speed</u>          |                   |       |                                                               |                                                               |                                                                       |
| Level-flight power        | 8.8               | .049  | -.0013                                                        | .0027                                                         | -.0048                                                                |
| $\delta_f = 20^\circ$     |                   |       |                                                               |                                                               |                                                                       |
| <u>Intermediate speed</u> |                   |       |                                                               |                                                               |                                                                       |
| Level-flight power        | 2.2               | 0.040 | -0.0013                                                       | 0.0027                                                        | -0.0062                                                               |
| Take-off                  |                   |       |                                                               |                                                               |                                                                       |
| Idling power              | 10.3              | 0     | -.0012                                                        | .0026                                                         | -.0048                                                                |
| Take-off power            | 10.1              | .250  | -.0018                                                        | .0036                                                         | -.0080                                                                |
| $\delta_f = 55^\circ$     |                   |       |                                                               |                                                               |                                                                       |
| <u>Intermediate speed</u> |                   |       |                                                               |                                                               |                                                                       |
| Idling power              | -0.1              | 0     | -0.0015                                                       | 0.0026                                                        | -0.0046                                                               |
| Approach                  |                   |       |                                                               |                                                               |                                                                       |
| Level-flight power        | 4.4               | .089  | -.0015                                                        | .0032                                                         | -.0053                                                                |
| <u>Low speed</u>          |                   |       |                                                               |                                                               |                                                                       |
| Idling power              | 8.2               | .005  | -.0012                                                        | .0024                                                         | -.0049                                                                |

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TABLE II

EFFECT OF ANGLE OF YAW ON  $\frac{\partial C_m}{\partial \delta_e}$  AND  $\frac{\partial C_{he}}{\partial \delta_e}$ 

| Attitude                  | $\alpha$<br>(deg) | $T_c$ | $\delta_f$<br>(deg) | $\left(\frac{\partial C_m}{\partial \delta_e}\right)_{C_m=0}$ |                  |                     | $\left(\frac{\partial C_{he}}{\partial \delta_e}\right)_{C_{he}=0}$ |                     |                     |
|---------------------------|-------------------|-------|---------------------|---------------------------------------------------------------|------------------|---------------------|---------------------------------------------------------------------|---------------------|---------------------|
|                           |                   |       |                     | $\psi = -9.8^\circ$                                           | $\psi = 0^\circ$ | $\psi = 10.0^\circ$ | $\psi = -9.8^\circ$                                                 | $\psi = 10.0^\circ$ | $\psi = 10.0^\circ$ |
| <u>Low speed</u>          |                   |       |                     |                                                               |                  |                     |                                                                     |                     |                     |
| Level-flight power        | 8.8               | 0.049 | 0                   | -0.0170                                                       | -0.0208          | -0.0198             | -0.0088                                                             | -0.0083             |                     |
| <u>Intermediate speed</u> |                   |       |                     |                                                               |                  |                     |                                                                     |                     |                     |
| Level-flight power        | 2.2               | .040  | 20                  | -.0182                                                        | -.0189           | -.0164              | -.0058                                                              | -.0048              |                     |
| <u>Approach</u>           |                   |       |                     |                                                               |                  |                     |                                                                     |                     |                     |
| Level-flight power        | 4.4               | .089  | 55                  | -.0183                                                        | -.0205           | -.0171              | -.0060                                                              | -.0072              |                     |

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TABLE III

SUMMARY OF  $\frac{\partial C_l}{\partial \delta_a}$  AND  $\frac{\partial C_{h_a}}{\partial \delta_a}$  AT ZERO AILERON DEFLECTION  
LEFT AILERON ONLY; PROPELLER REMOVED

| $\alpha$<br>(deg) | $\delta_f$<br>(deg) | $\left(\frac{\partial C_l}{\partial \delta_a}\right)_{\delta_a=0}$ | $\left(\frac{\partial C_{h_a}}{\partial \delta_a}\right)_{\delta_a=0}$ |
|-------------------|---------------------|--------------------------------------------------------------------|------------------------------------------------------------------------|
| -1.2              | 0                   | 0.0016                                                             | -0.0044                                                                |
| 2.1               | 0                   | .0014                                                              | -.0050                                                                 |
| 6.7               | 0                   | .0013                                                              | -.0065                                                                 |
| 11.7              | 0                   | .0013                                                              | -.0045                                                                 |
| 2.0               | 20                  | .0016                                                              | -.0043                                                                 |
| 6.5               | 20                  | .0012                                                              | -.0070                                                                 |
| 11.5              | 20                  | .0011                                                              | -.0065                                                                 |
| -1.5              | 55                  | .0013                                                              | -.0068                                                                 |
| 6.5               | 55                  | .0012                                                              | -.0072                                                                 |
| 11.5              | 55                  | .0011                                                              | -.0035                                                                 |

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TABLE IV

SUMMARY OF  $\left(\frac{\partial C_m}{\partial \psi}\right)_{\delta_f=0}$ ,  $\left(\frac{\partial C_m}{\partial \psi}\right)_{C_{h_f}=0}$ ,  $\left(\frac{\partial C_l}{\partial \psi}\right)$ 

| Attitude                        | $\alpha$ | $T_c$ | $\left(\frac{\partial C_m}{\partial \psi}\right)_{\delta_f=0}$ | $\left(\frac{\partial C_m}{\partial \psi}\right)_{C_{h_f}=0}$ | $\left(\frac{\partial C_l}{\partial \psi}\right)_{\delta_f=0}$ | $\left(\frac{\partial C_y}{\partial \psi}\right)_{\delta_f=0}$ |
|---------------------------------|----------|-------|----------------------------------------------------------------|---------------------------------------------------------------|----------------------------------------------------------------|----------------------------------------------------------------|
| Condition $\delta_f = 0$        |          |       |                                                                |                                                               |                                                                |                                                                |
| <u>Dive</u>                     |          |       |                                                                |                                                               |                                                                |                                                                |
| Idling power                    | -2.4     | 0     | -0.0009                                                        | -0.0006                                                       | 0.0011                                                         | 0.0113                                                         |
| <u>High speed</u>               |          |       |                                                                |                                                               |                                                                |                                                                |
| Level-flight power              | -1.2     | .021  | -.0010                                                         | -.0007                                                        | .0011                                                          | .0013                                                          |
| <u>Intermediate speed</u>       |          |       |                                                                |                                                               |                                                                |                                                                |
| Level-flight power              | 4.8      | .038  | -.0008<br>-.0015                                               | -.0010                                                        | .0005                                                          | .0115                                                          |
| <u>Low speed</u>                |          |       |                                                                |                                                               |                                                                |                                                                |
| Level-flight power              | 8.8      | .049  | -.0009                                                         | -.0006                                                        | .0006                                                          | .0090                                                          |
| Condition $\delta_f = 20^\circ$ |          |       |                                                                |                                                               |                                                                |                                                                |
| <u>Intermediate speed</u>       |          |       |                                                                |                                                               |                                                                |                                                                |
| Level-flight power              | 2.2      | 0.040 | -0.0015                                                        | -0.0008                                                       | 0.0001                                                         | 0.0135                                                         |
| <u>Take-off</u>                 |          |       |                                                                |                                                               |                                                                |                                                                |
| Idling power                    | 10.3     | 0     | -.0006                                                         | -.0006                                                        | .0005                                                          | .0110                                                          |
| Take-off power                  | 10.1     | .258  | -.0015                                                         | -.0007                                                        | <sup>a</sup> -.0009                                            | .0176                                                          |
| Condition $\delta_f = 55^\circ$ |          |       |                                                                |                                                               |                                                                |                                                                |
| <u>Intermediate speed</u>       |          |       |                                                                |                                                               |                                                                |                                                                |
| Idling power                    | -0.1     | 0     | -0.0016                                                        | -0.0006                                                       | 0.0006                                                         | 0.0138                                                         |
| <u>Approach</u>                 |          |       |                                                                |                                                               |                                                                |                                                                |
| Level-flight power              | 4.4      | .089  | -.0016                                                         | -.0006                                                        | <sup>a</sup> -.0006                                            | .0160                                                          |
| <u>Low speed</u>                |          |       |                                                                |                                                               |                                                                |                                                                |
| Idling power                    | 8.2      | .005  | -.0012                                                         | -.0004                                                        | -----                                                          | .0129                                                          |

<sup>a</sup>Unstable dihedral effect.NATIONAL ADVISORY  
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## FIGURE LEGENDS

Figure 1.- Three-view drawing of the XP-77 airplane used in the full-scale tunnel.

Figure 2.- Modifications made to wing to provide smooth contours and smaller wing-aileron or wing-flap gaps.

Figure 3.- The XP-77 airplane mounted in the NACA full-scale tunnel.  $\psi, 0^\circ$ .

Figure 4.- The XP-77 airplane mounted in the NACA full-scale tunnel.  $\psi, 10^\circ$ .

Figure 5.- Variation of  $T_c$  with  $C_L$  and  $V/nD$  for constant power operation at sea level.  $\beta, 22^\circ$ .

Figure 6.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane  $\psi, 0^\circ$ ;  $\delta_f, 0^\circ$ ;  $\beta, 22^\circ$ .

(a)  $C_{h_r}$ ,  $C_y$ ,  $C_n$ .

Figure 6.- Concluded.

(b)  $C_l$ .

Figure 7.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane.  $\psi, 0^\circ$ ;  $\delta_f, 0^\circ$ ;  $\beta, 22^\circ$ .

(a)  $C_l$ ,  $C_n$ .

Figure 7.- Concluded.

(b)  $C_{h_r}$ ,  $C_y$ .

Figure 8.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane.  $\psi, 0^\circ$ ;  $\delta_f, 20^\circ$ ;  $\beta, 22^\circ$ .

(a)  $C_{h_r}$ .

Figure 8.- Continued.

(b)  $C_n$ .

Figure 8.- Concluded.

(c)  $C_y$ ,  $C_l$ .

## FIGURE LEGENDS - Continued

Figure 9.- Effect on rudder deflection on the aerodynamic characteristics of the XP-77 airplane.  $\psi$ ,  $0^\circ$ ;  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_{h_r}$ .

Figure 9.- Continued.

(b)  $C_n$ .

Figure 9.- Concluded.

(c)  $C_y$ .

Figure 10.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane in yaw.  $\psi$ ,  $-9.8^\circ$ ;  $\delta_f$ ,  $0^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_{h_r}$ ,  $C_y$ ,  $C_n$ .

Figure 10.- Concluded.

(b)  $C_l$ .

Figure 11.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane in yaw.  $\psi$ ,  $-9.8^\circ$ ;  $\delta_f$ ,  $20^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_n$ ,  $C_y$ ,  $C_{h_r}$ .

Figure 11.- Concluded.

(b)  $C_l$ .

Figure 12.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane.  $\psi$ ,  $-9.8^\circ$ ;  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ ;  $\alpha$ ,  $4.4^\circ$ ;  $T_c$ , 0.089.

(a)  $C_n$ ,  $C_y$ ,  $C_{h_r}$ .

Figure 12.- Concluded.

(b)  $C_l$ .

Figure 13.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane in yaw.  $\psi$ ,  $100^\circ$ ;  $\delta_f$ ,  $0^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_{h_r}$ ,  $C_y$ ,  $C_n$ .

## FIGURE LEGENDS - Continued

Figure 13.- Concluded.

(b)  $C_L$ .

Figure 14.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane in yaw.  $\psi$ ,  $10.0^\circ$ ;  $\delta_f$ ,  $20^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_n$ ,  $C_y$ ,  $C_{h_r}$ .

Figure 14.- Concluded.

(b)  $C_L$ .

Figure 15.- Effect of rudder deflection on the aerodynamic characteristics of the XP-77 airplane in yaw.  $\psi$ ,  $10.0^\circ$ ;  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ ;  $\alpha$ ,  $4.4^\circ$ ;  $T_c$ , 0.089.

(a)  $C_n$ ,  $C_y$ ,  $C_{h_r}$ .

Figure 15.- Concluded.

(b)  $C_L$ .

Figure 16.- Effect of yaw angle on rudder effectiveness and rate of change of rudder hinge-moment coefficient with rudder deflection.  $\delta_f$ ,  $0^\circ$ ;  $\beta$ ,  $22^\circ$ .

Figure 17.- Effect of yaw angle on rudder effectiveness and rate of change of rudder hinge-moment coefficient with rudder deflection.  $\delta_f$ ,  $20^\circ$ ;  $\beta$ ,  $22^\circ$ .

Figure 18.- Effect of yaw angle on rudder effectiveness and rate of change of rudder hinge-moment coefficient with rudder deflection.  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ .

Figure 19.- Variation of  $(C_{h_r})_{\delta_r=0^\circ}$  with angle of yaw  $\beta$ ,  $22^\circ$ .

Figure 20.- Variation of rudder angle with angle of yaw for  $C_n = 0$ .  $\delta_f$ ,  $0^\circ$ ;  $\beta$ ,  $22^\circ$ .

Figure 21.- Variation of rudder angle with angle of yaw for  $C_n = 0$ .  $\delta_f$ ,  $20^\circ$ ;  $\beta$ ,  $22^\circ$ .

Figure 22.- Variation of rudder angle with angle of yaw for  $C_n = 0$ .  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ .

## FIGURE LEGENDS - Continued

Figure 23.- Effect of elevator deflection on the aerodynamic characteristics of the XP-77 airplane.

$\psi, 0^\circ; \delta_r, 0^\circ; \delta_a, 0^\circ.$

Figure 24.- Effect of elevator deflection on the aerodynamic characteristics of the XP-77 airplane.

$\psi, -9.8^\circ; \delta_r, 0^\circ; \delta_a, 0^\circ.$

Figure 25.- Effect of elevator deflection on the aerodynamic characteristics of the XP-77 airplane.

$\psi, 10^\circ; \delta_r, 0^\circ; \delta_a, 0^\circ.$

Figure 26.- Effect of aileron deflection on the aerodynamic characteristics of the XP-77 airplane, propeller removed; right aileron locked;  $\psi, 0^\circ; \delta_f, 0^\circ.$

(a)  $C_n, C_{h_a}.$

Figure 26.- Concluded.

(b)  $C_l.$

Figure 27.- Effect of aileron deflection on the aerodynamic characteristics of the XP-77 airplane.

Propeller removed; right aileron locked;  $\psi, 0^\circ; \delta_f, 2.0^\circ.$

(a)  $C_n, C_{h_a}.$

Figure 27.- Concluded.

(b)  $C_l.$

Figure 28.- Effect of aileron deflection on the aerodynamic characteristics of the XP-77 airplane.

Propeller removed; right aileron locked;  $\psi, 0^\circ; \delta_f, 55^\circ.$

(a)  $C_n, C_{h_a}.$

Figure 28.- Concluded.

(b)  $C_l.$

Figure 29.- Variation of  $C_m, C_L,$  and  $C_{DR}$  with angle of yaw for  $\delta_r = 0^\circ. \delta_f, 0^\circ; \beta, 22^\circ.$

(a)  $\alpha, -2.4^\circ$  and  $T_c, 0; \alpha, -1.2^\circ$  and  $T_c 0.021.$

FIGURE LEGENDS - Continued

Figure 29.- Concluded.

(b)  $\alpha$ ,  $4.8^\circ$  and  $T_c$ , 0.038;  $\alpha$ ,  $8.8^\circ$  and  $T_c$ , 0.049.

Figure 30.- Variation of  $C_{DR}$ ,  $C_L$ , and  $C_m$  with angle of yaw for  $\delta_r = 0^\circ$ .  $\delta_f$ ,  $20^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_{DR}$ ,  $C_L$ .

Figure 30.- Concluded.

(b)  $C_m$ .

Figure 31.- Variation of  $C_{DR}$ ,  $C_L$ , and  $C_m$  with angle of yaw for  $\delta_r = 0^\circ$ ;  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_{DR}$ ,  $C_L$ .

Figure 31.- Concluded.

(b)  $C_m$ .

Figure 32.- Variation of  $C_n$ ,  $C_l$ , and  $C_y$  with angle of yaw for  $\delta_r = 0^\circ$ .  $\delta_f$ ,  $0^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $\alpha$ ,  $-24^\circ$  and  $T_c$ , 0;  $\alpha$ ,  $-1.2^\circ$  and  $T_c$ , 0.021.

Figure 32.- Concluded.

(b)  $\alpha$ ,  $4.8^\circ$  and  $T_c$ , 0.038;  $\alpha$ ,  $8.8^\circ$  and  $T_c$ , 0.049.

Figure 33.- Variation of  $C_n$ ,  $C_l$ , and  $C_y$  with angle of yaw for  $\delta_r = 0^\circ$ .  $\delta_f$ ,  $20^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_n$ ,  $C_l$

Figure 33.- Concluded.

(b)  $C_y$ .

Figure 34.- Variation of  $C_n$ ,  $C_l$ , and  $C_r$  with angle of yaw for  $\delta_r = 0^\circ$ .  $\delta_f$ ,  $55^\circ$ ;  $\beta$ ,  $22^\circ$ .

(a)  $C_n$ ,  $C_l$ .

Figure 34.- Concluded.

(b)  $C_y$ .

## FIGURE LEGENDS - Concluded

Figure 35.- Variation of yawing-moment coefficient for  $C_{hr} = 0$  with angle of yaw.  $\delta_f, 0^\circ$ ;  $\beta, 22^\circ$ .

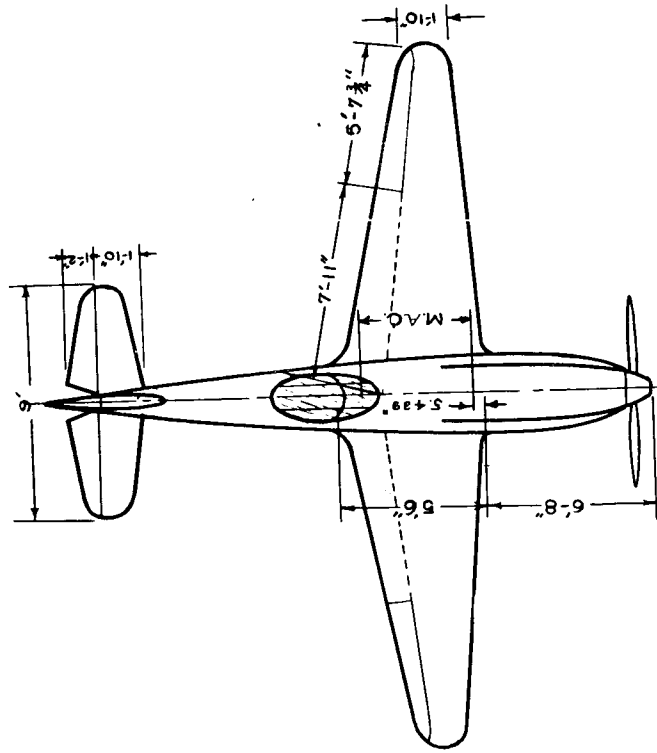
(a)  $\alpha, -2.4^\circ$  and  $T_c, 0$ ;  $\alpha, -1.2^\circ$  and  $T_c, 0.021$ .

Figure 35.- Concluded.

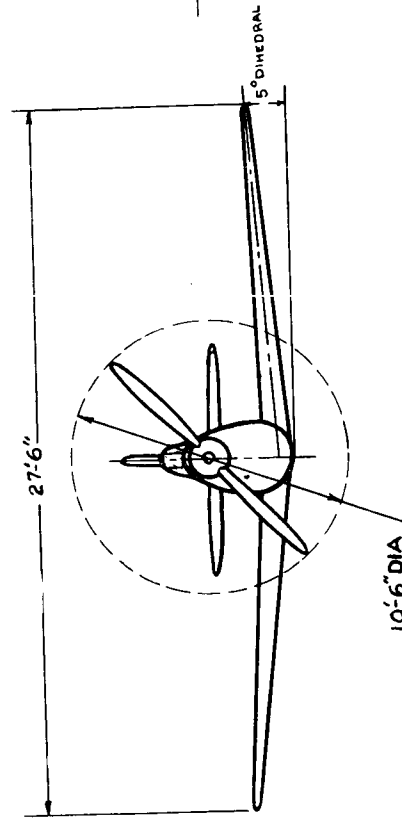
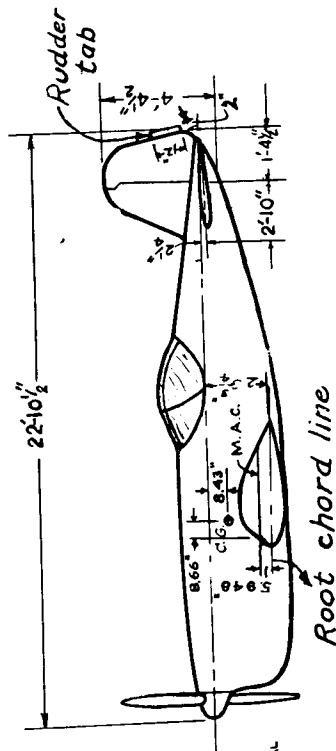
(b)  $\alpha, 4.8^\circ$  and  $T_c, 0.038$ ;  $\alpha, 8.8^\circ$  and  $T_c, 0.049$ .

Figure 36.- Variation of yawing-moment coefficient for  $C_{hr} = 0$  with angle of yaw.  $\delta_f, 20^\circ$ ;  $\beta, 22^\circ$ .

Figure 37.- Variation of yawing-moment coefficient for  $C_{hr} = 0$  with angle of yaw.  $\delta_f, 55^\circ$ ;  $\beta, 22^\circ$ .

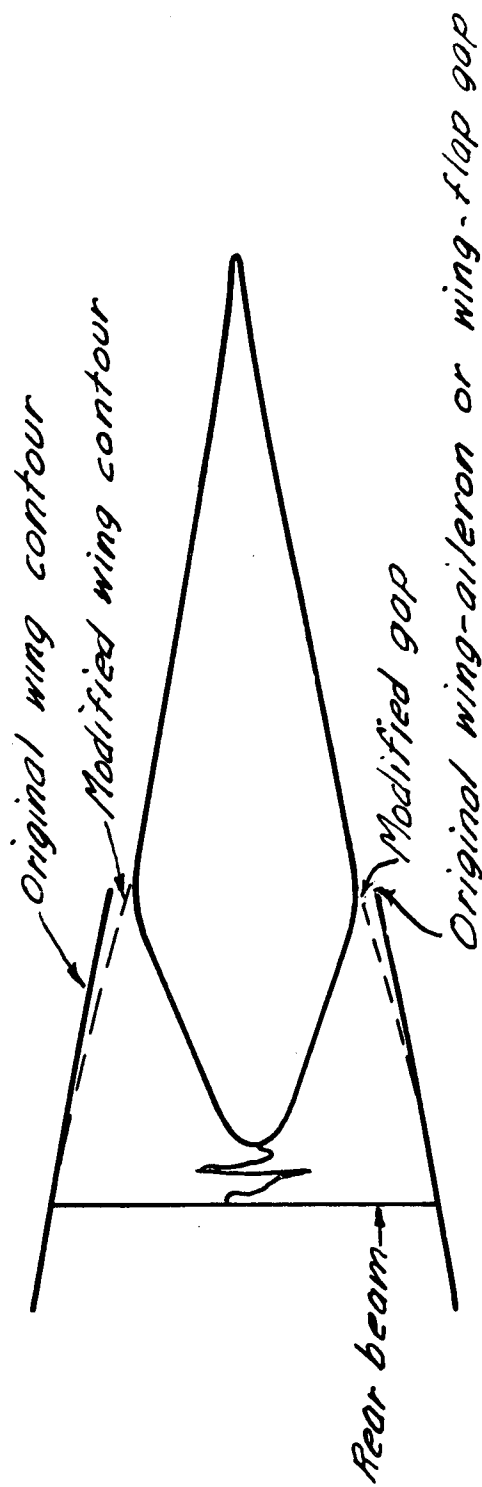


|                                                             |                           |
|-------------------------------------------------------------|---------------------------|
| Gross weight                                                | 3500 lbs                  |
| Areas                                                       | sq ft                     |
| Wing                                                        | 100.00                    |
| Aileron (total)                                             | 504                       |
| Horizontal tail (total)                                     | 18.90                     |
| Stabilizer                                                  | 12.50                     |
| Elevator, aft of hinge line                                 | 6.40                      |
| Vertical tail (total)                                       | 1237                      |
| Fin                                                         | 738                       |
| Rudder, aft of hinge line                                   | 475                       |
| Flap, aft of hinge line (total)                             | 858                       |
| Mean aerodynamic chord                                      | 3.99 ft                   |
| Airfoil section                                             |                           |
| Root                                                        | Modified NACA-65(216)-017 |
| Tip                                                         | Modified NACA-671-(13)15  |
| Angle of incidence (at root section)                        | 2°                        |
| Washout                                                     | 1°30'                     |
| Propeller - Aeroproducts A-15-132-two blades constant speed |                           |



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Figure 1.-Three-view showing of the XP-77 airplane used in the full-scale tunnel.



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Figure 2.- Modifications made to wing to provide smooth contours and smaller wing-aileron or wing-flap gaps.



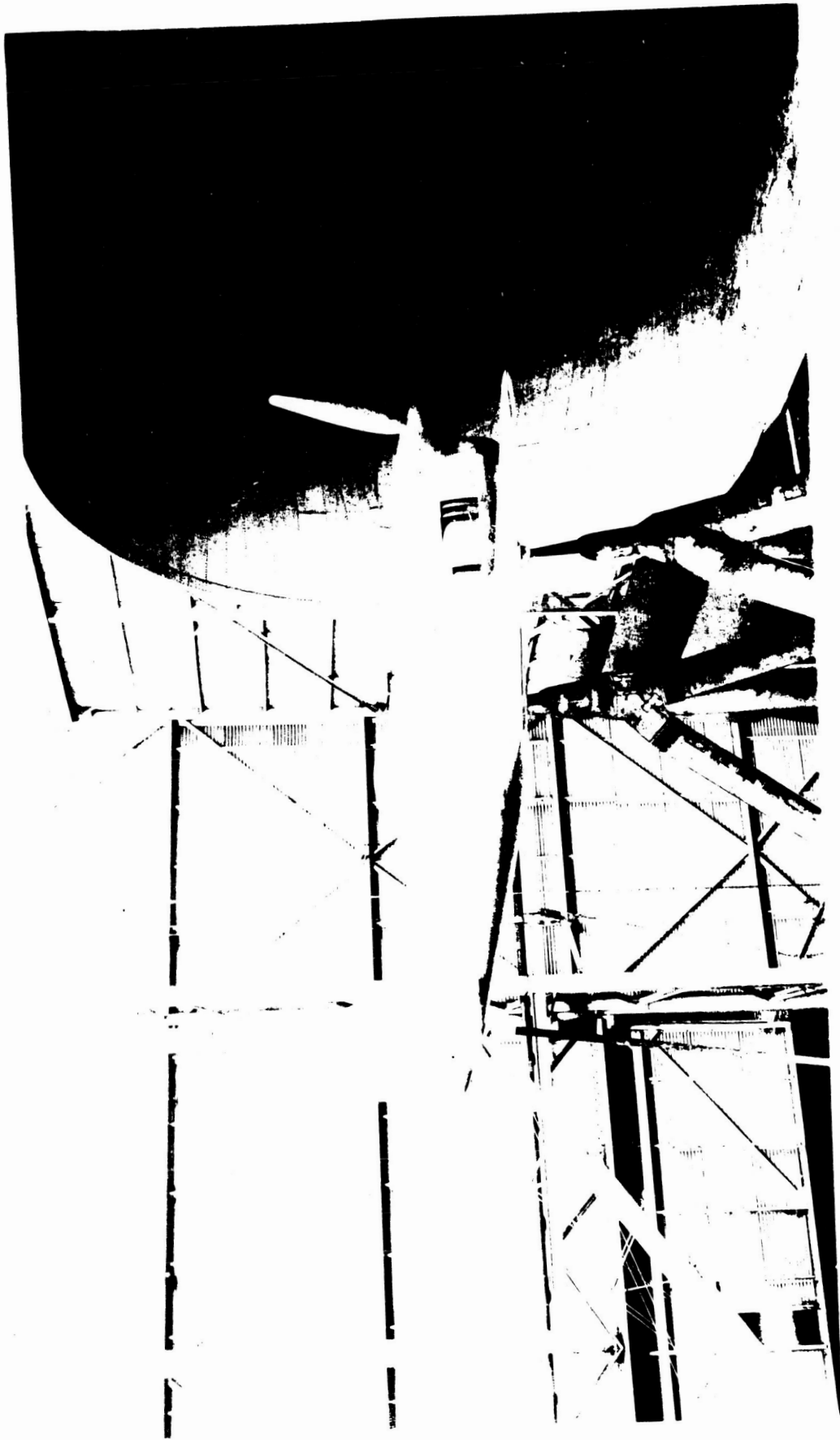


Figure 3.- The XP-77 airplane mounted in the NACA full-scale tunnel.  $\psi$ ,  $0^\circ$ .

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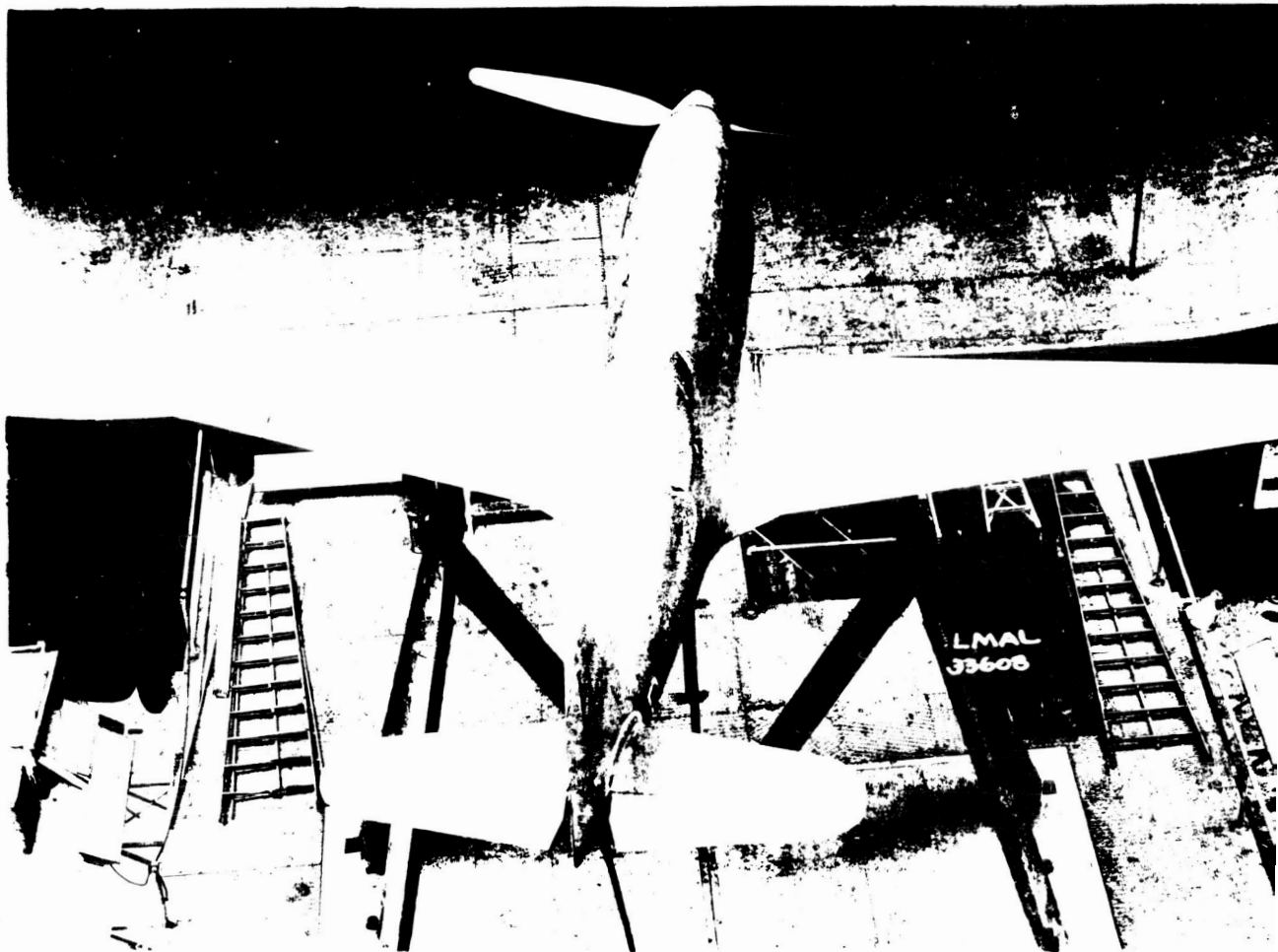
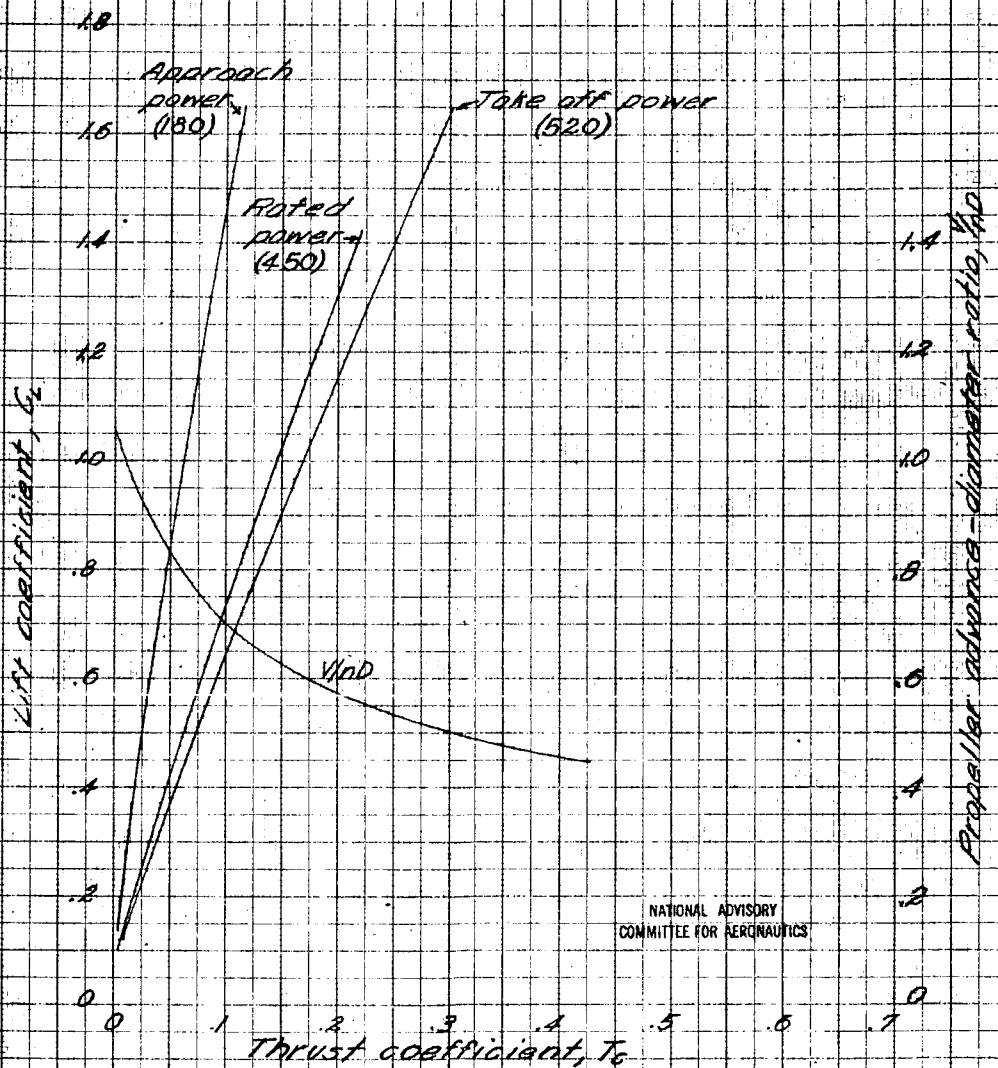


Figure 4.- The XP-77 airplane mounted in the NACA full-scale tunnel.  $\psi, 10^{\circ}$ .

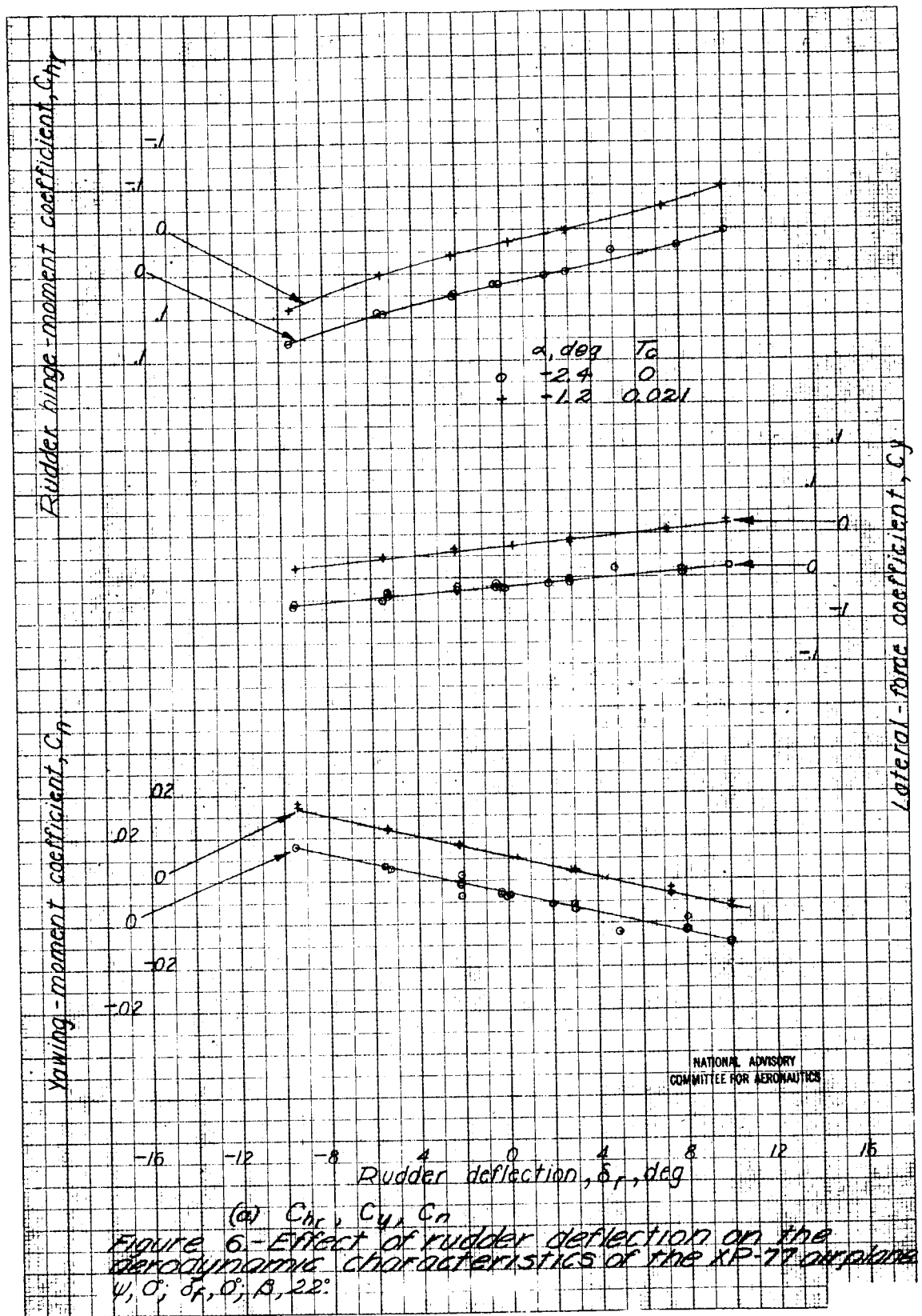
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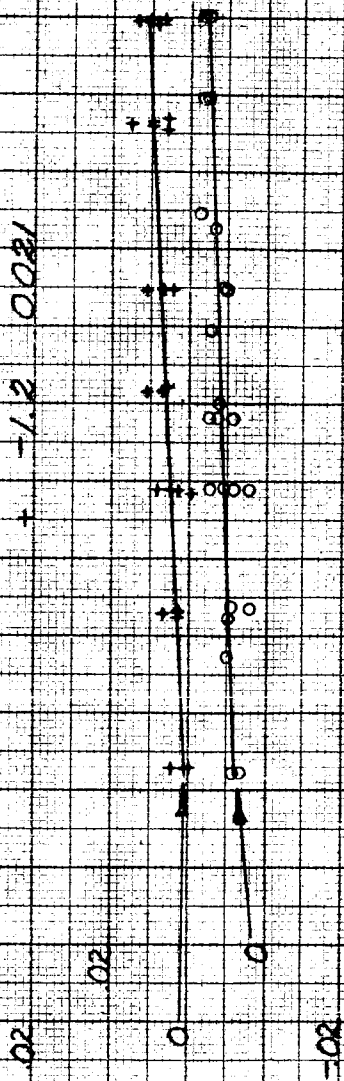
Figure 5. - Variation of  $T_c$  with  $C_L$  and  $V_{AD}$  for constant power operation at 500 level.  $\beta, 22^\circ$ .

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Rolling-moment coefficient,  $C_2$

$\alpha, \text{deg}$   $T_c$   
 $\circ$  -2.4 0  
 $+$  -1.2 0.081

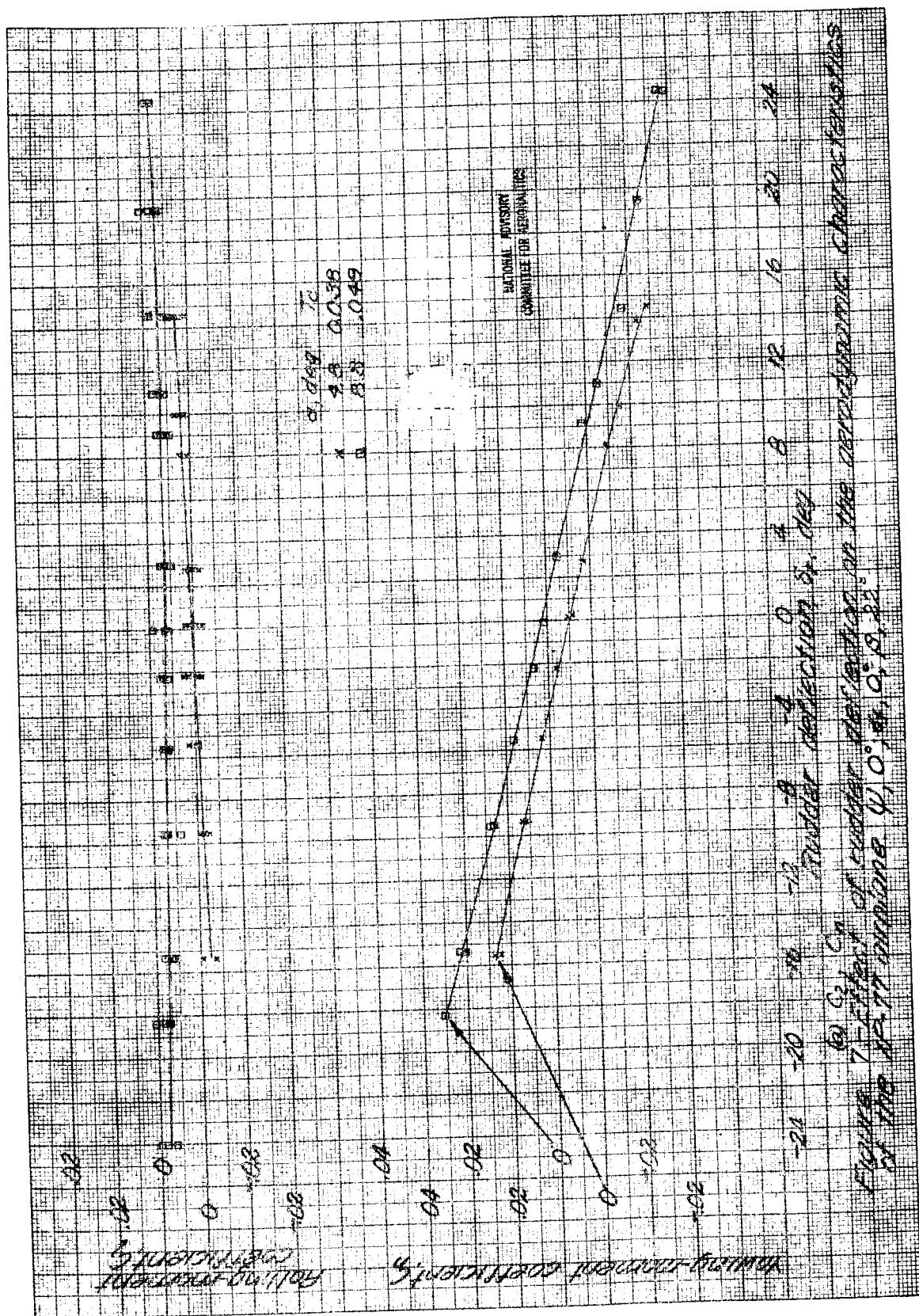


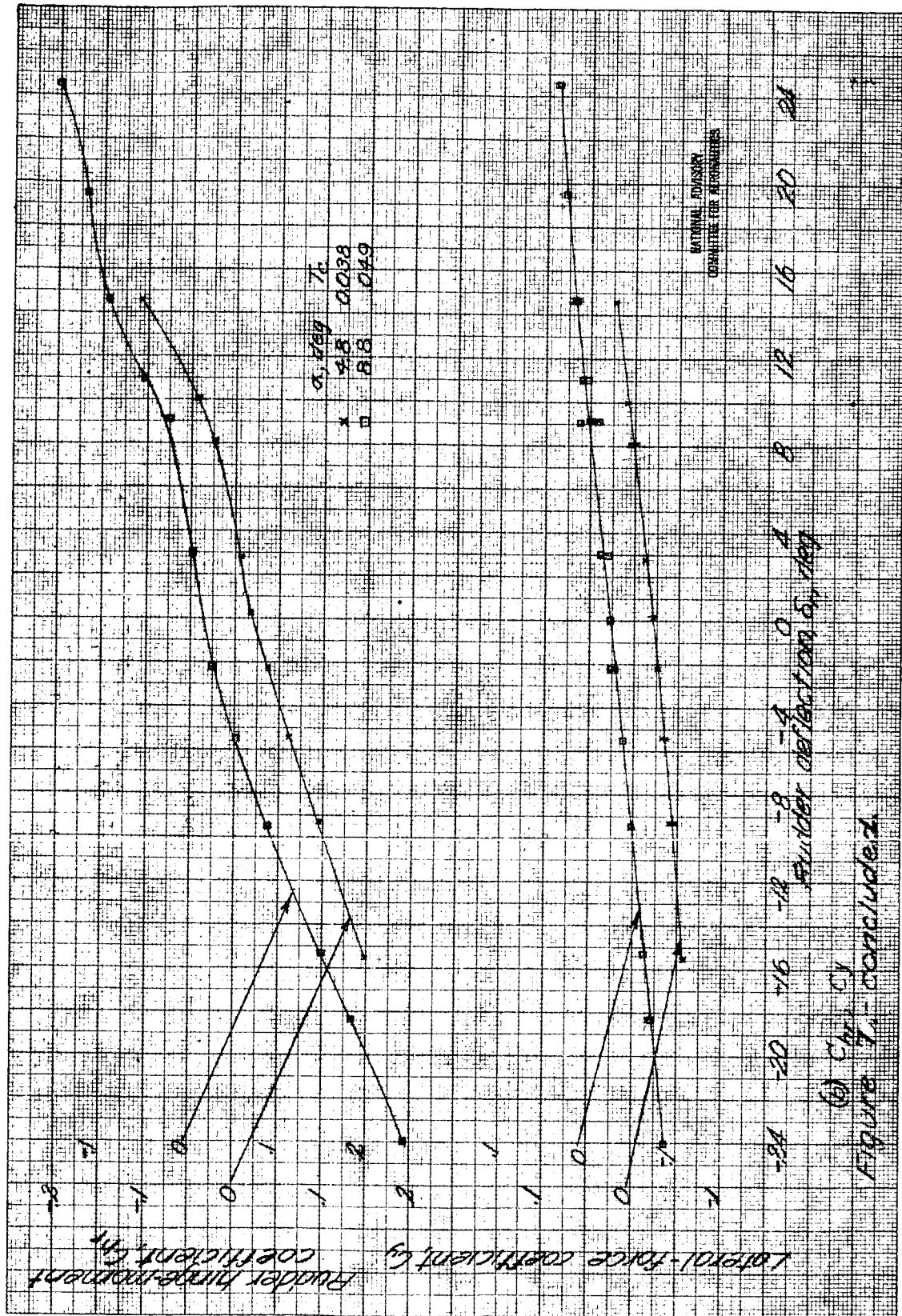
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-12 -8 -4 0 4 8 12  
 Sudder deflection,  $\delta$ , deg

(b)  $C_2$

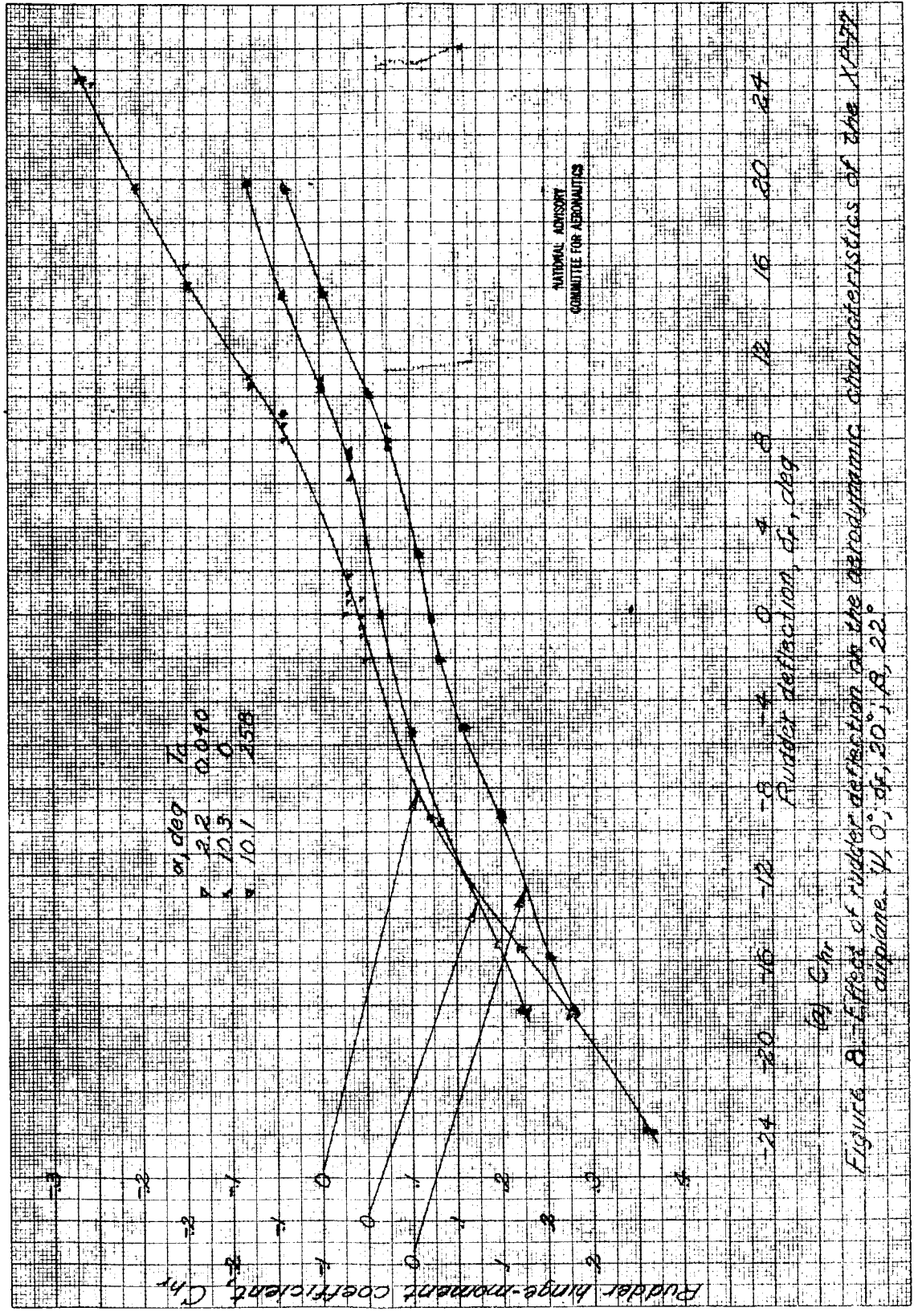
Figure 6-1—continued.





(b)  $C_{L, C_V}$   
 Figure 7. - Concluded

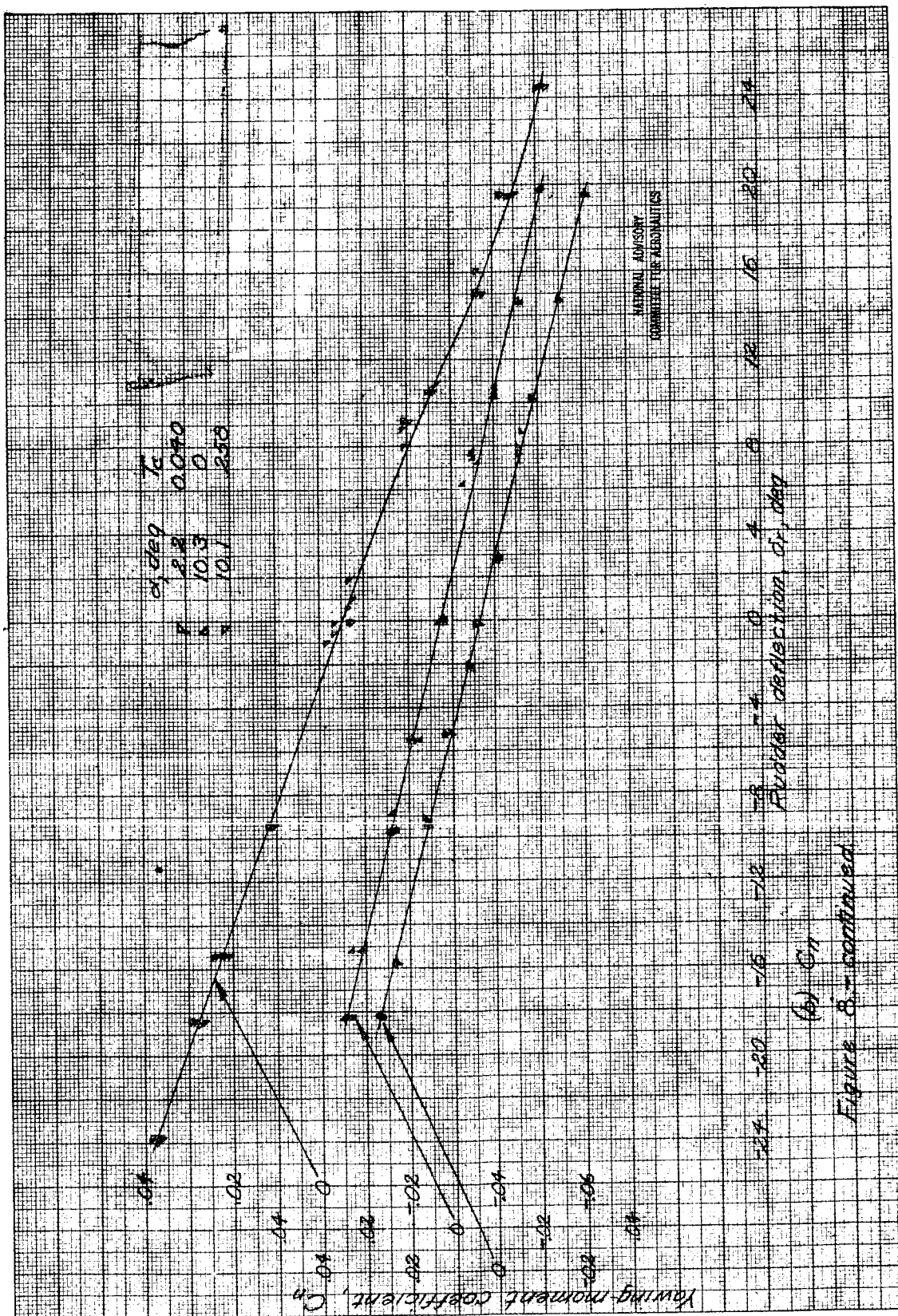


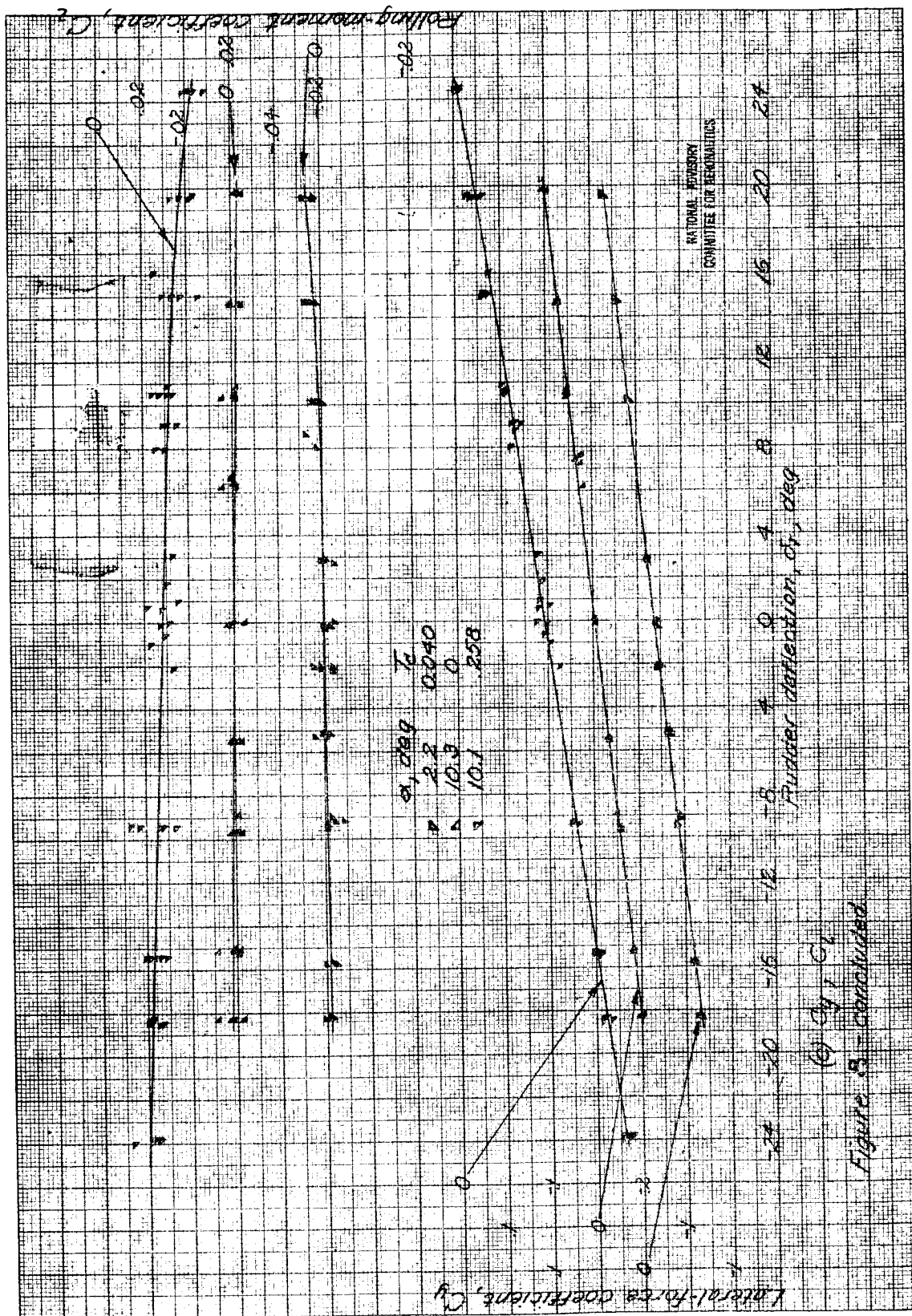


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69 C hr







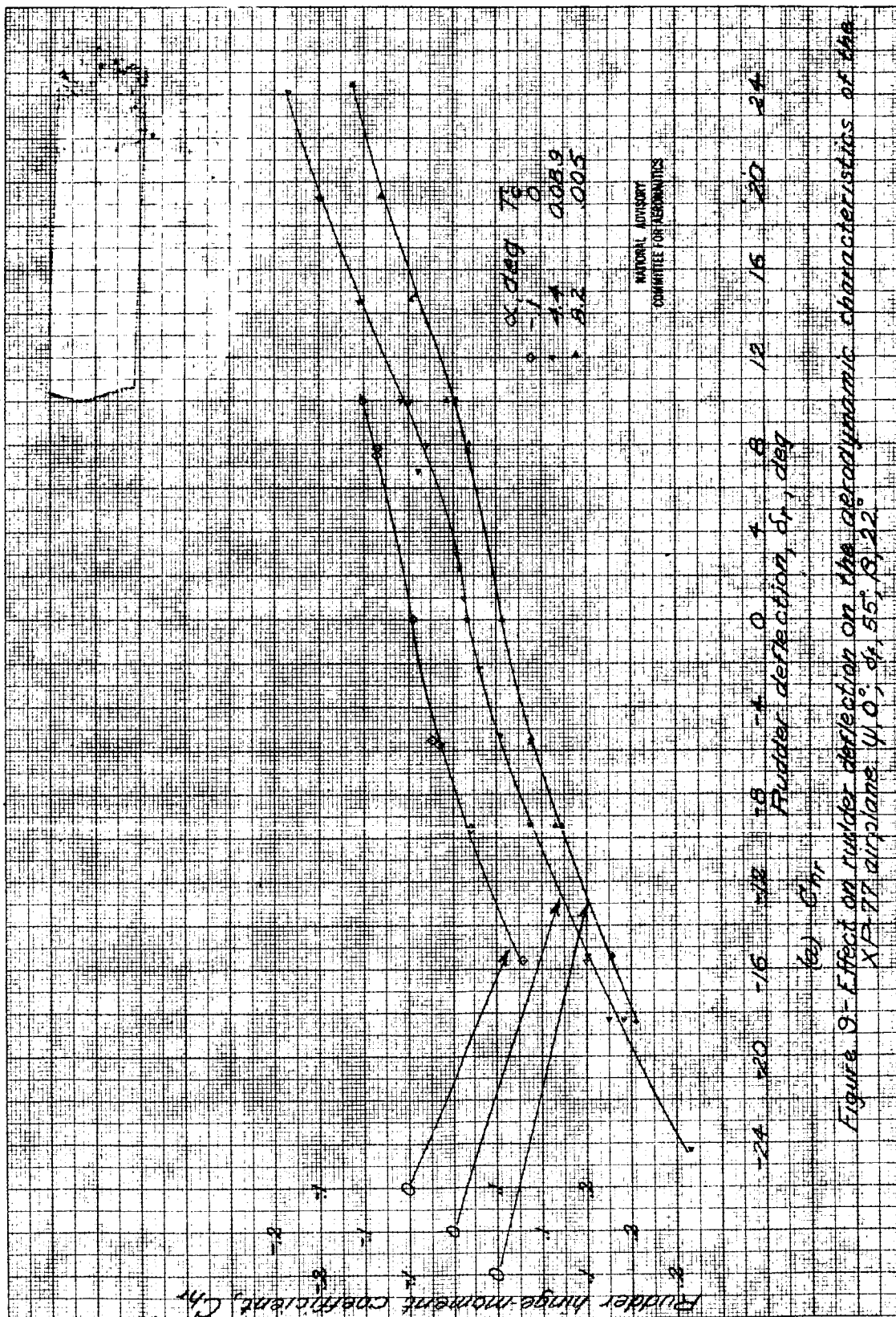
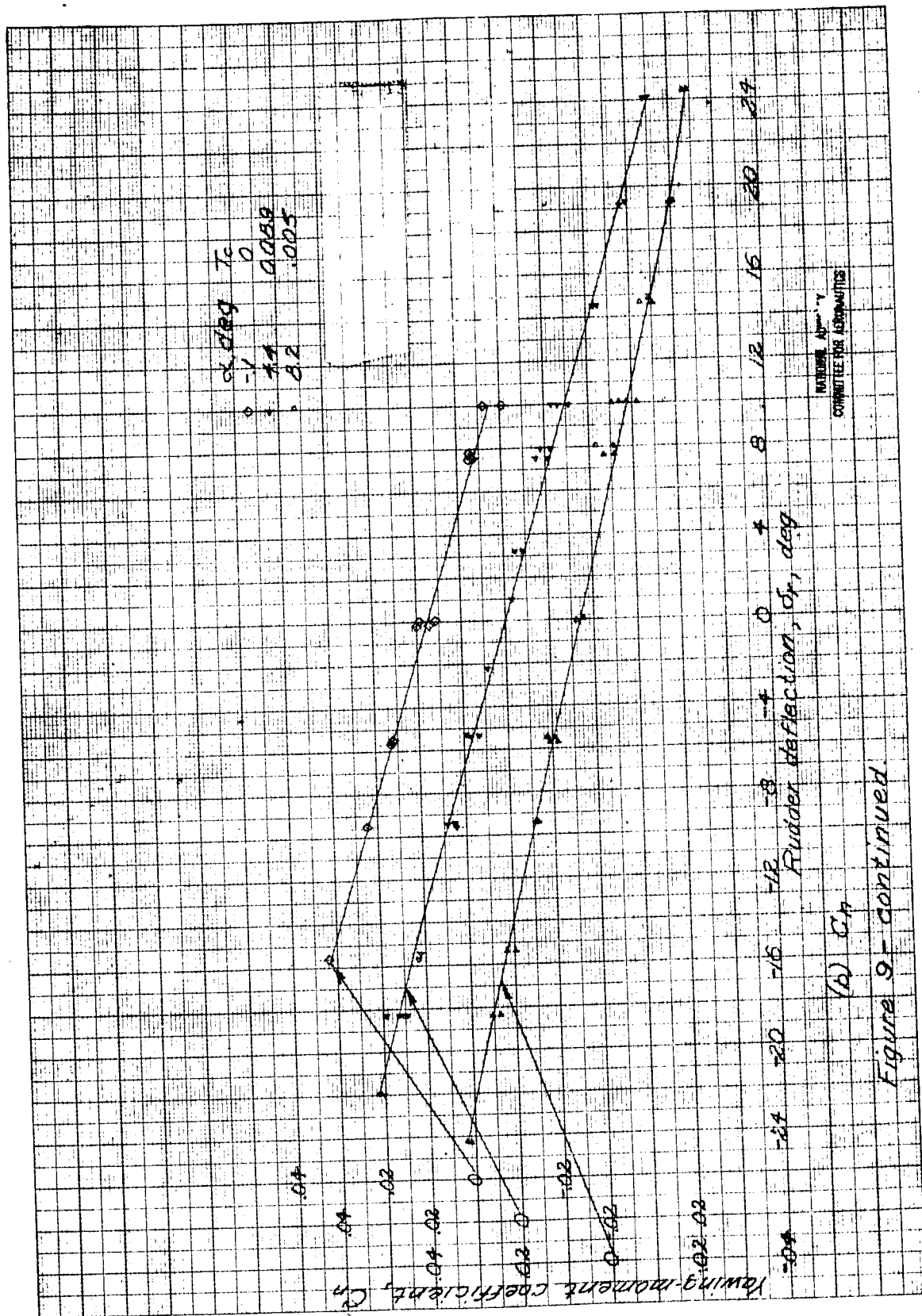
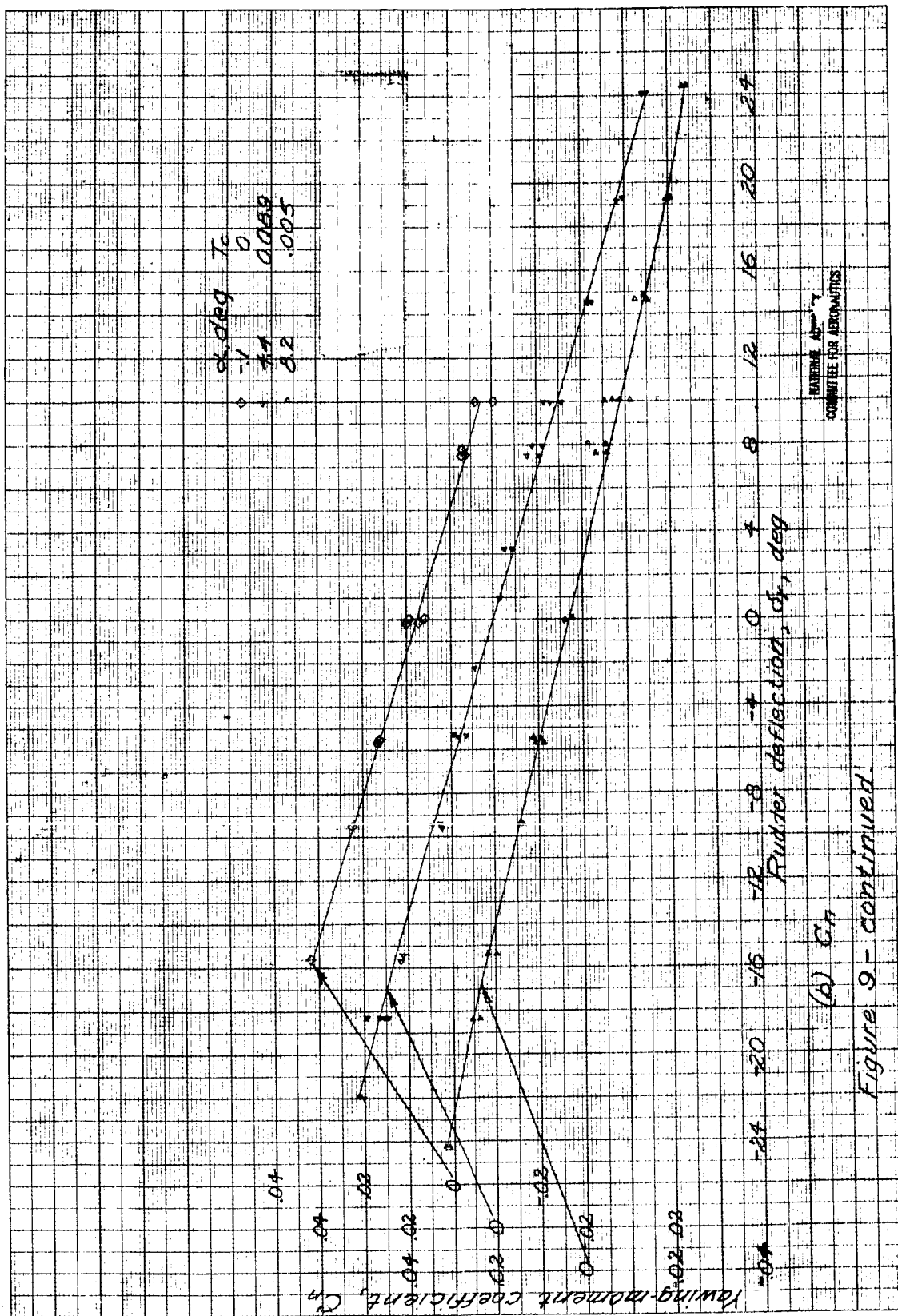


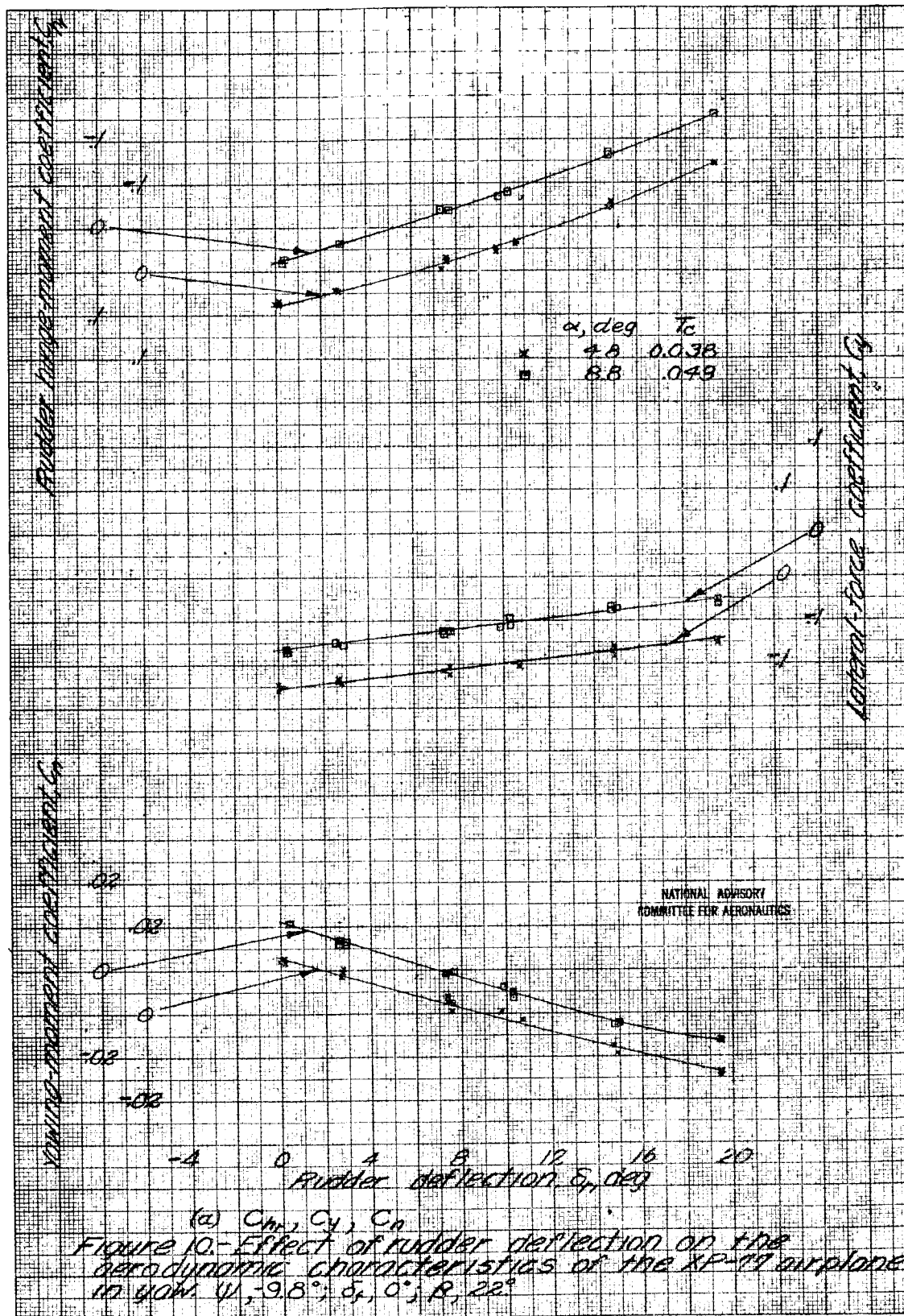
Figure 9- Effect on rudder deflection on the aerodynamic characteristics of the XP-77 airplane.  $\delta_r$ , 0°, 5°, 10°, 15°, 20°, 22°



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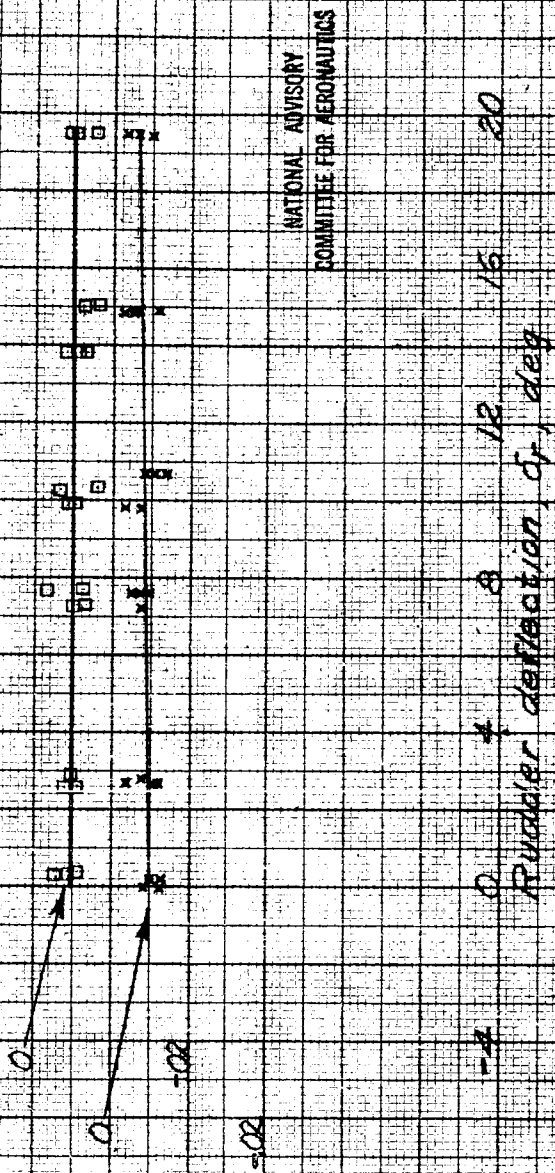






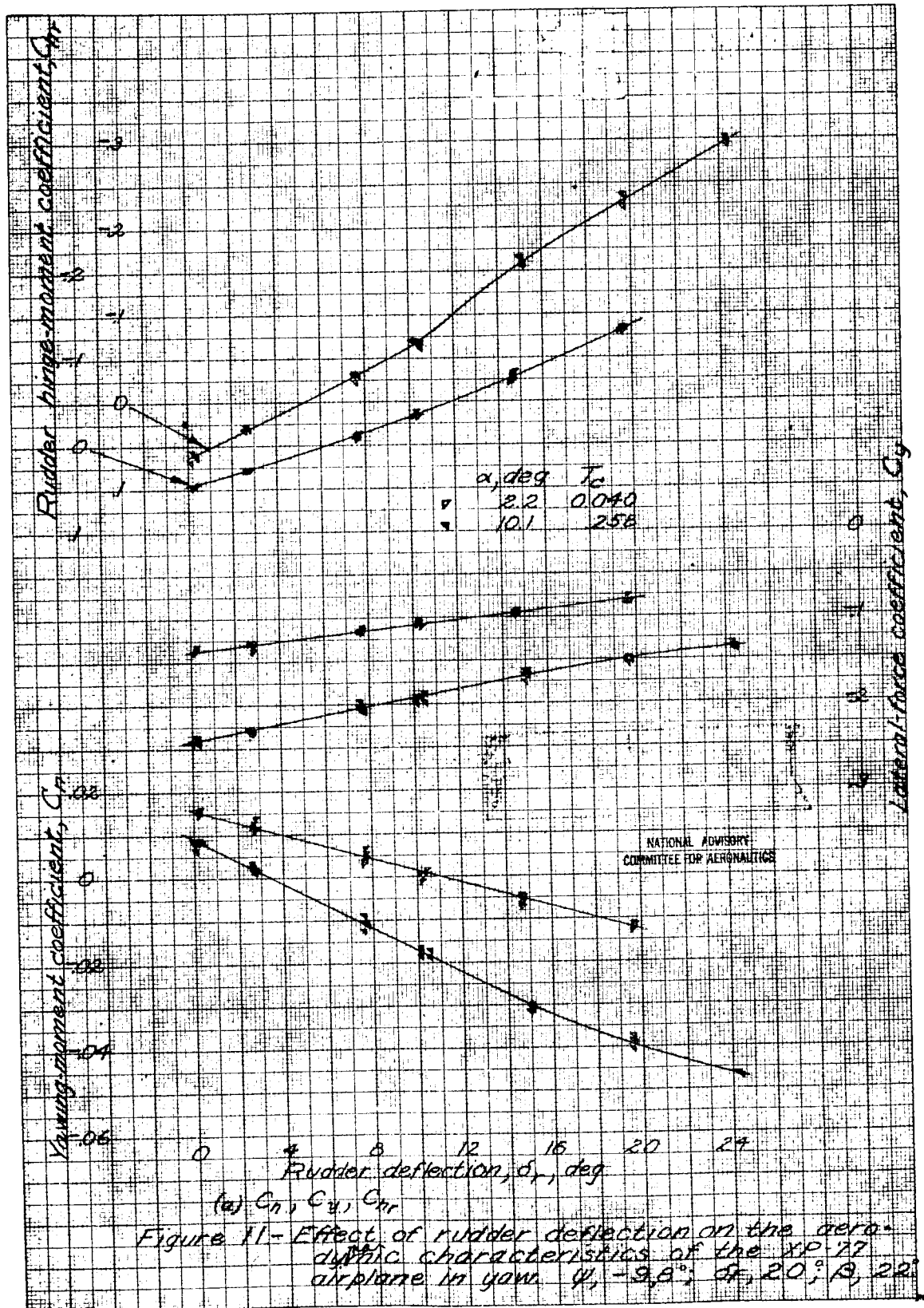
Rolling-moment coefficient,  $C_r$

$\alpha$ , deg  $T_c$   
4.8 0.038  
6.8 0.49



(b)  $C_r$

FIGURE 10-continued.





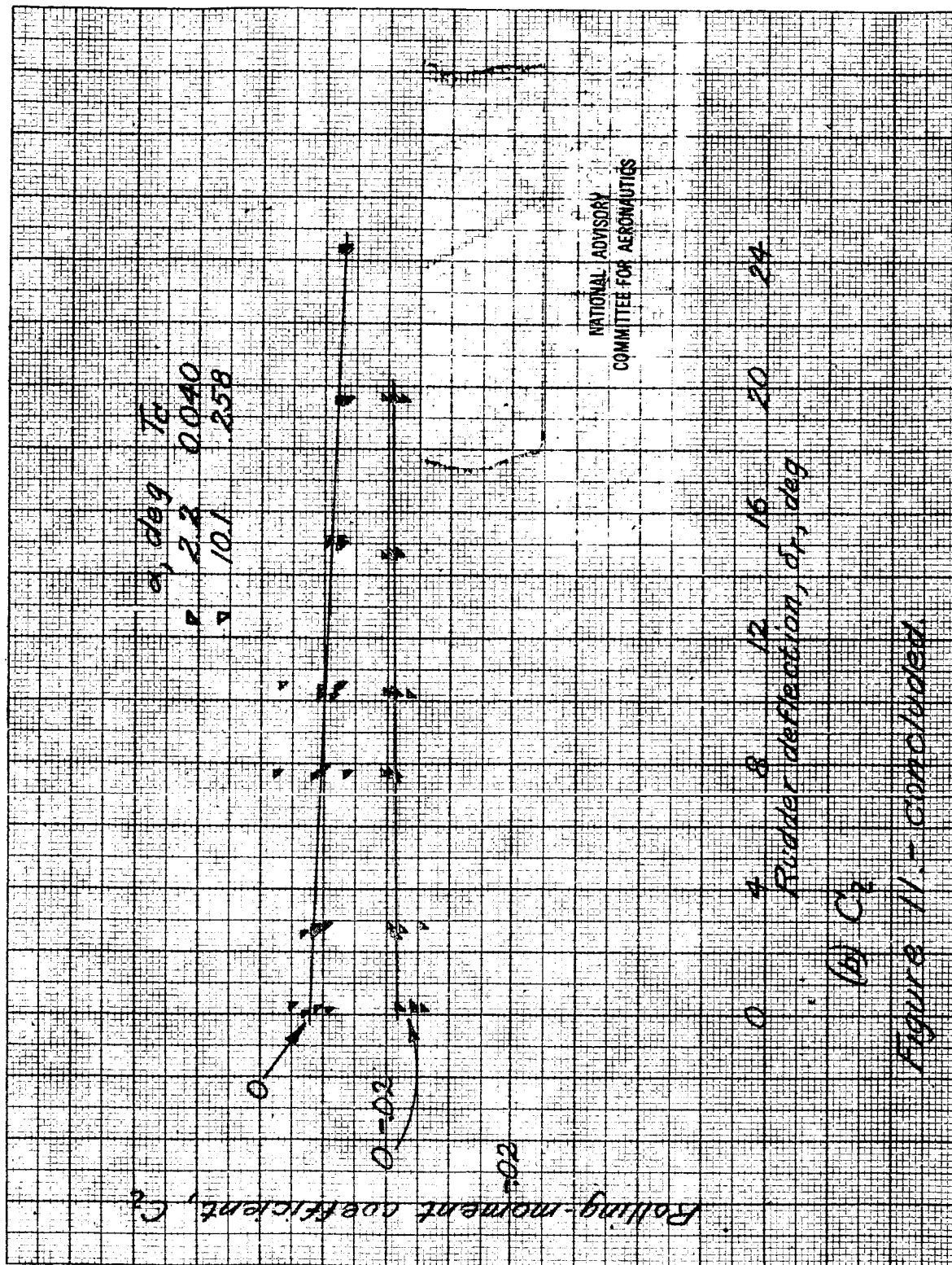
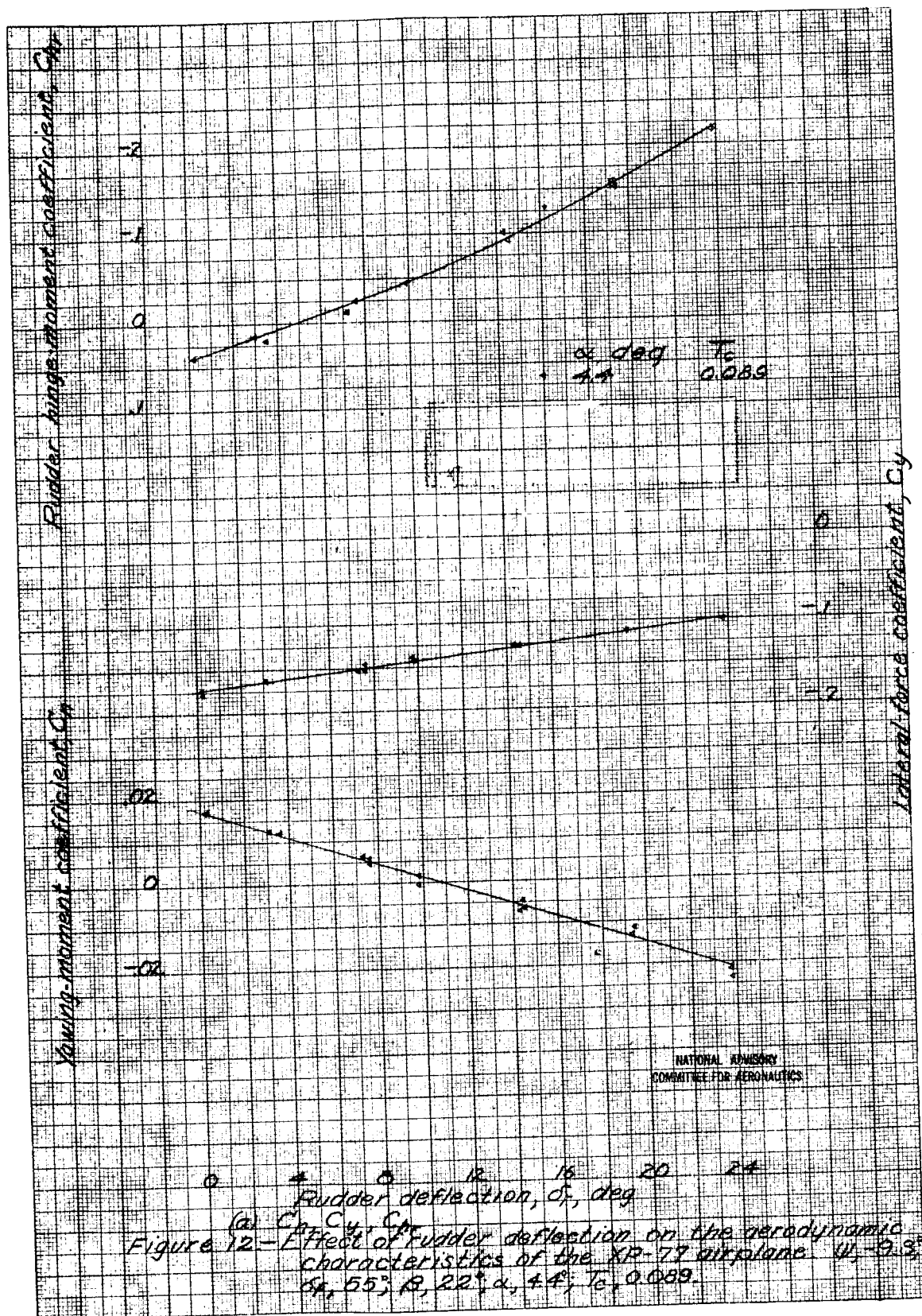


Figure 11.- continued



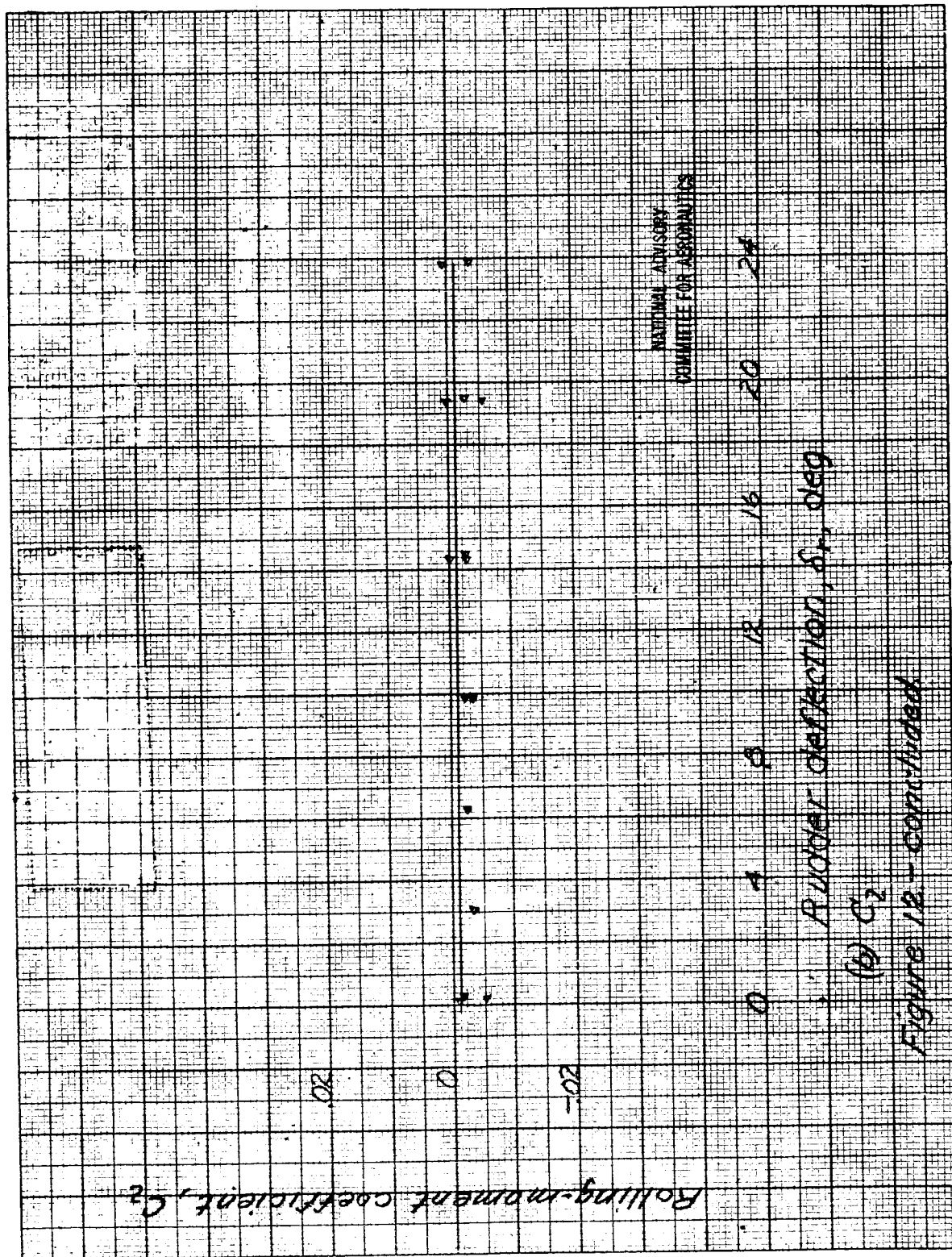
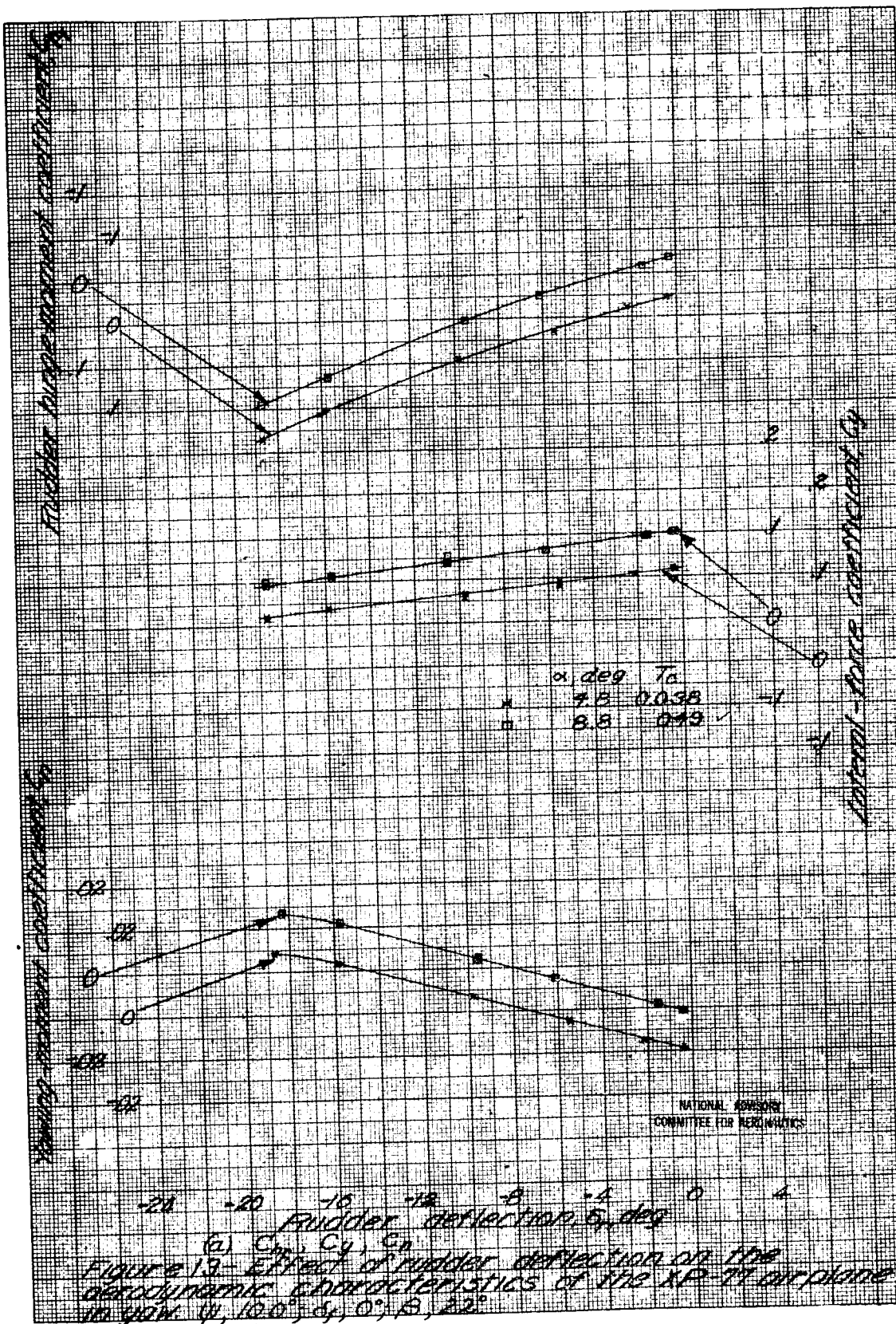
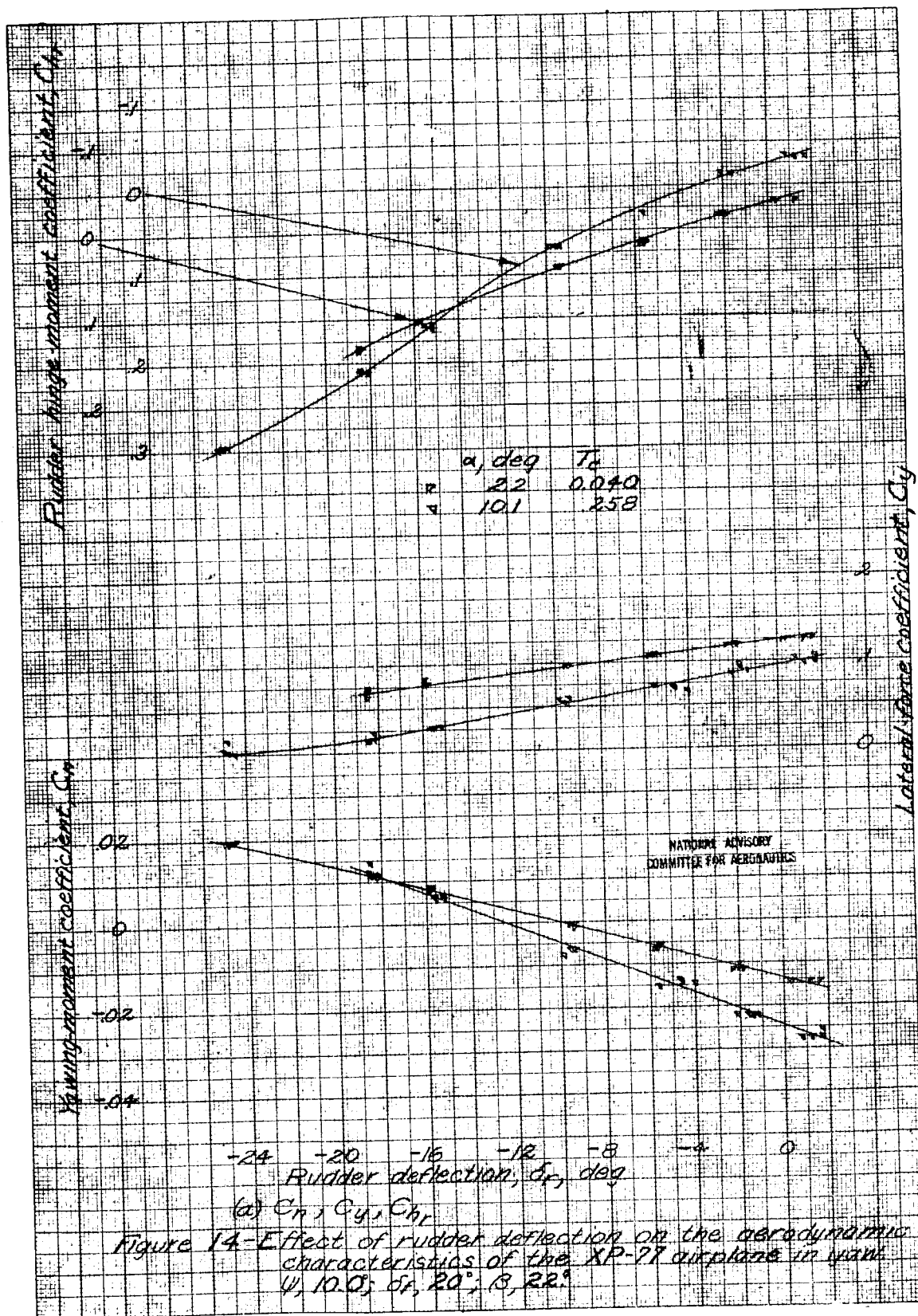


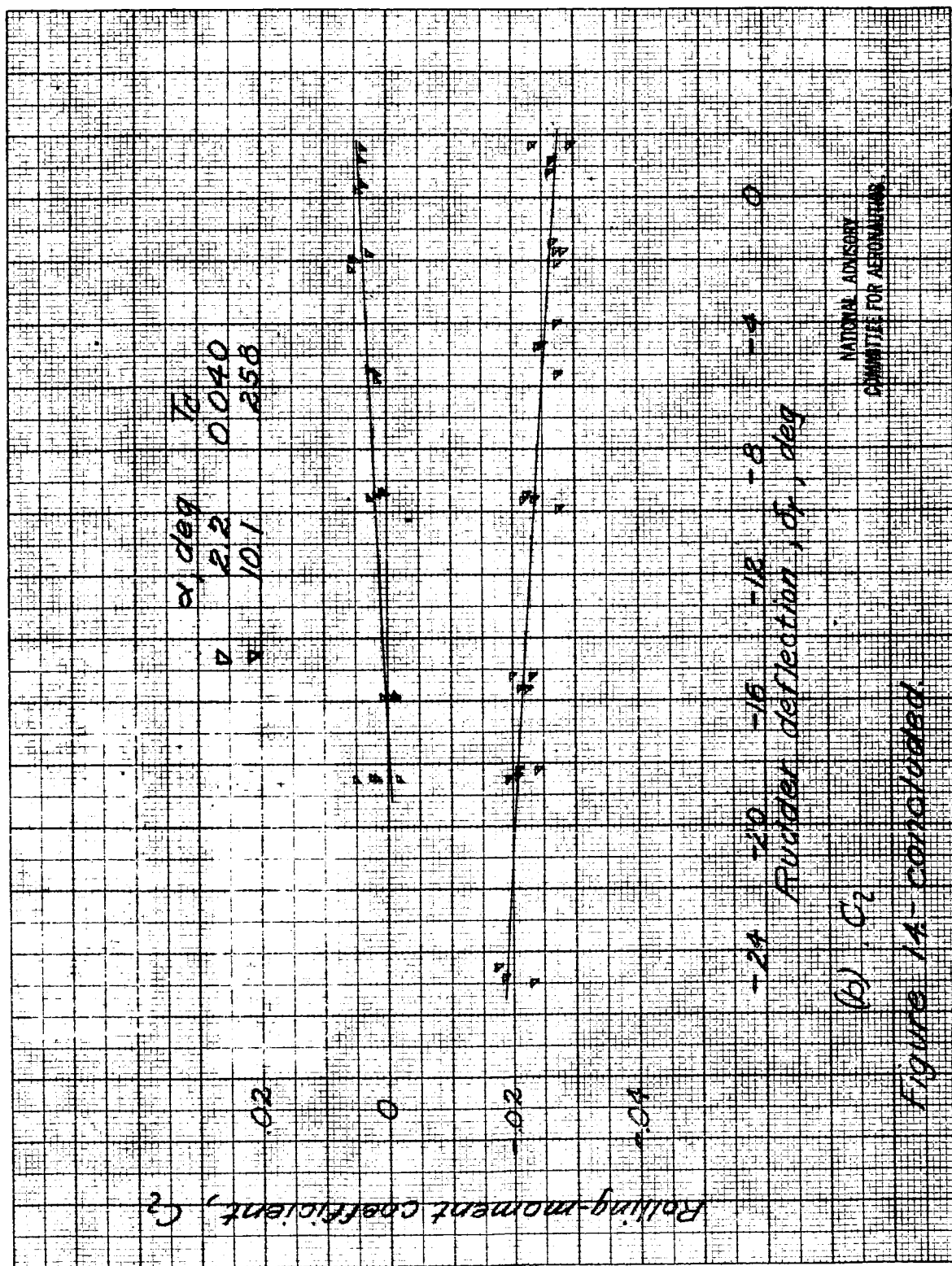
Figure 12 - continued

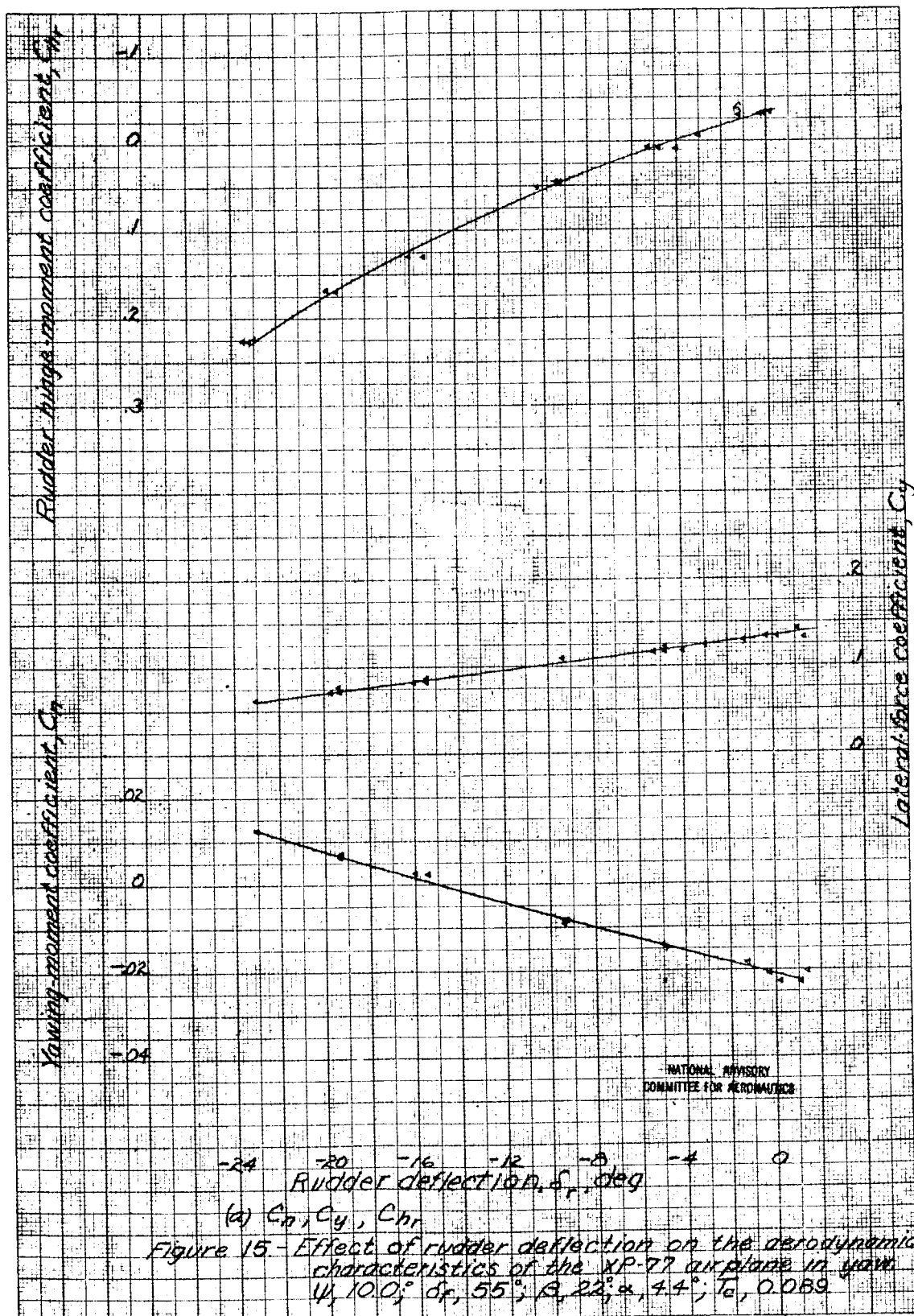




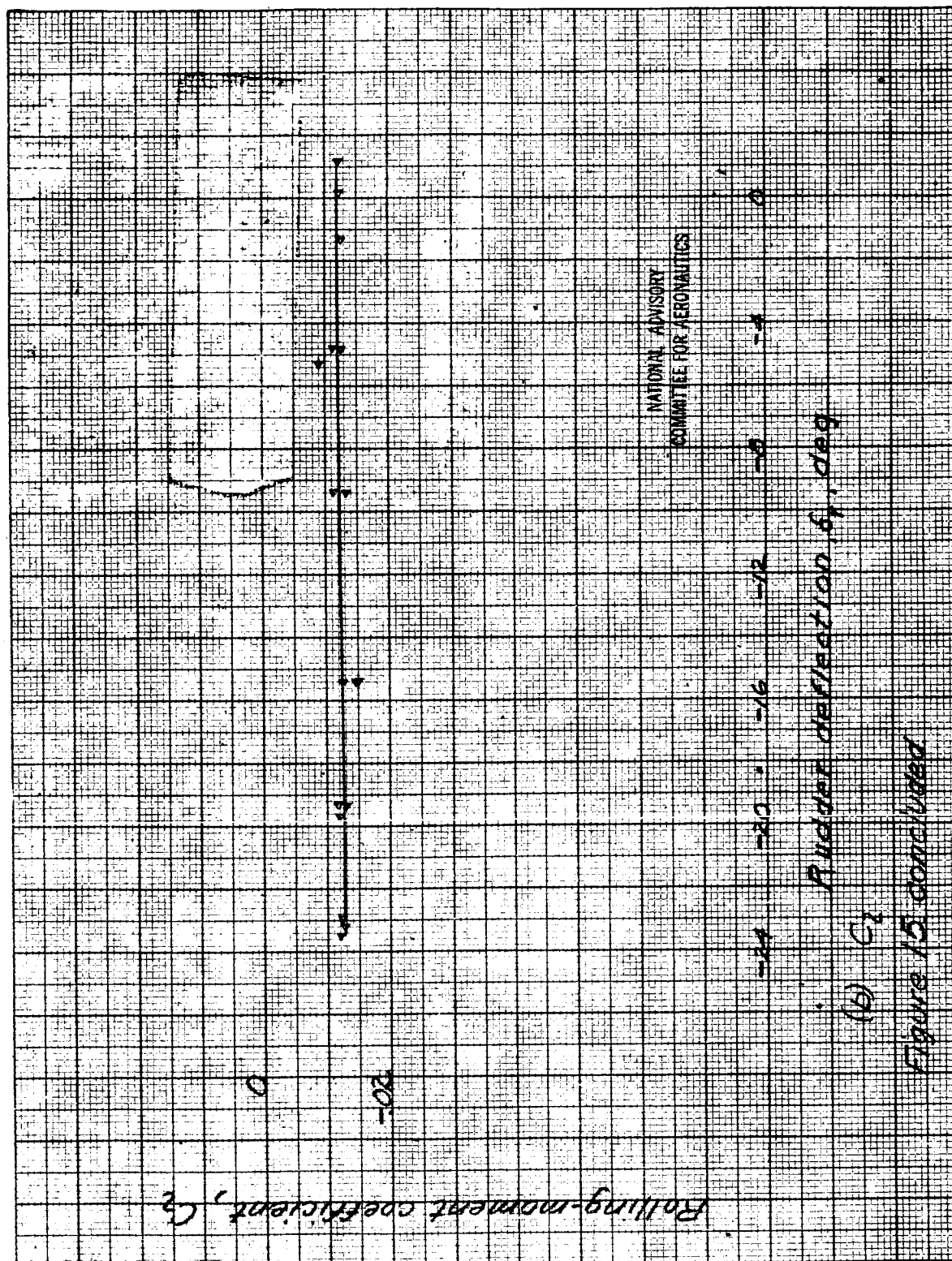


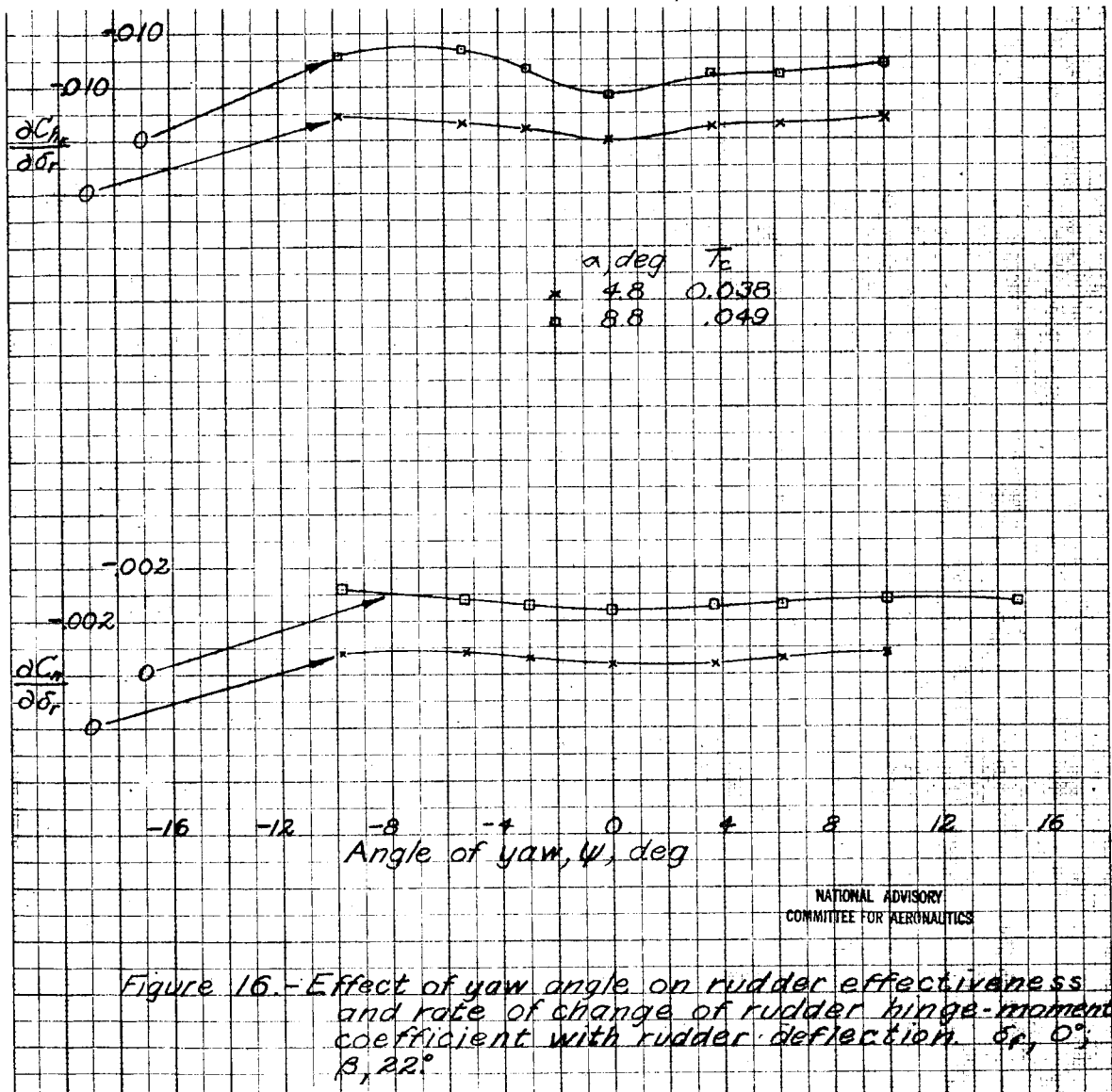
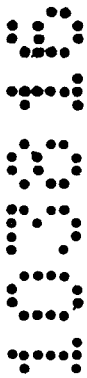












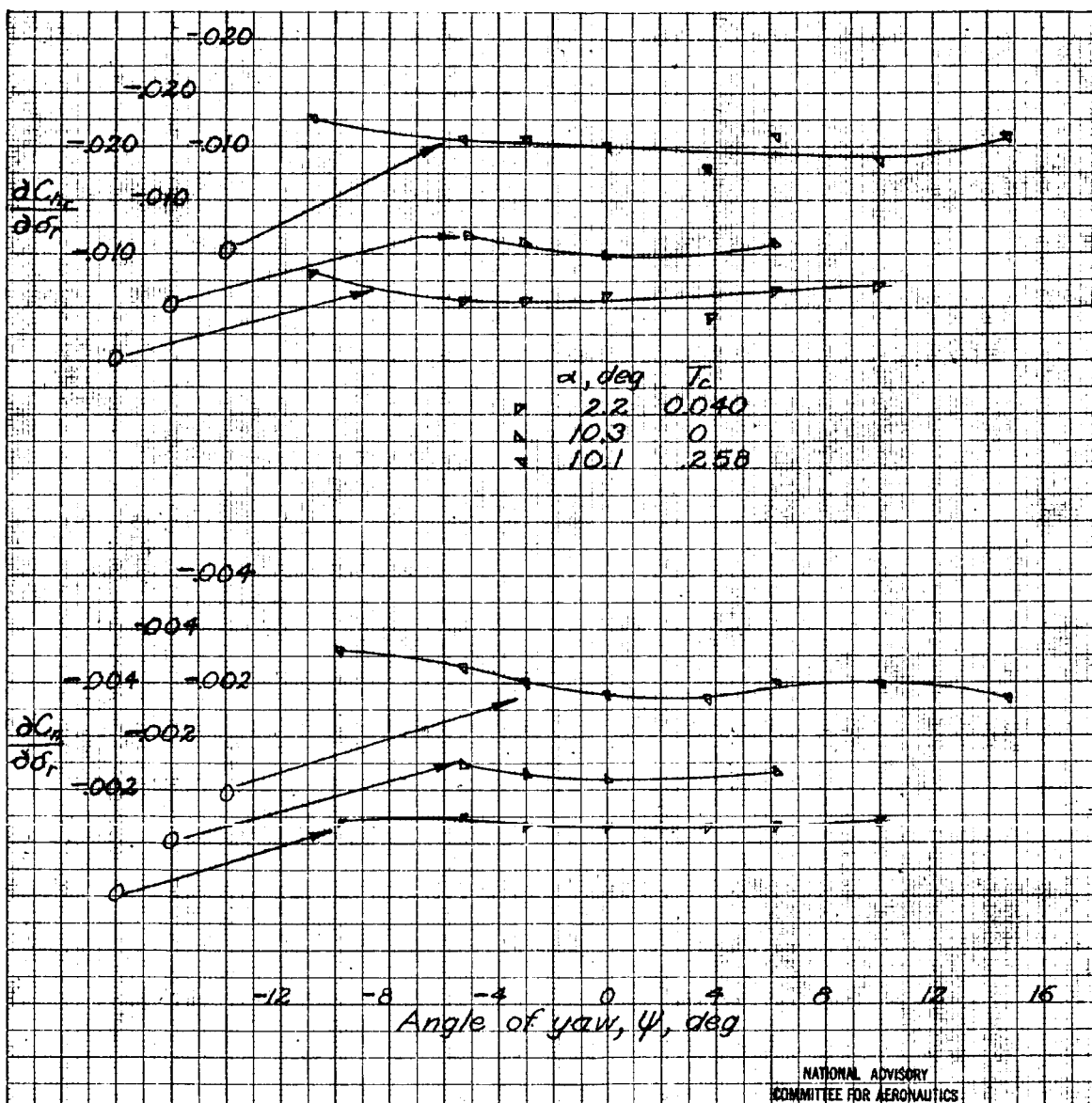
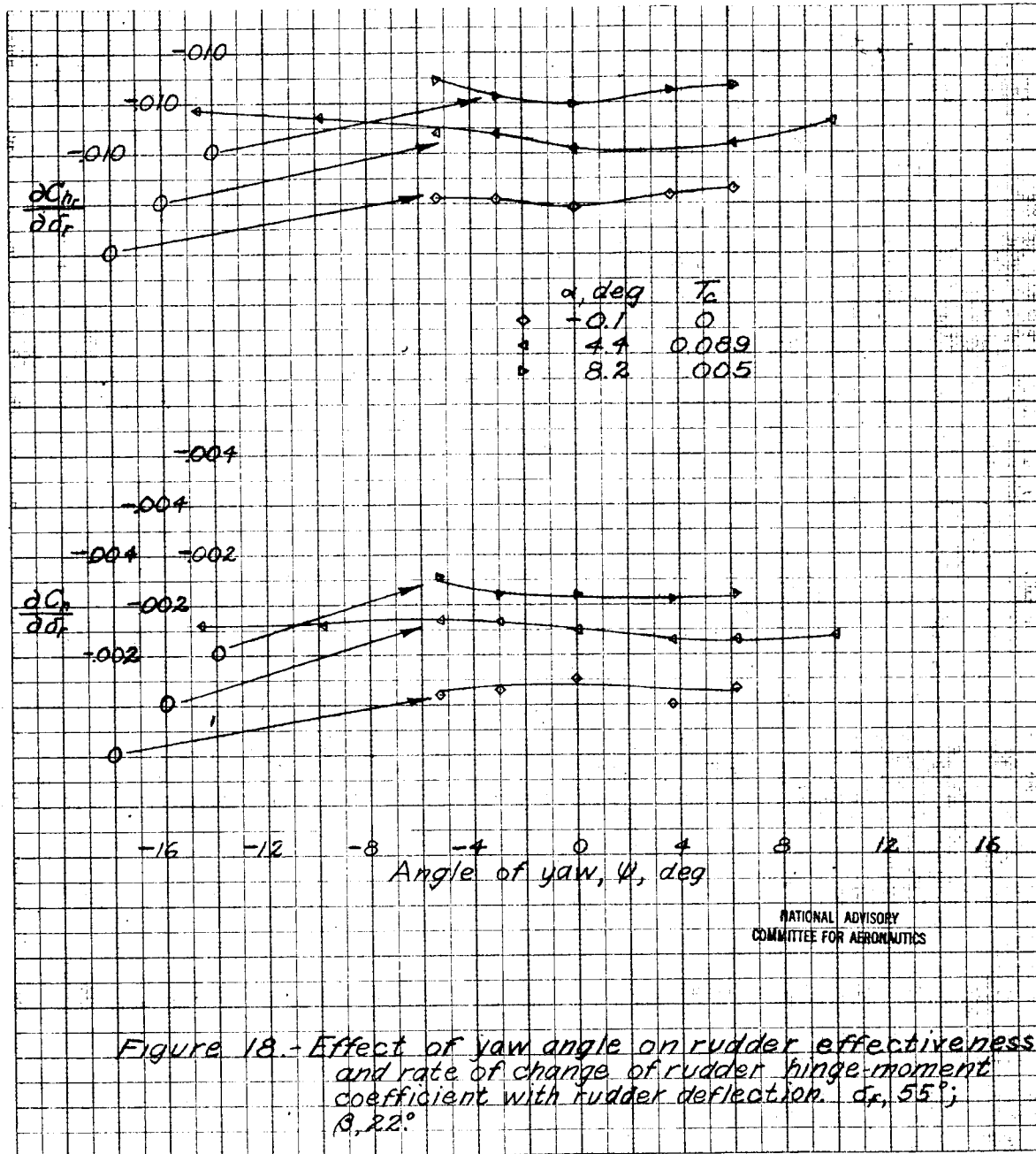


Figure 17.-Effect of yaw angle on rudder effectiveness and rate of change of rudder hinge-moment coefficient with rudder deflection  $\delta_r$ , 20°;  $\beta$ , 22°.



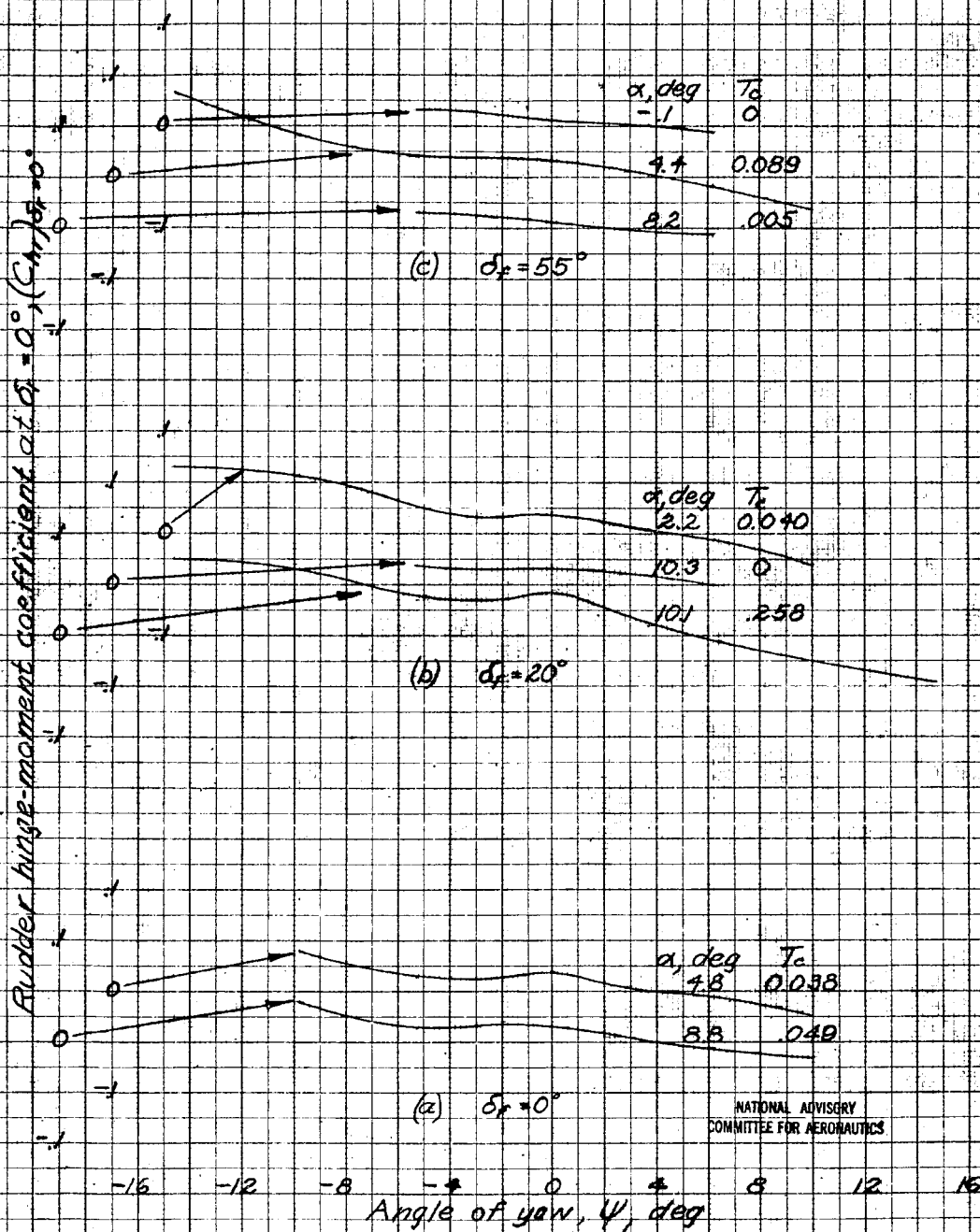
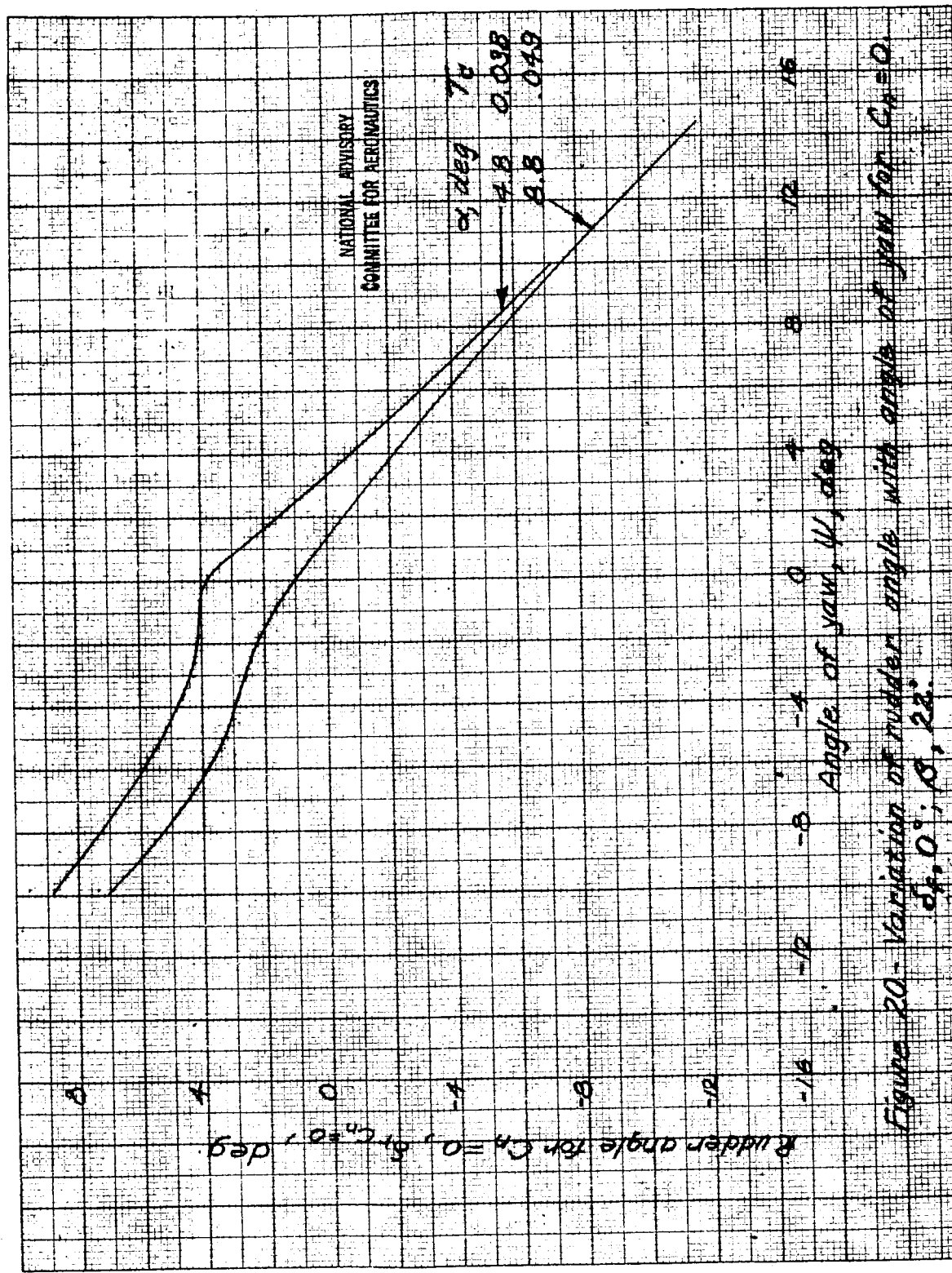


Figure 19.-Variation of  $(C_{H_r})_{\alpha=0^\circ}$  with angle of yaw  $\beta, \text{deg}$



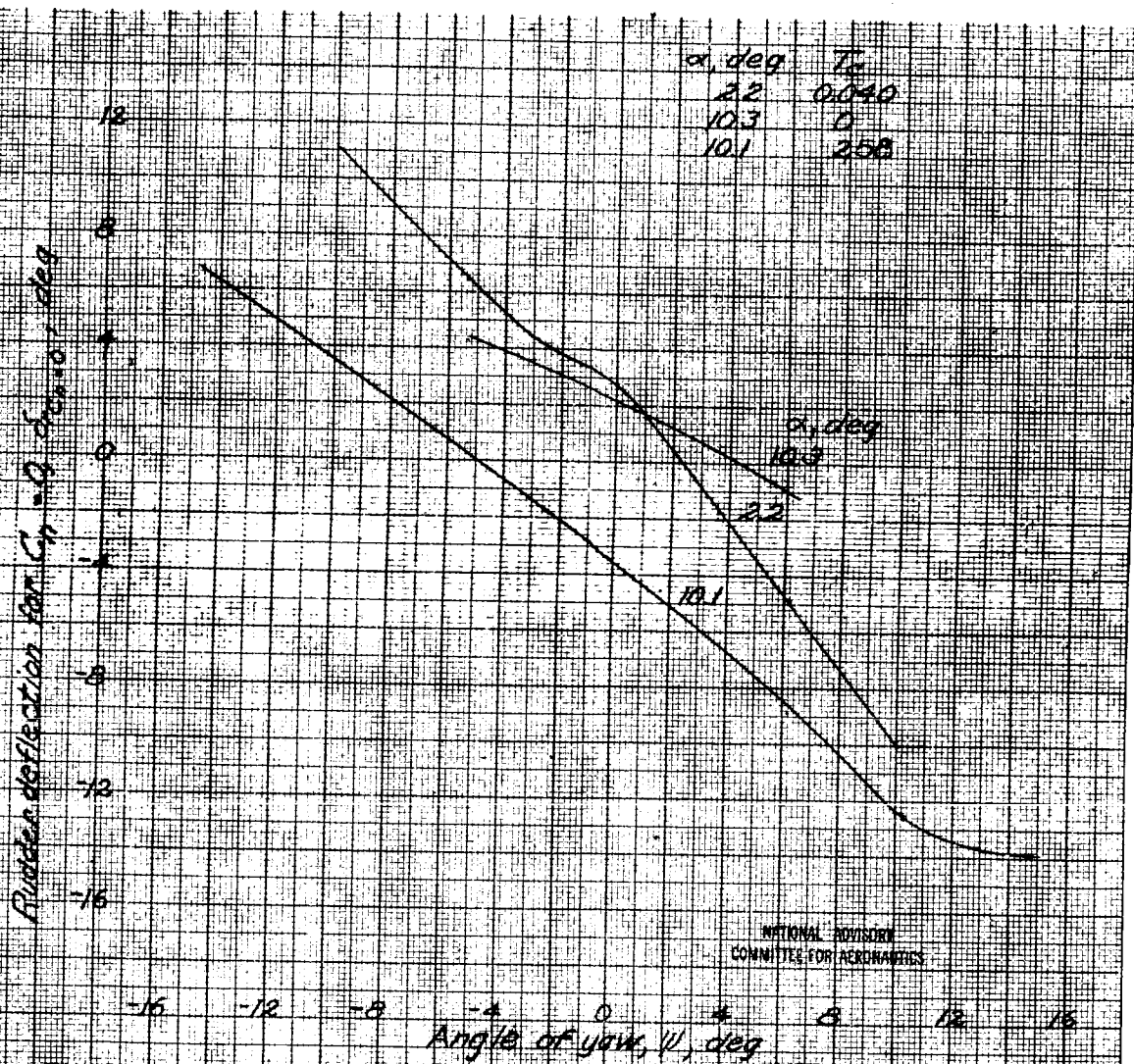
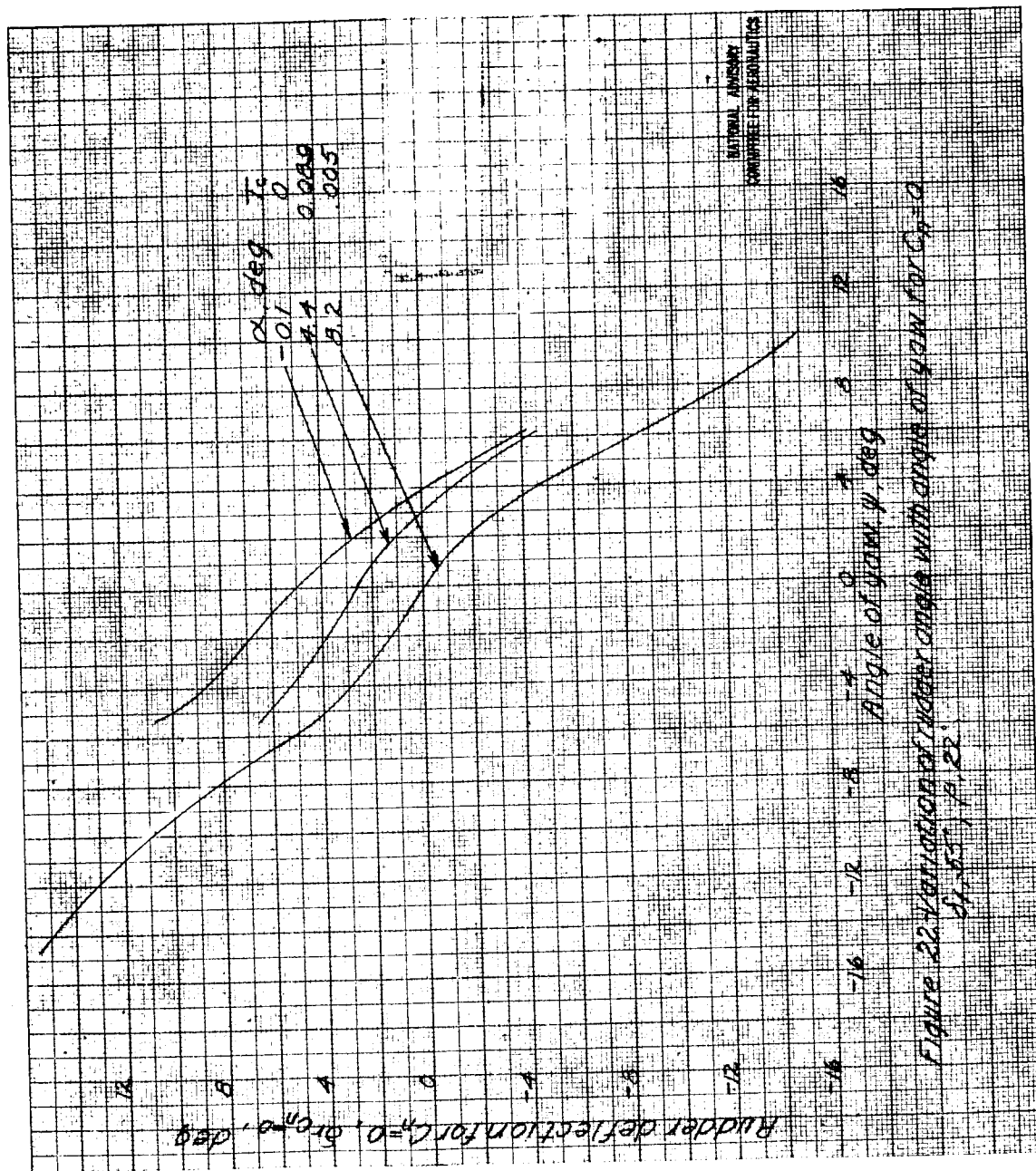


Figure 21 - Variation of rudder angle with angle of yaw  
 for  $C_D = 0$ ,  $\alpha$ , 20°;  $\beta$ , 22°



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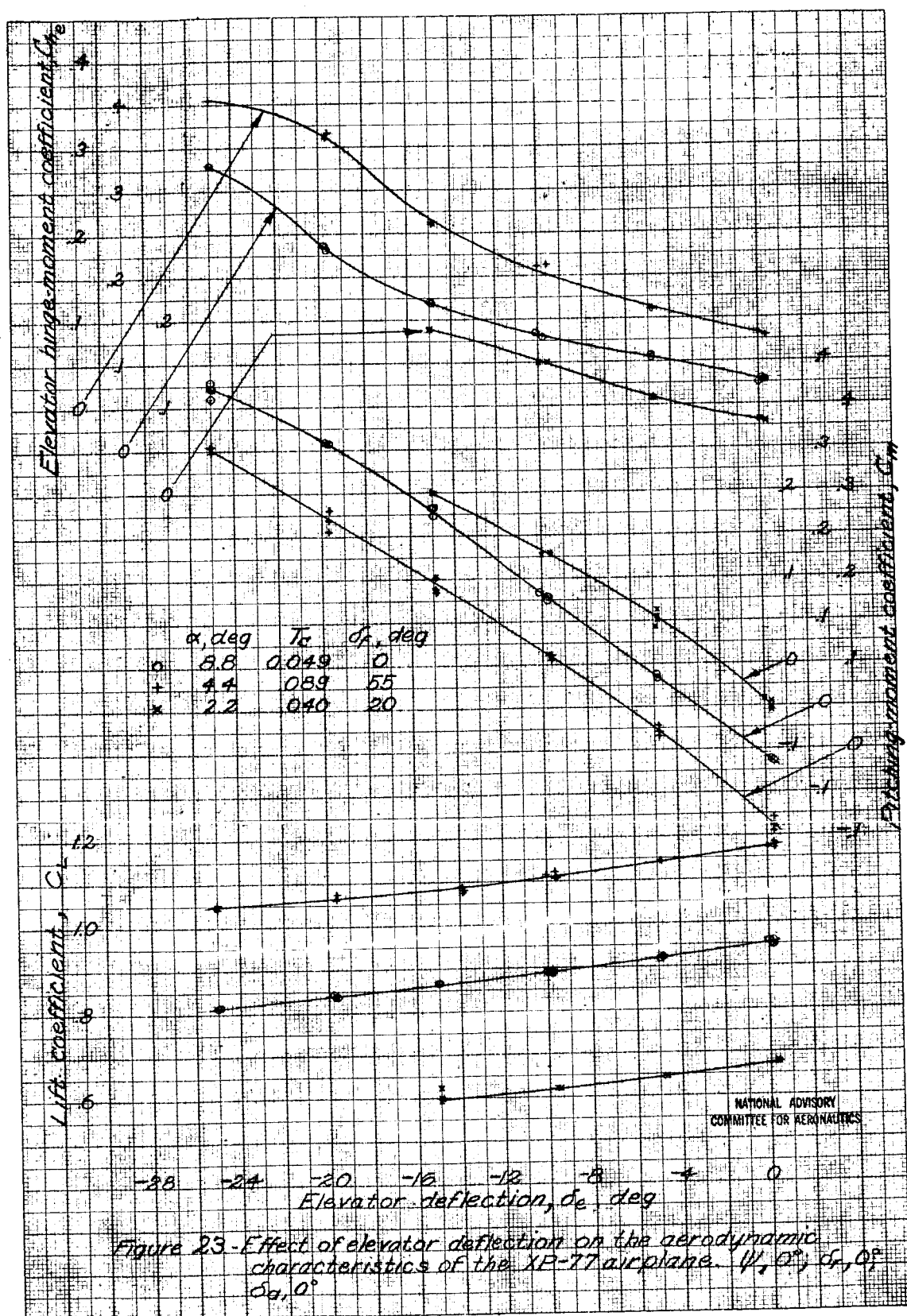
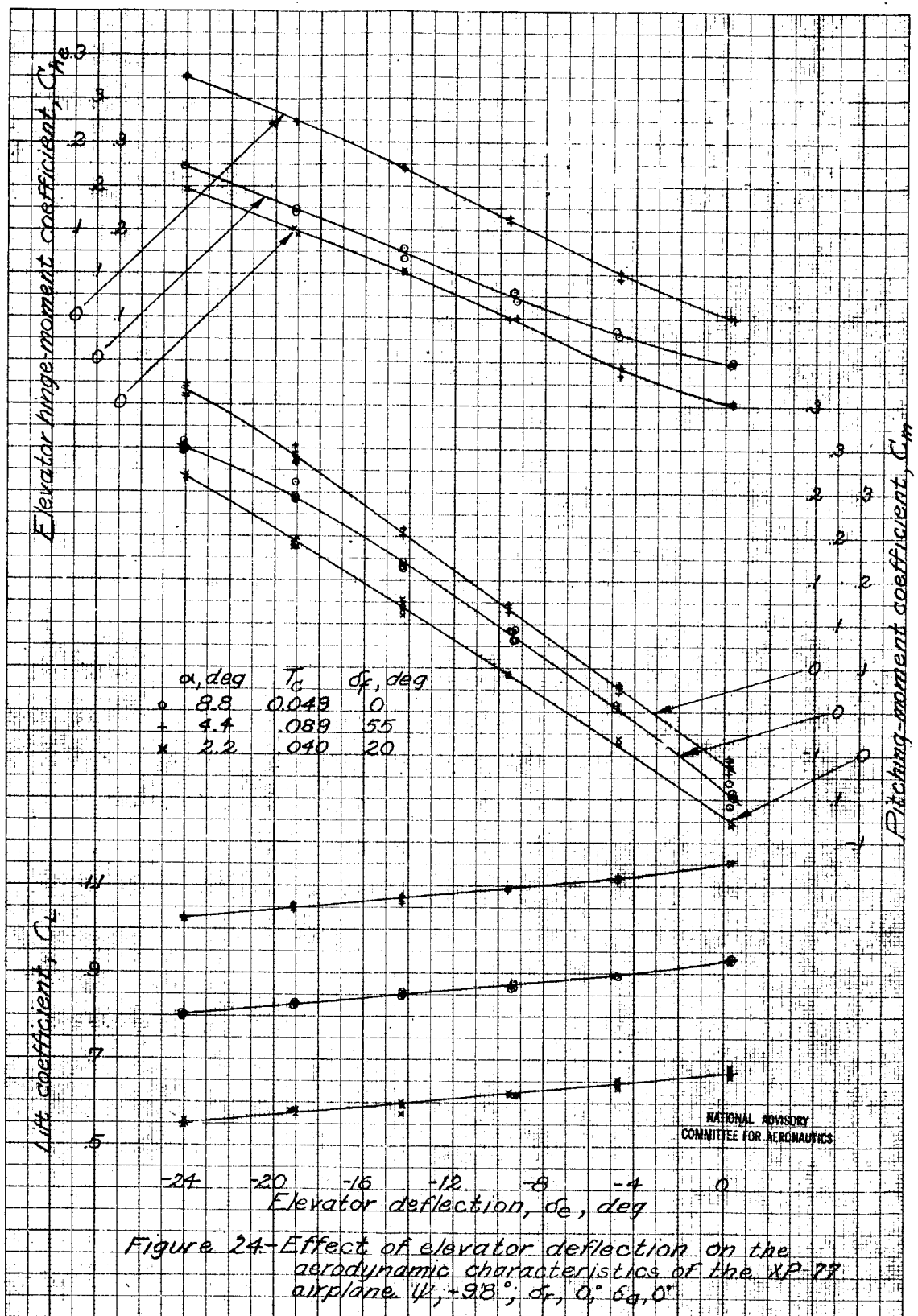
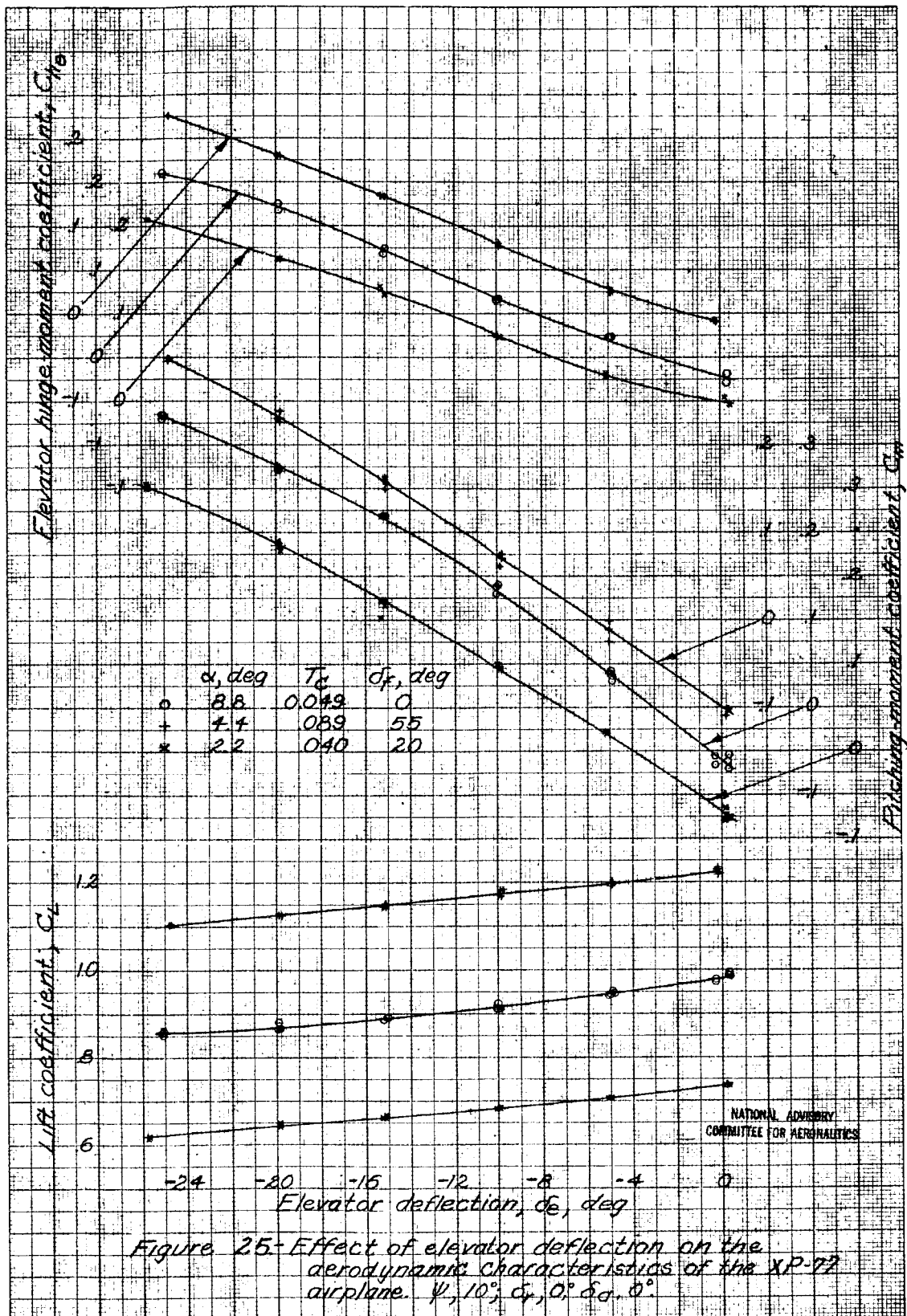
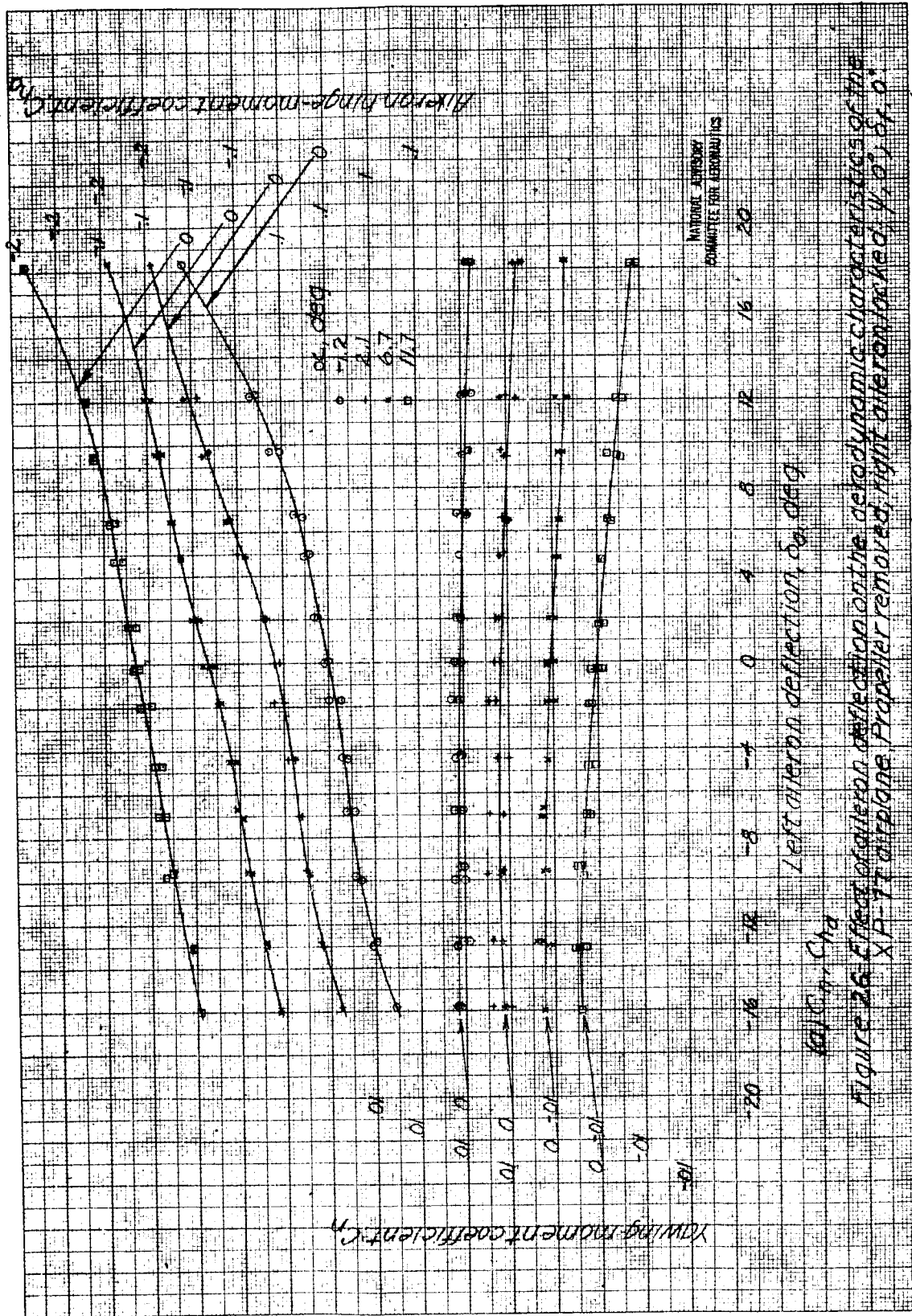


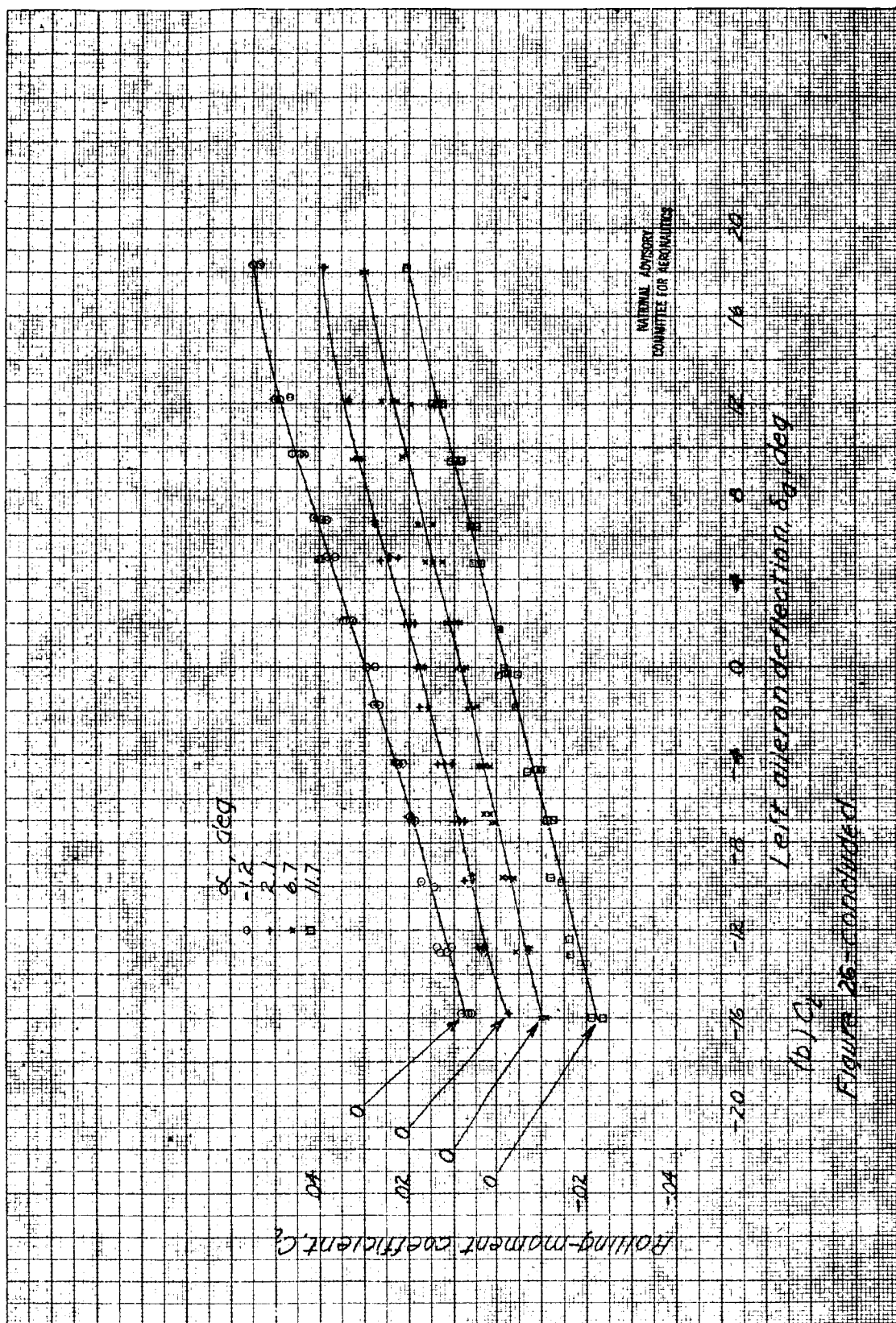
Figure 23 - Effect of elevator deflection on the aerodynamic characteristics of the XP-77 airplane.  $\psi, 0^\circ$ ;  $\delta_r, 0^\circ$ ;  $\alpha_a, 0^\circ$



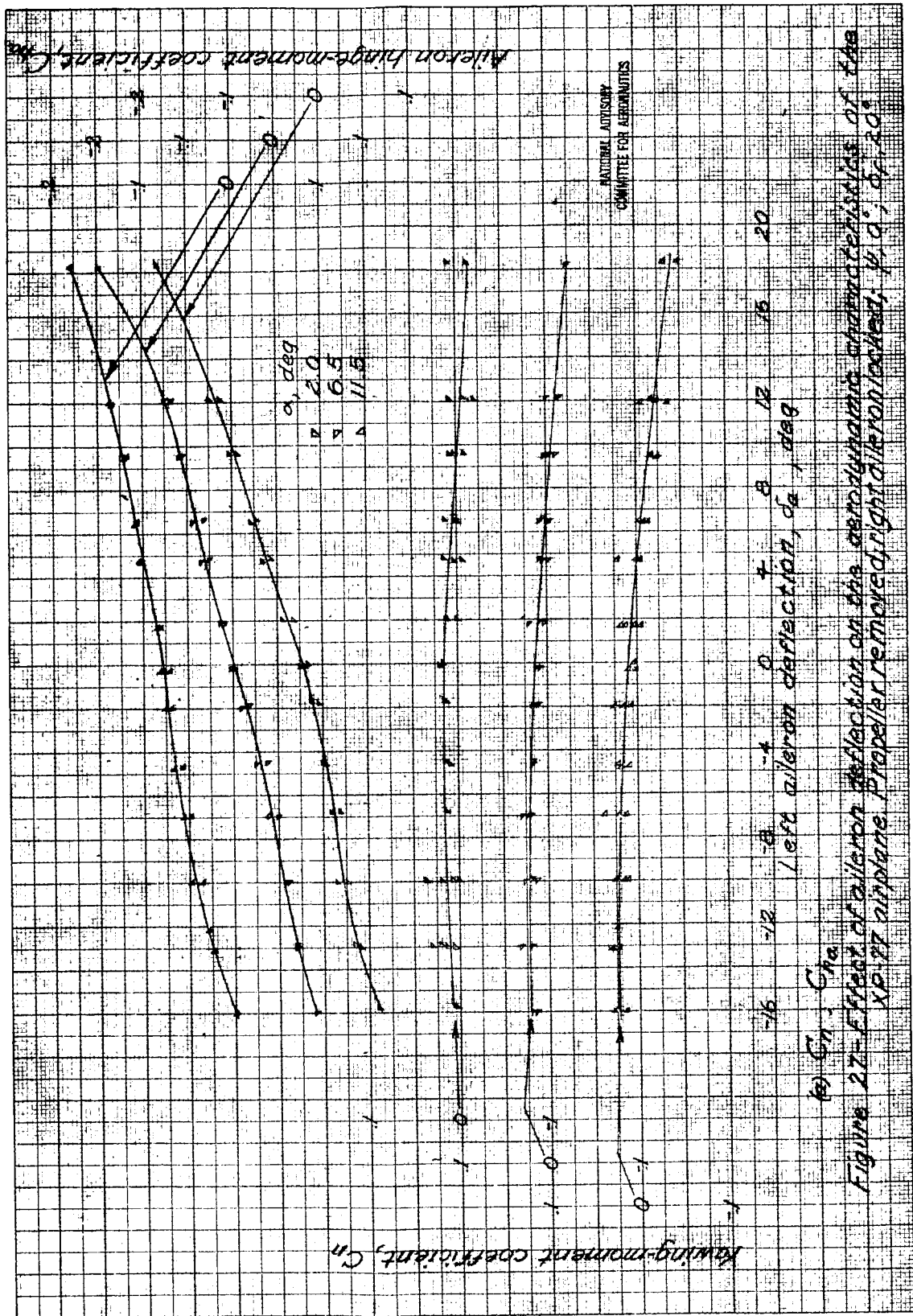


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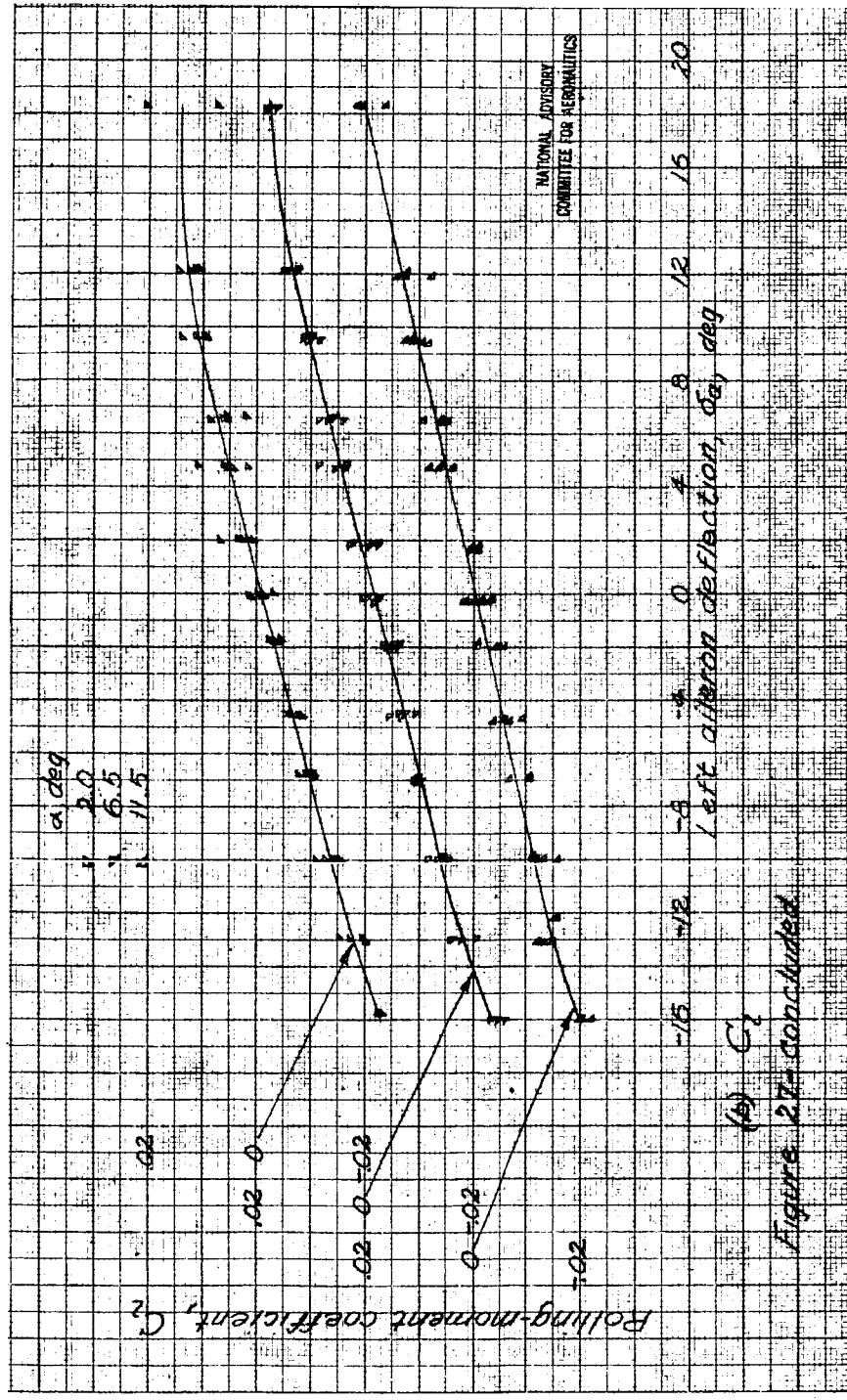


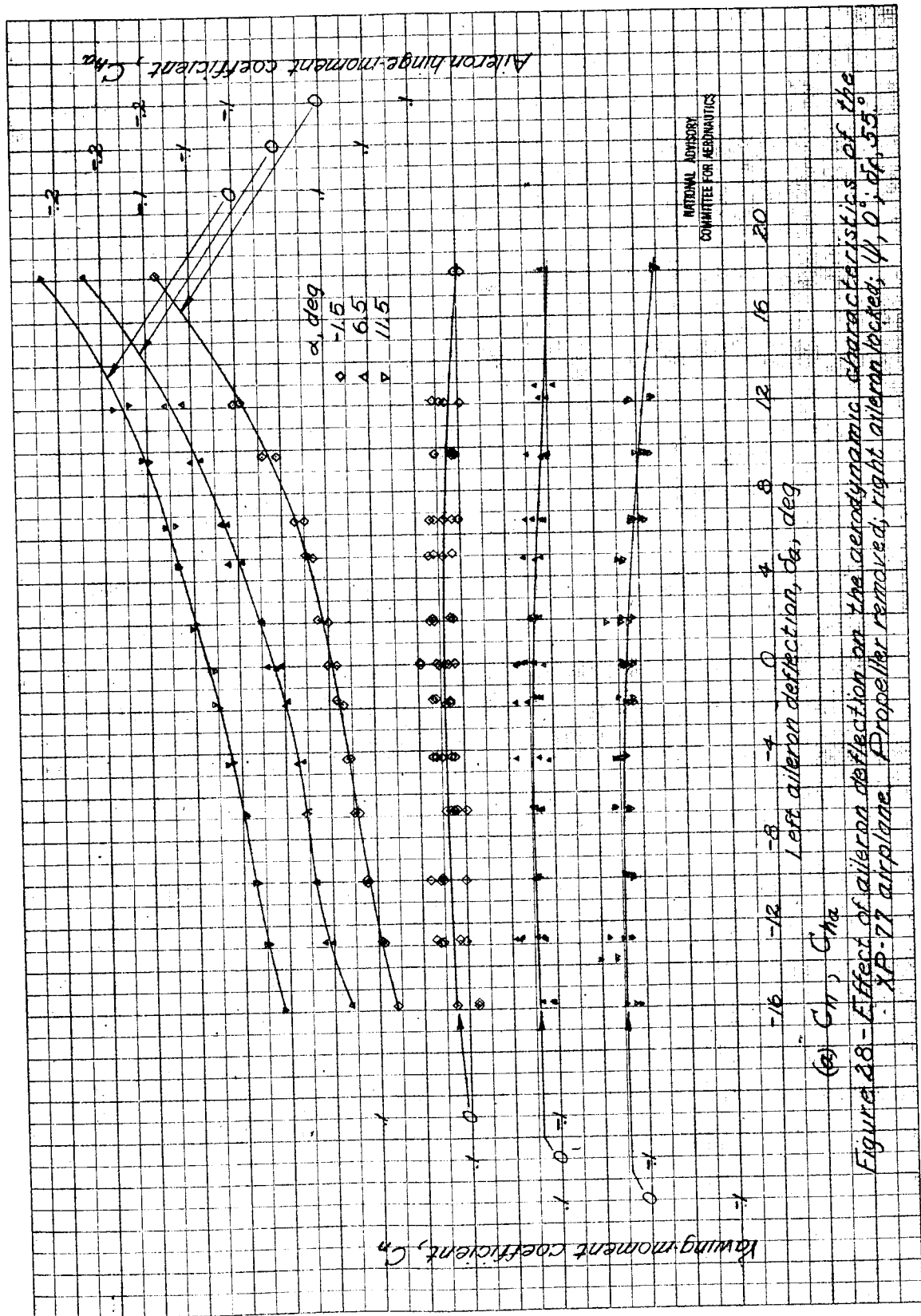






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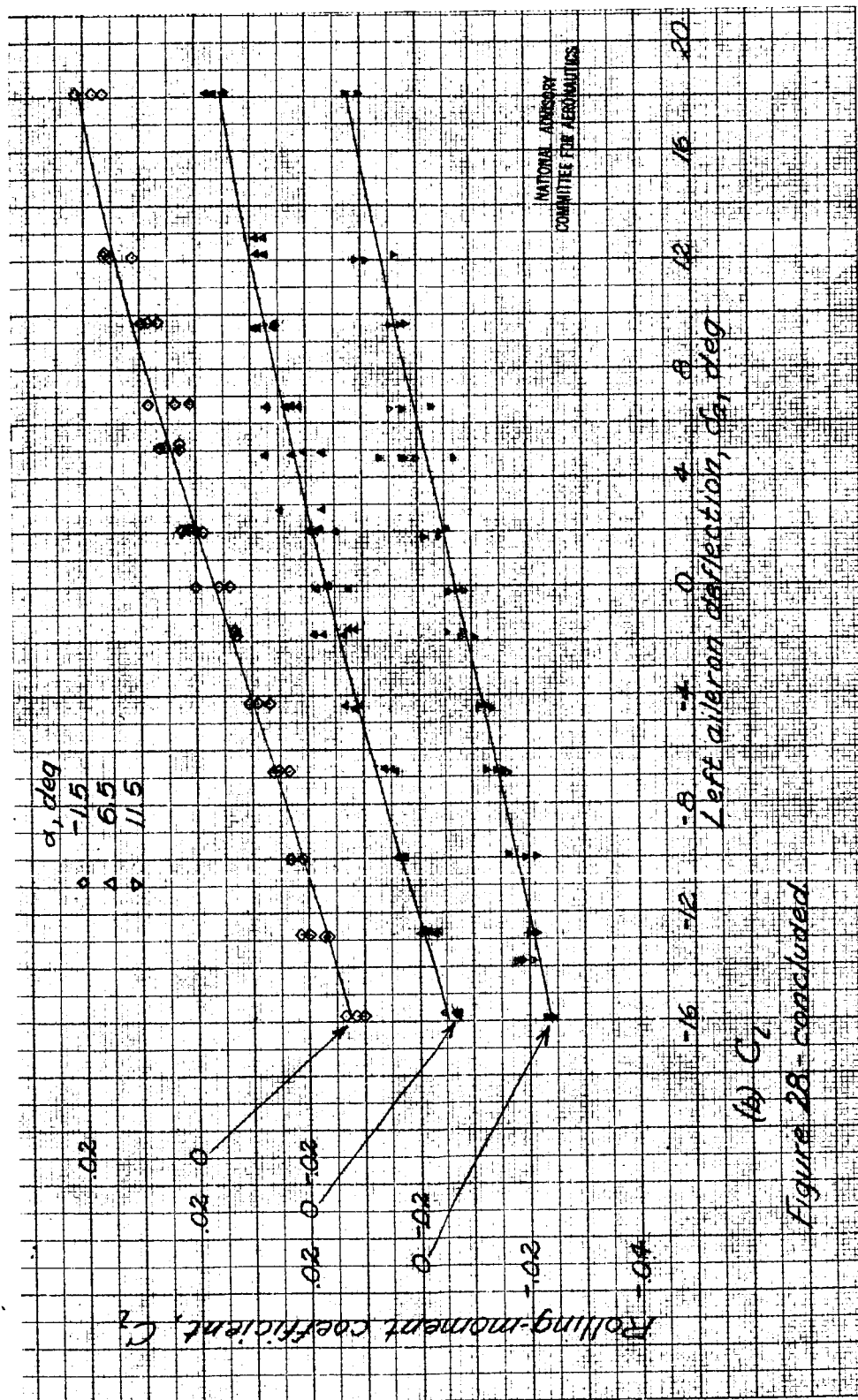
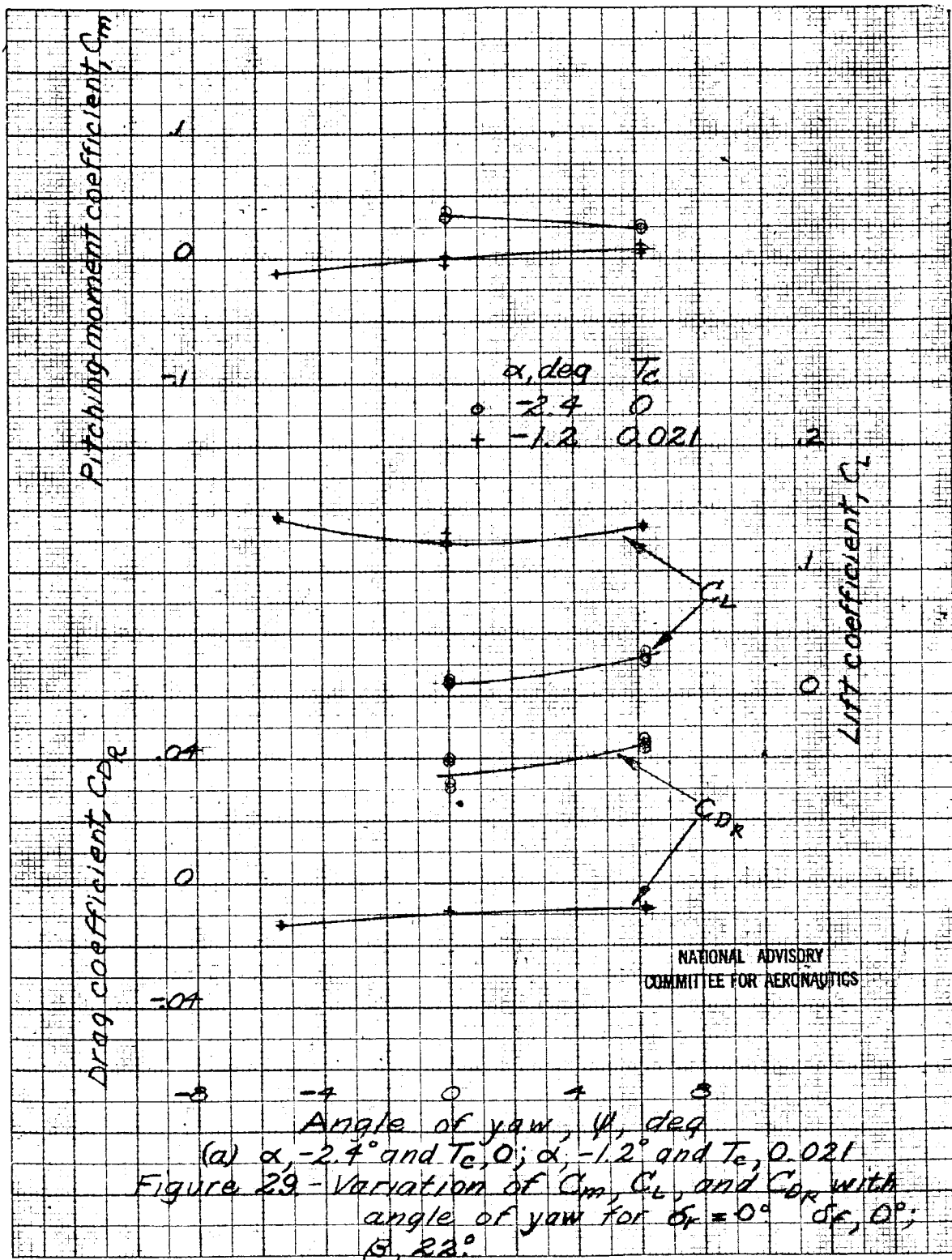
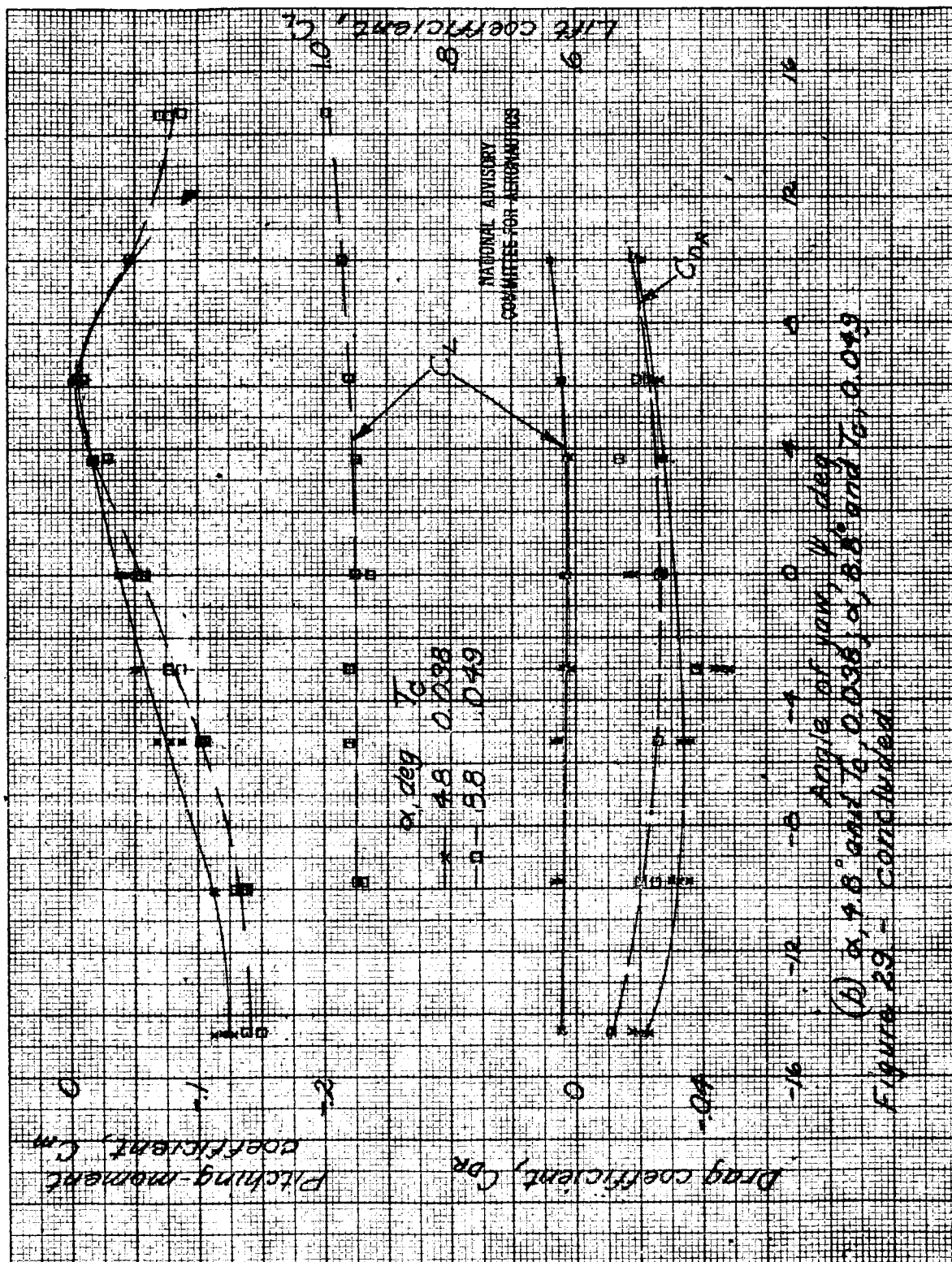
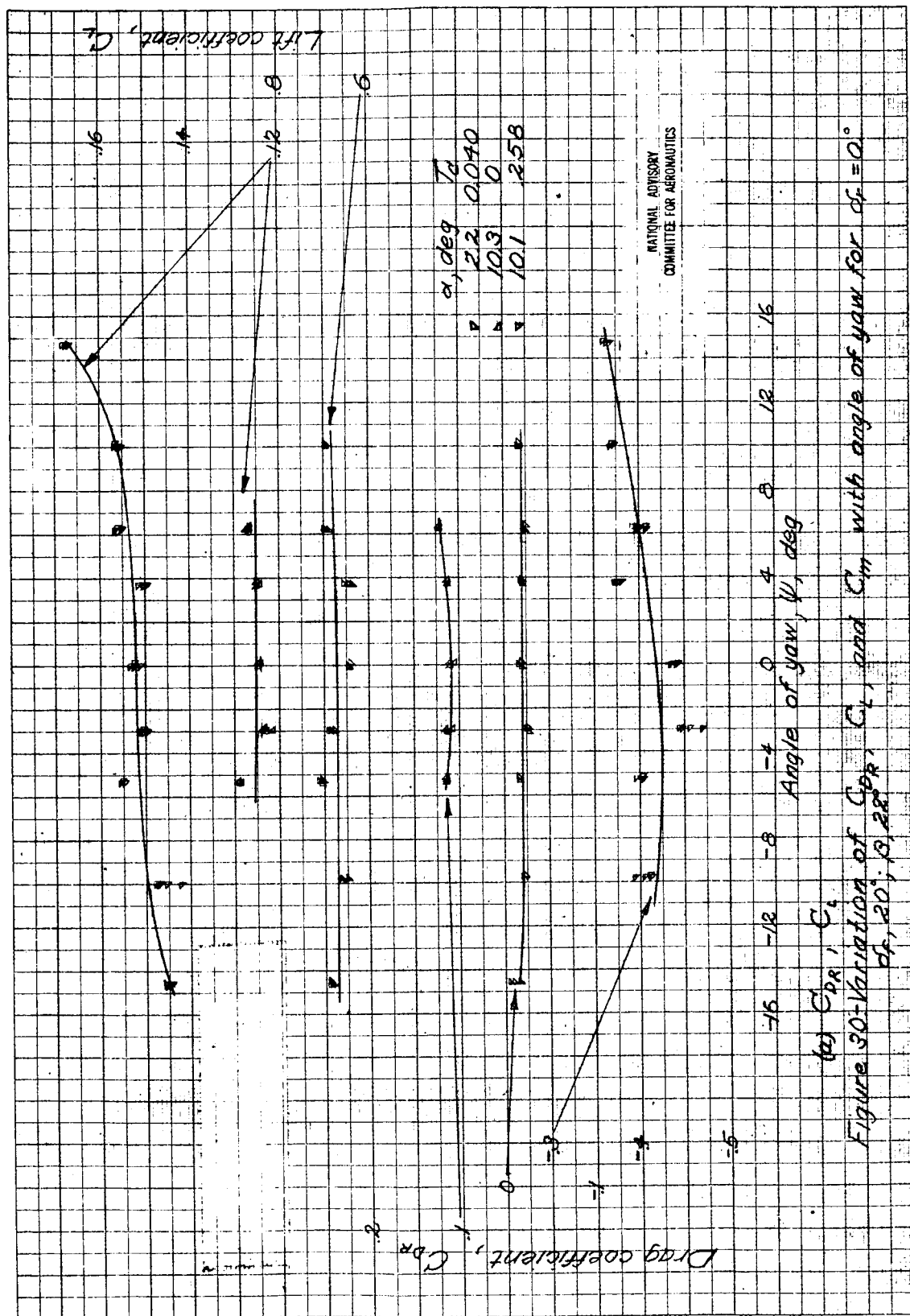


Figure 28 - concluded







Pitching-moment coefficient,  $C_m$

| $\alpha$ , deg | $T_c$ |
|----------------|-------|
| 2.2            | 0.040 |
| 10.3           | 0     |
| 10.1           | 2.58  |

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Angle of yaw,  $\mu$ , deg

(b)  $C_m$

Figure 30- concluded

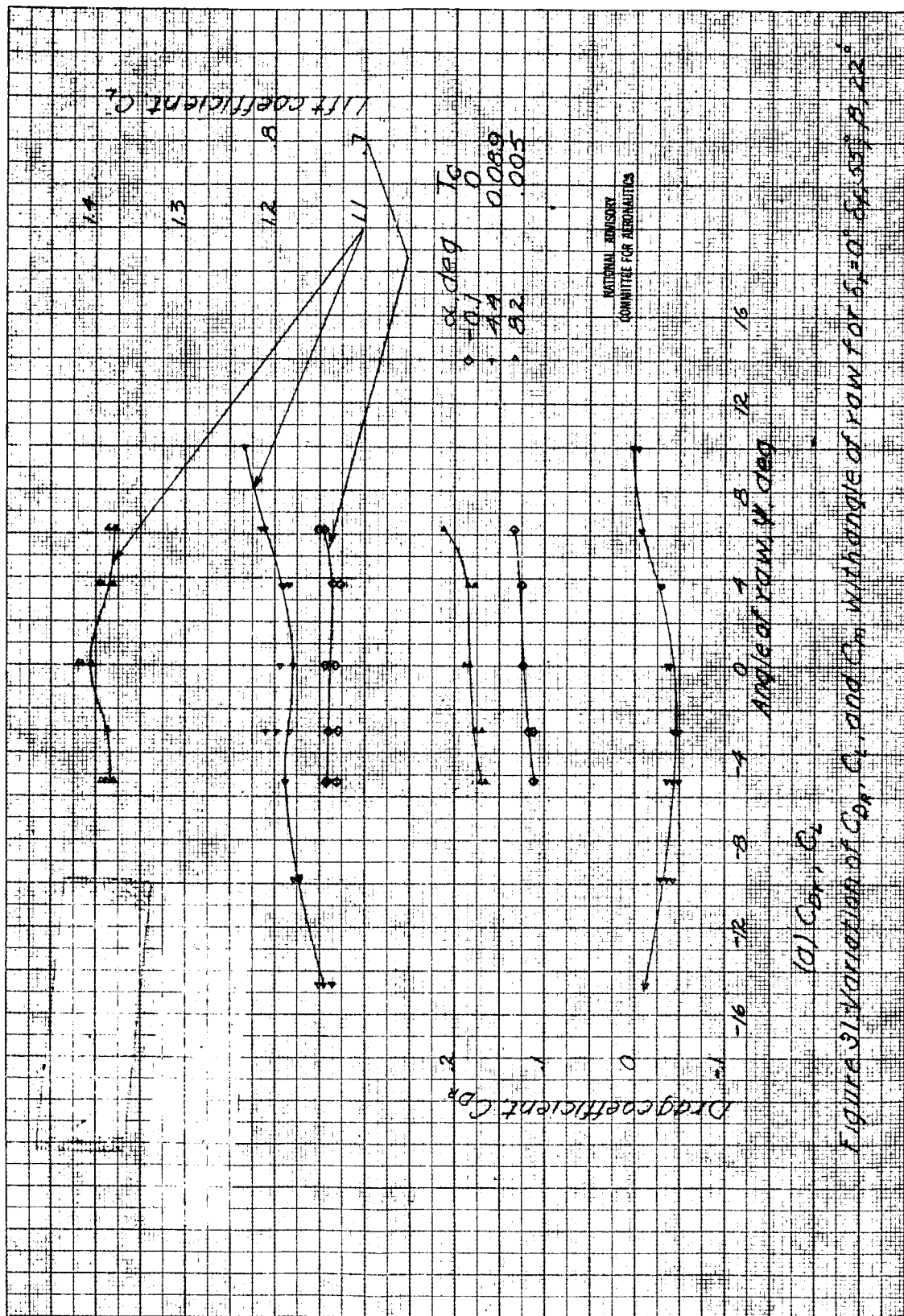
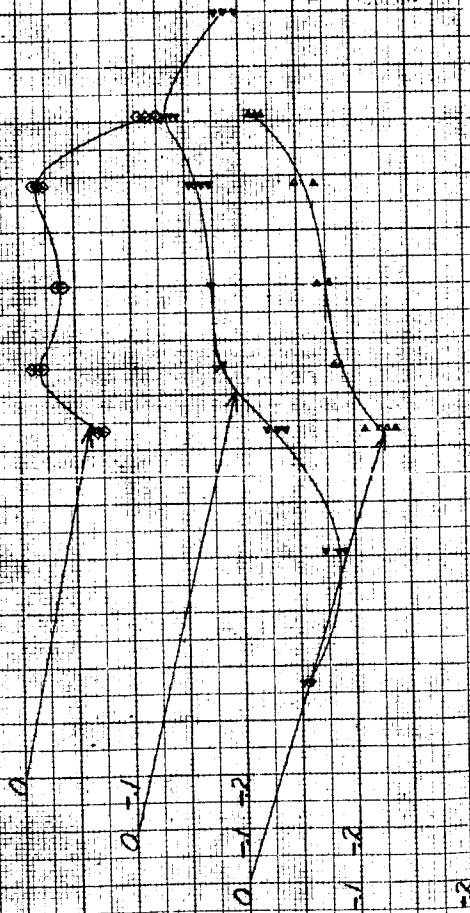


Figure 31. Variation of  $C_D$ ,  $C_L$ , and  $C_m$  with angle of row for  $Re = 0, 0.089, 0.05$ .

Pitching-moment coefficient,  $C_m$



$\alpha$ , deg

|    |        |
|----|--------|
| 0  | 0.0099 |
| -1 | 0.0099 |
| -2 | 0.0099 |
| -3 | 0.0099 |

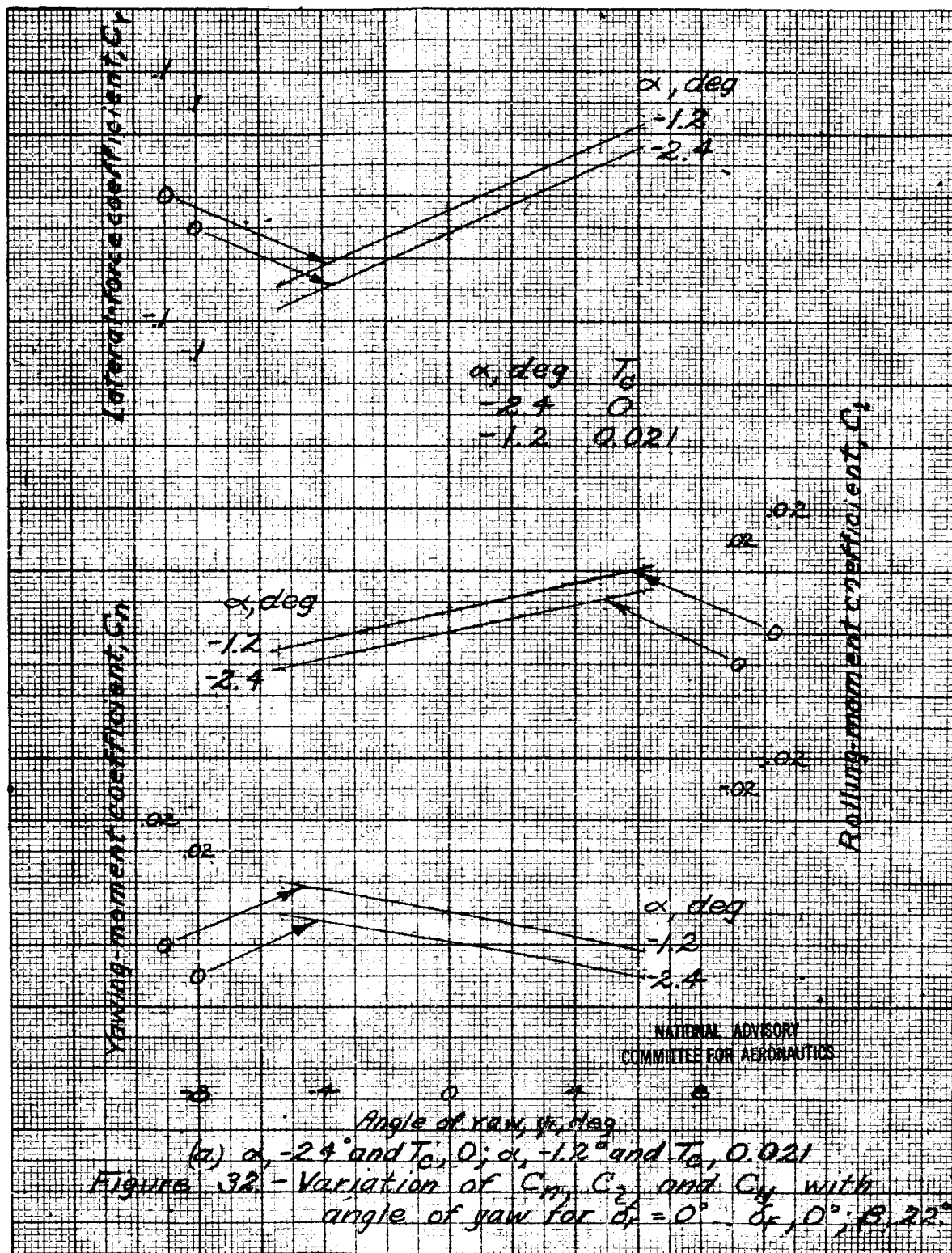
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Angle of yaw,  $\psi$  deg

(b)  $C_m$

Figure 31 - continued







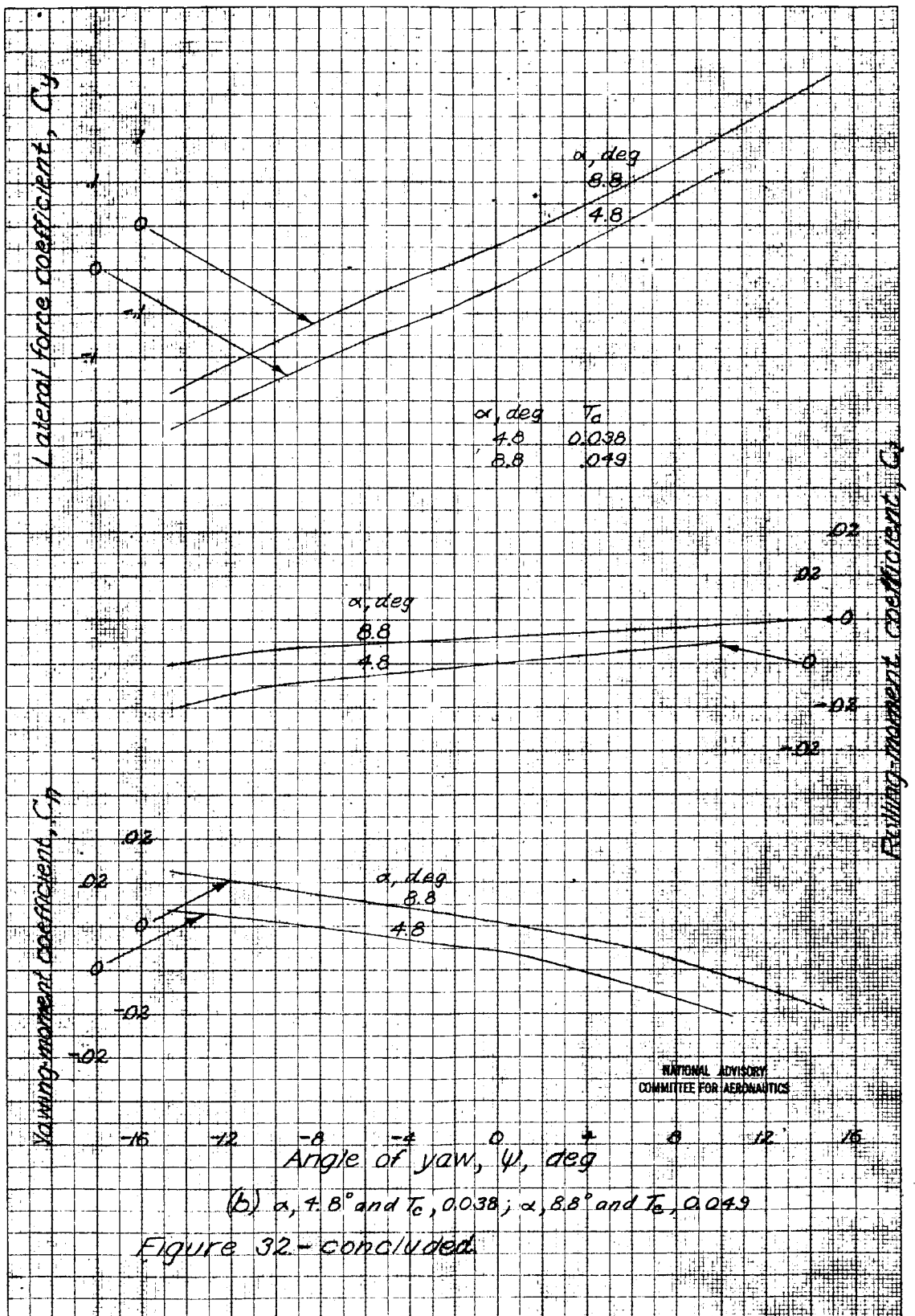
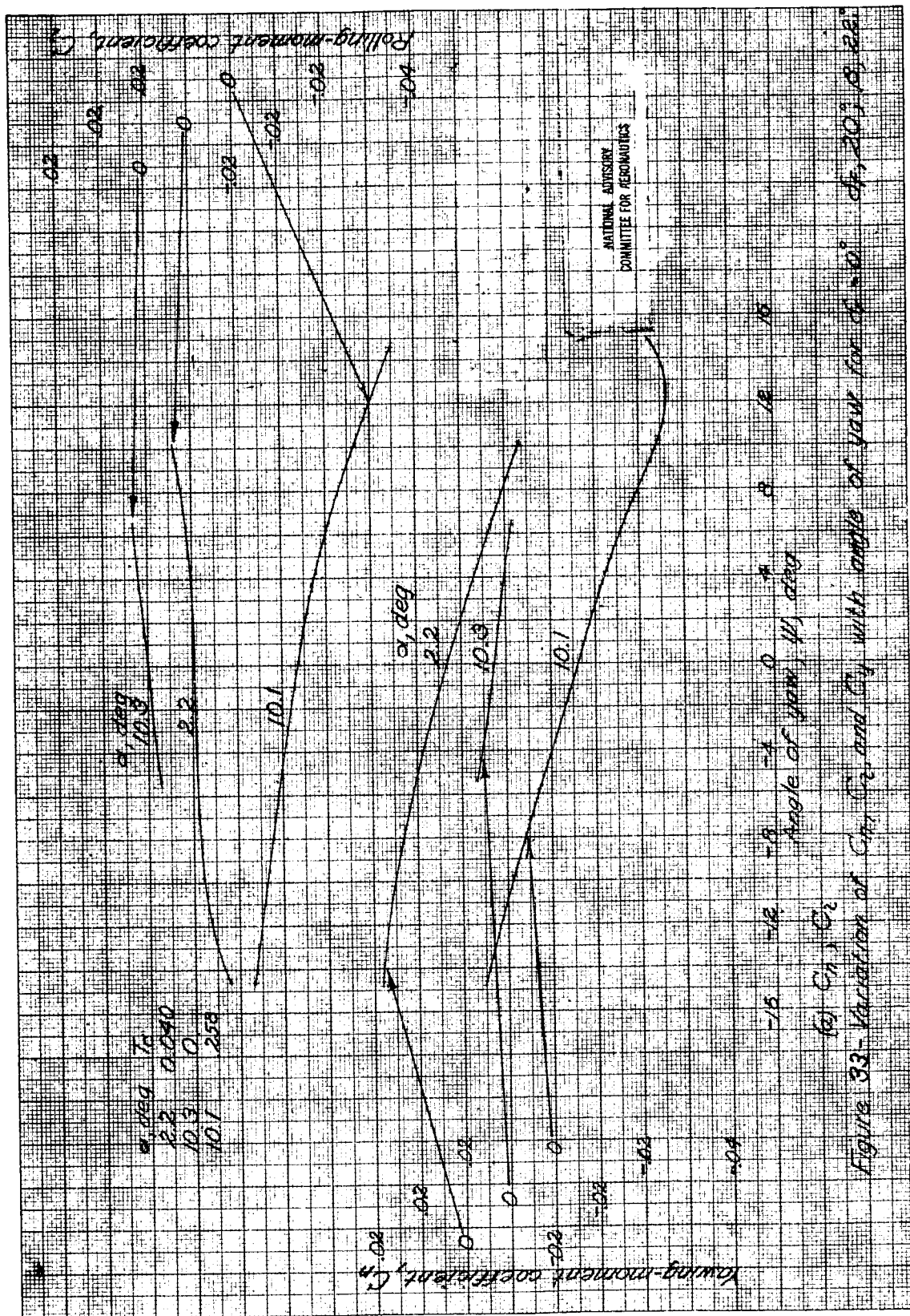
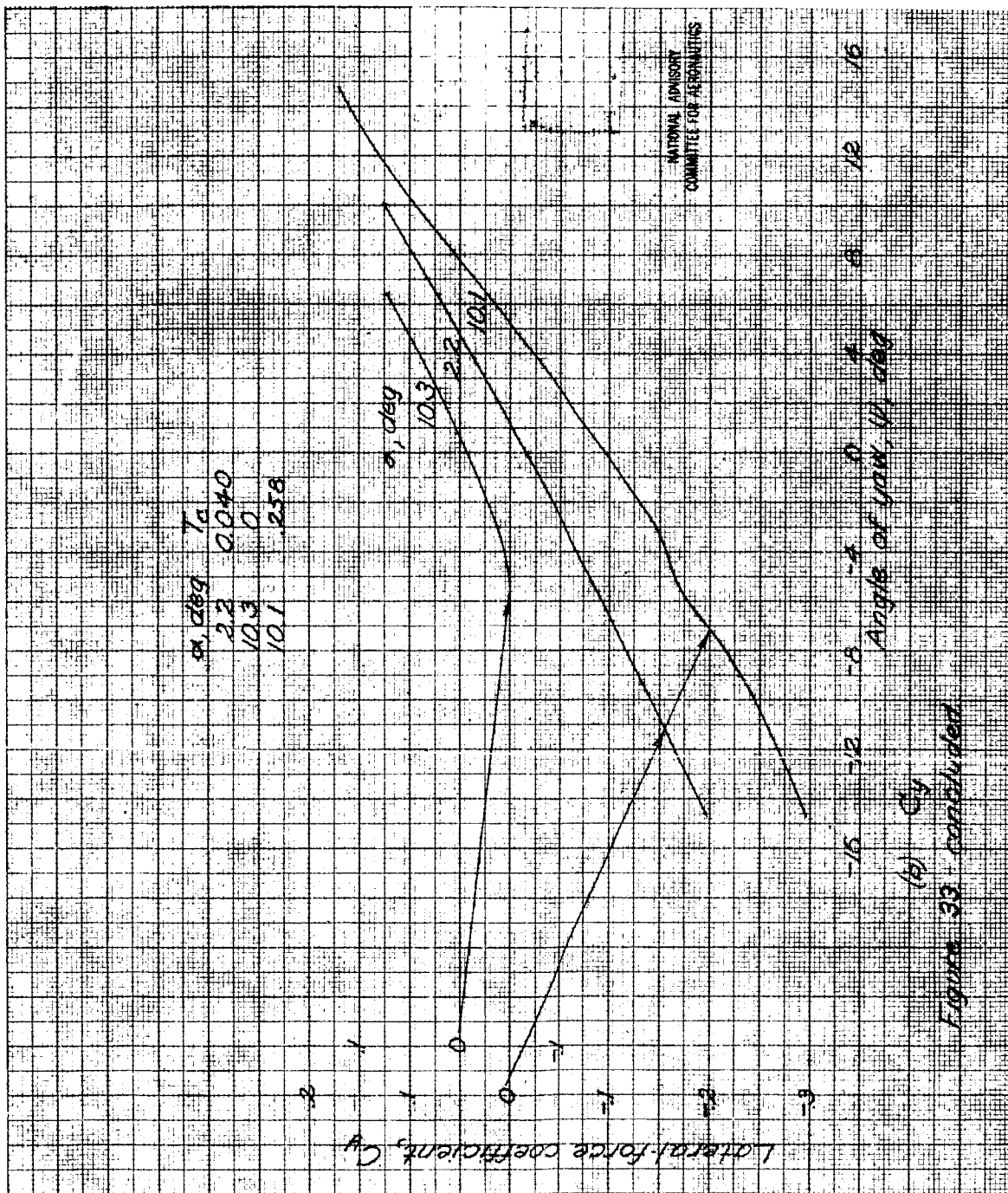


Figure 32 - concluded





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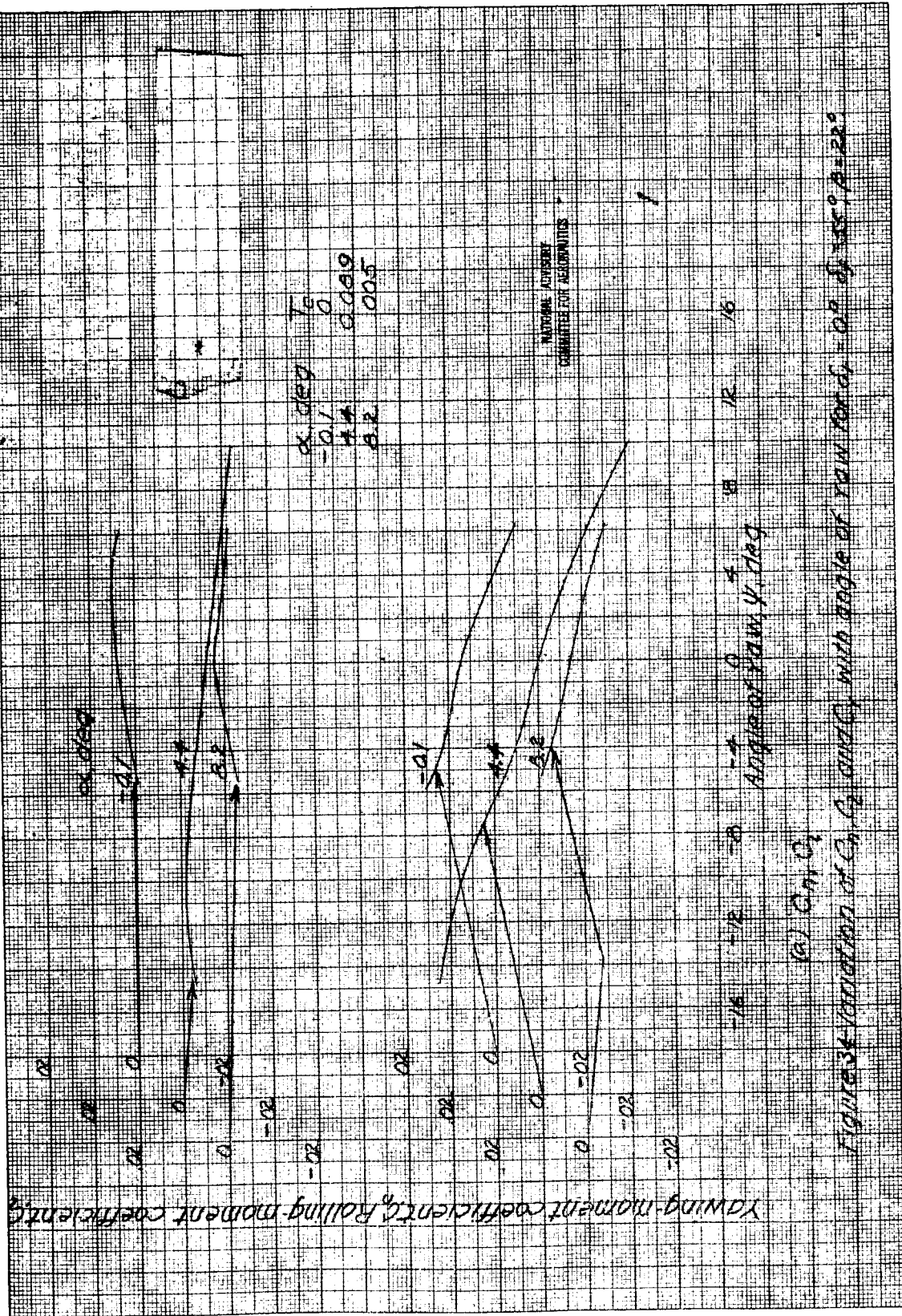
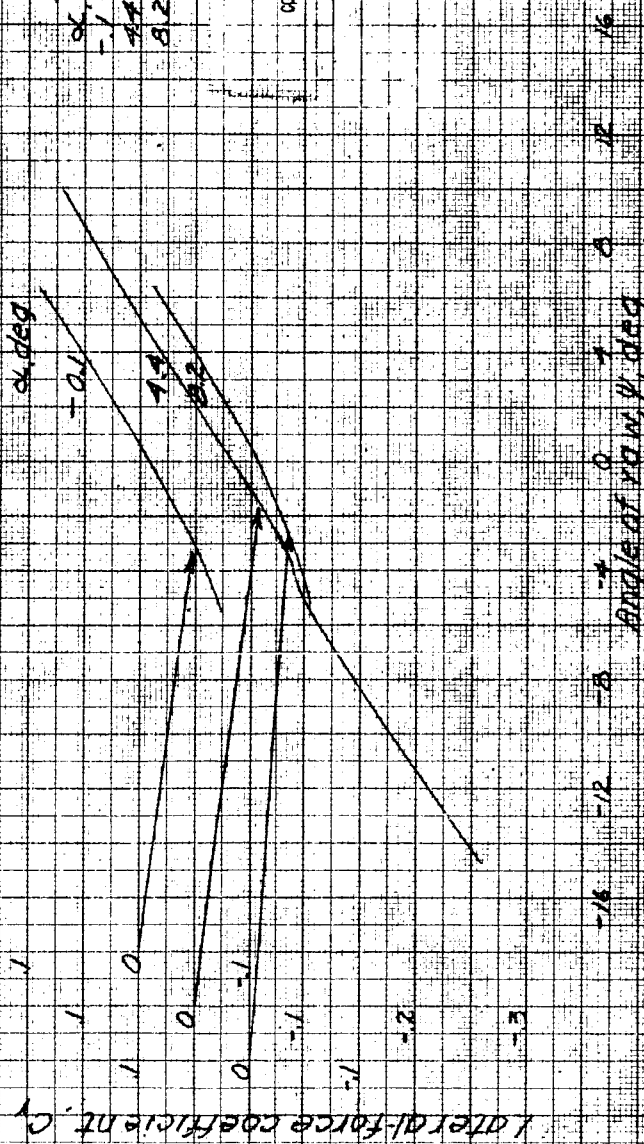


FIGURE 34-Variation of  $C_{YR}$ ,  $C_{YR}$  and  $C_{YR}$  with angle of yaw for  $\alpha = 0.2^\circ$ ,  $\beta = 0.5^\circ$ ,  $\beta = 2.2^\circ$



$\alpha, \text{deg}$   
 $-1$   
 $0$   
 $0.1$   
 $0.005$

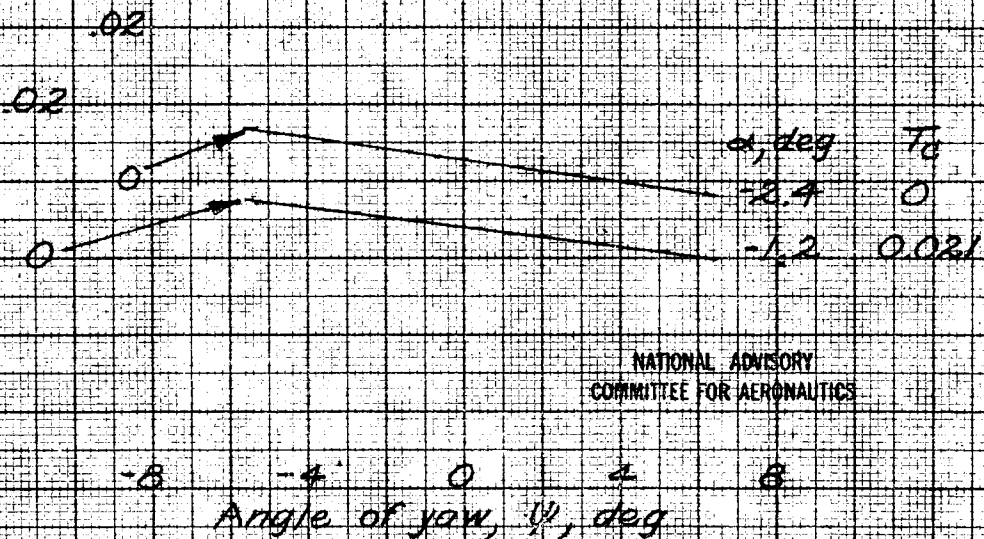
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(b)  $C_y$

Figure 34-continued



Yawing-moment coefficient  
for  $C_{Hr} = 0$ ,  $(C_{Hr})_{C_{Hr}=0}$



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(a)  $\alpha$ ,  $-2.4^\circ$  and  $T_0$ , 0;  $\alpha$ ,  $-1.2^\circ$  and  $T_0$ , 0.021

Figure 35. - Variation of yawing-moment coefficient for  $C_{Hr} = 0$  with angle of yaw.  $\alpha$ ,  $0^\circ$ ;  $\beta$ ,  $22^\circ$

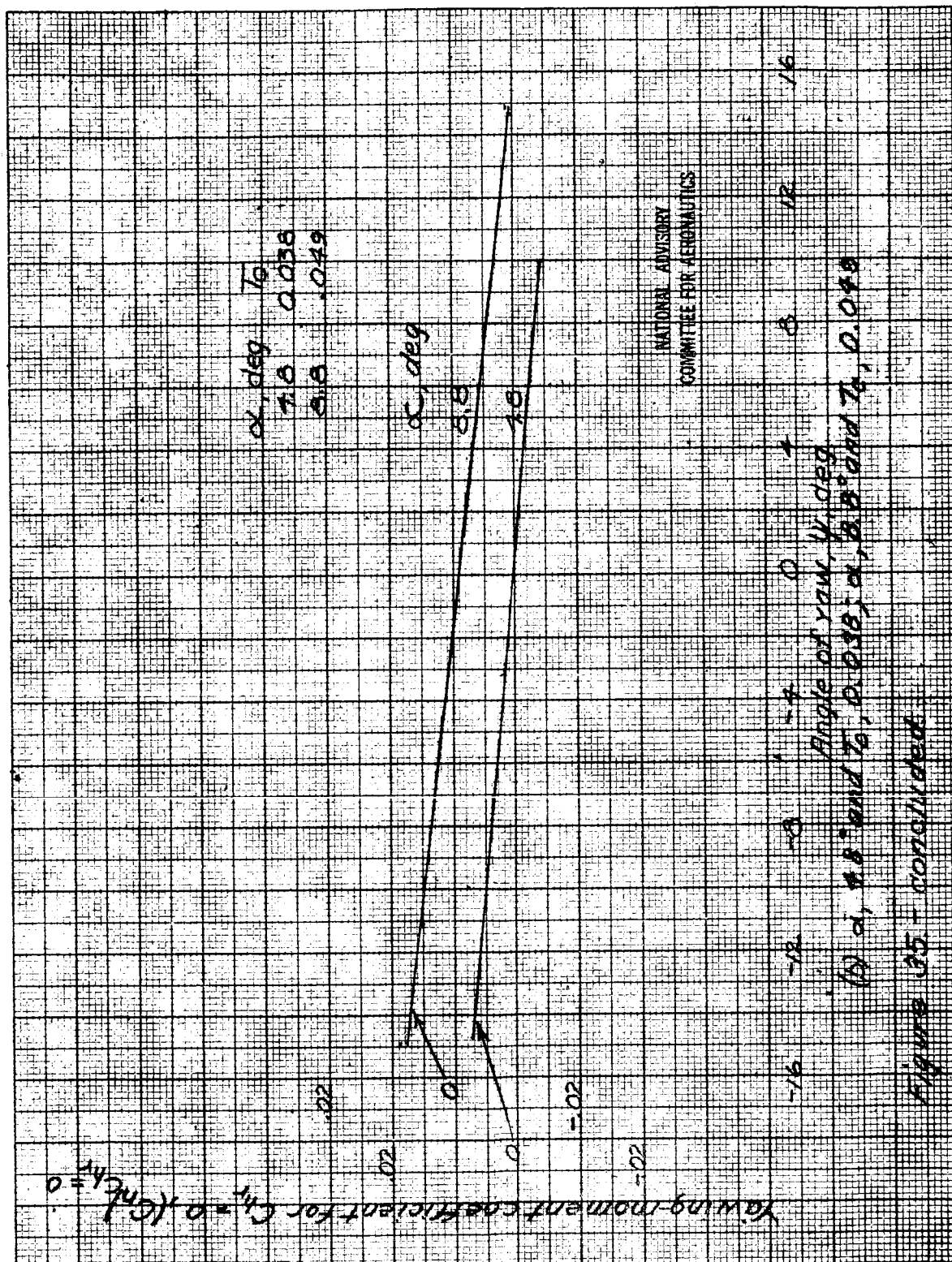


Figure 35 - concluded

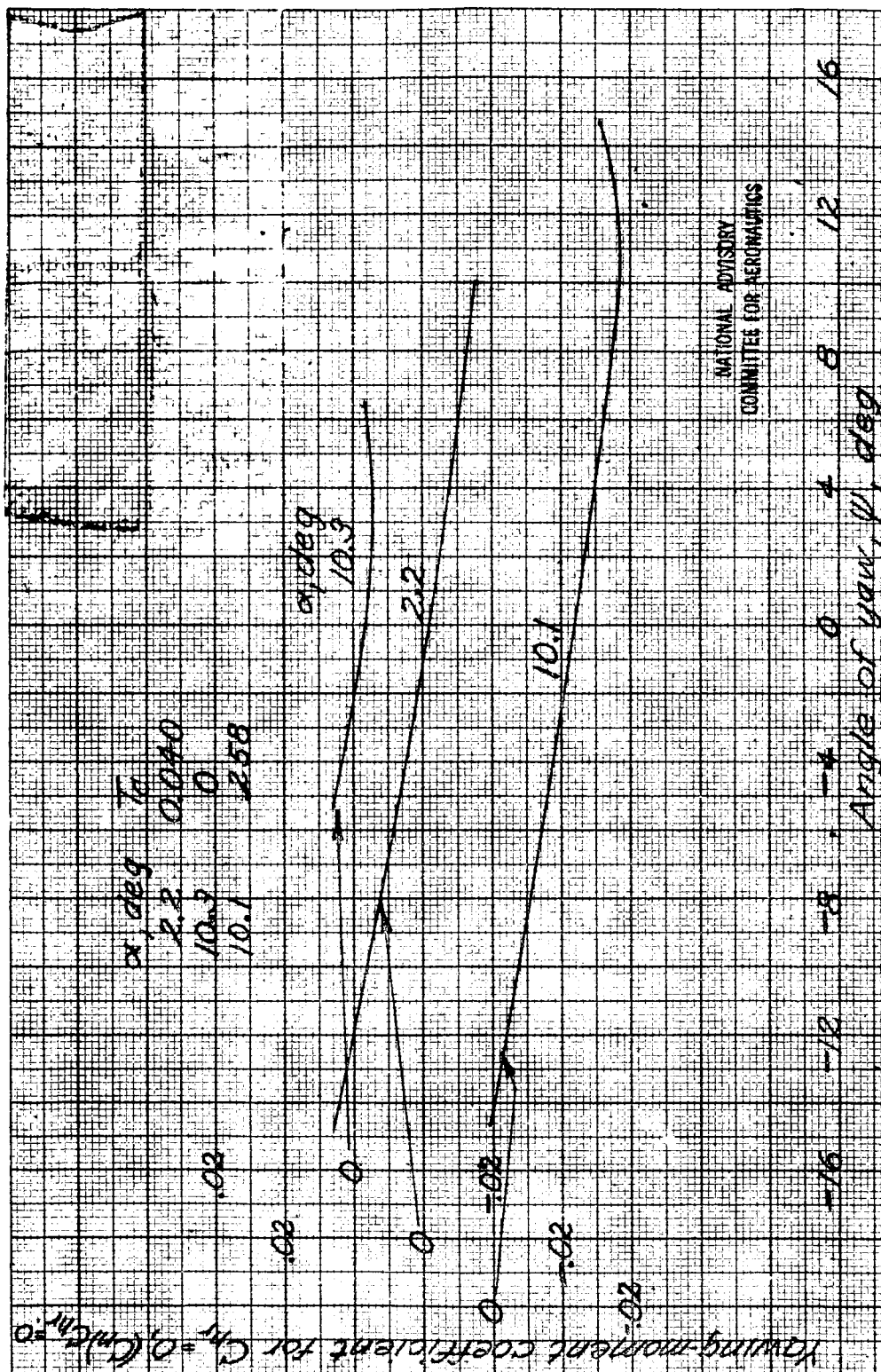
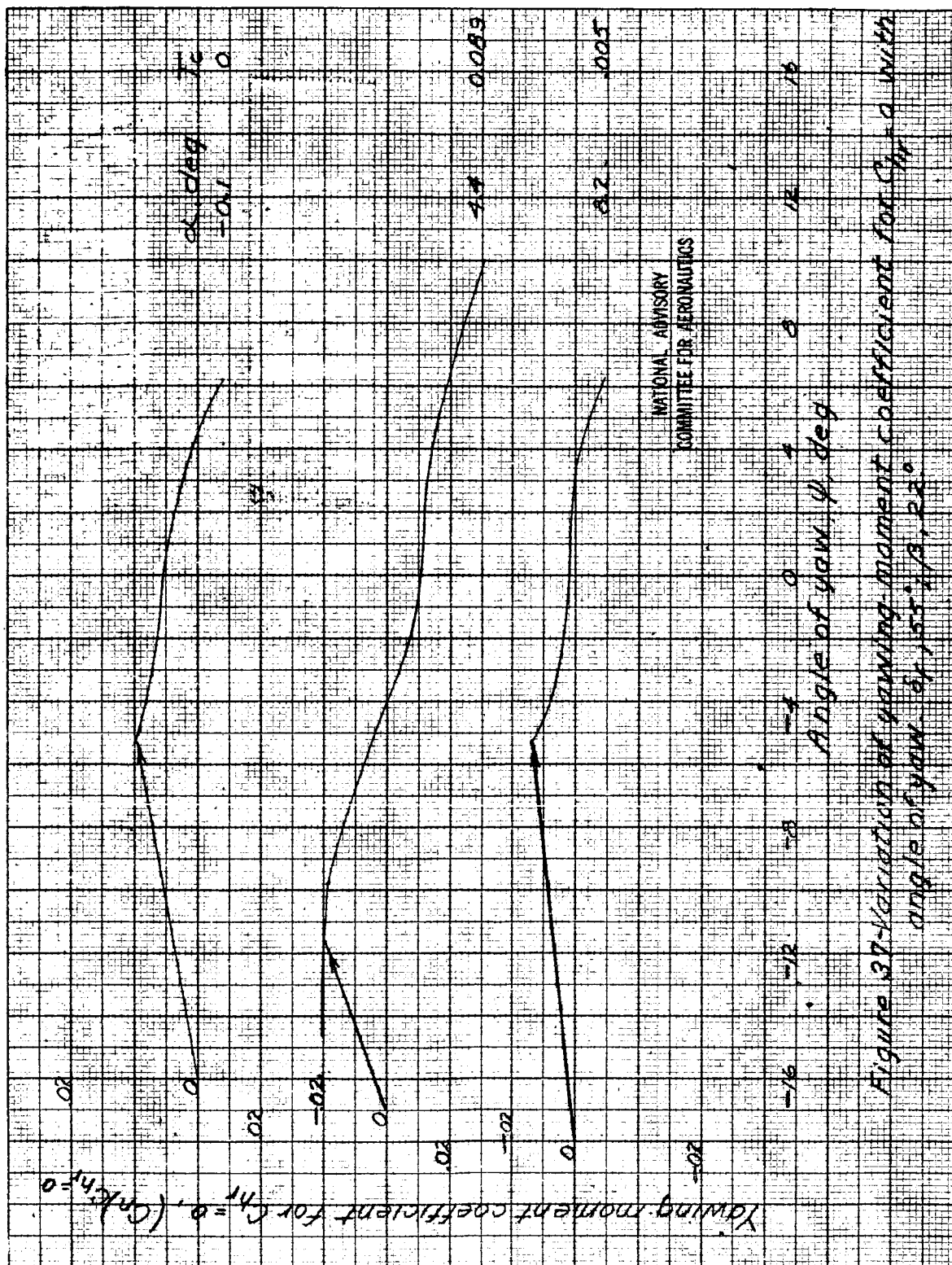


Figure 36-Variation of yawing-moment coefficient for  $C_H = 0$  with angle of yaw  $\psi$ , 20; 10, 22°





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MEMORANDUM REPORT

for the

Army Air Forces, Materiel Command

THE LATERAL FLYING QUALITIES OF THE BELL XP-77 AIRPLANE

AS ESTIMATED FROM FULL-SCALE TUNNEL TESTS

By C. J. Donlan and K. R. Czarnecki

Langley Memorial Aeronautical Laboratory  
Langley Field, Va.

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## MEMORANDUM REPORT

for the

Army Air Forces, Materiel Command

### THE LATERAL FLYING QUALITIES OF THE BELL XP-77 AIRPLANE AS ESTIMATED FROM FULL-SCALE TUNNEL TESTS

By C. J. Donlan and K. R. Czarnecki

#### INTRODUCTION

At the request of the Army Air Forces, Materiel Command, an investigation has been made to determine the lateral flying qualities of the Bell XP-77 airplane. This report contains the results of the investigation which is based on the tests made of the full-scale model of the airplane in the NACA full-scale tunnel (reference 1). These tests were planned so as to cover as completely as possible the lateral flying quality requirements for pursuit-type airplanes contracted for by the United States Army Air Forces (reference 2). Inasmuch as the requirements listed in reference 2 form the basis of the presentation which follows, a copy of this reference should be available to permit rapid cross-reference.

#### BASIS OF ANALYSIS

As the analysis is based on the results of the NACA full-scale tunnel tests reported in reference 1, attention is called to a few pertinent facts concerning the circumstances under which these tests were conducted:

1. The thrust line location on the airplane tested was that associated with the low-altitude version of the XP-77 airplane (fig. 1).
2. The propeller available for the tests was the 10.5-foot propeller associated with the high-altitude version of the XP-77 airplane.
3. The landing gear was removed for all tests.

These circumstances prevented the simultaneous duplication of the thrust and the slipstream characteristics associated with the specific flight operation of either the low- or high-altitude version of the airplane. The quantitative results included in this report therefore cannot be associated specifically with either the low- or high-altitude version of the XP-77 airplane but may be taken as rather characteristic of the airplane in general.

All the calculations in this report pertain to the sea-level operating conditions and are referred to a center of gravity located at 21.66 percent of the mean aerodynamic chord and 1.80 percent of the mean aerodynamic chord below the thrust axis of the low-altitude airplane.

#### SYMBOLS

|          |                                                                       |
|----------|-----------------------------------------------------------------------|
| L        | lift                                                                  |
| D        | drag                                                                  |
| S        | wing area                                                             |
| b        | wing span                                                             |
| V        | velocity                                                              |
| $\rho$   | mass density of air                                                   |
| $\rho_0$ | mass density of air at sea level                                      |
| q        | dynamic pressure $\left(\frac{1}{2}\rho v^2\right)$                   |
| $V_i$    | correct indicated airspeed $\left(v\sqrt{\frac{\rho}{\rho_0}}\right)$ |
| p        | rolling velocity                                                      |
| r        | yawing velocity                                                       |
| $\phi$   | angle of bank                                                         |
| $\beta$  | angle of sideslip                                                     |
| $\alpha$ | airplane angle of attack (thrust line)                                |

|                   |                                                                                                          |                                                                             |
|-------------------|----------------------------------------------------------------------------------------------------------|-----------------------------------------------------------------------------|
| $\Delta \delta_a$ | change in total aileron deflection from trim                                                             |                                                                             |
| $\delta_f$        | landing flap deflection                                                                                  |                                                                             |
| $C_l$             | rolling-moment coefficient about center of gravity<br>$\left( \frac{\text{rolling moment}}{qSb} \right)$ |                                                                             |
| $C_{l_p}$         | rolling-moment coefficient due to rolling                                                                | $\left[ \frac{\partial C_l}{\partial \left( \frac{pb}{2v} \right)} \right]$ |
| t                 | time                                                                                                     |                                                                             |

## PRESENTATION OF RESULTS

In order to permit rapid cross-reference the results are presented in a manner parallel to that employed in reference 2 for listing the specifications for satisfactory lateral flying qualities. Each characteristic examined is identified by a paragraph index number that is directly associated with the corresponding specification in reference 2.

## RESULTS AND DISCUSSION

### E-2. Lateral and Directional Stability and Control.

#### E-2a. Dynamic Lateral and Directional Stability.

E-2a(1). Spiral Mode. The spiral divergence would appear to be mild when the airplane is trimmed for straight and level flight at the speed for maximum L/D at the design gross weight of the airplane. From computations based on a design gross weight of 3500 pounds and an indicated airspeed of 149 miles per hour it appears that the initial rate of divergence from the steady level-flight condition is such that the angle of bank approximately doubles itself every 18 seconds.

#### E-2a(2)(d). Short Period Oscillation.

(1) Calculations based on the method of reference 3 indicate that the control-free lateral

oscillation should damp to one-half amplitude in about one cycle at indicated airspeed of 149 miles per hour (near the speed for maximum L/D) at the design gross weight of 3500 pounds.

(2) Unless the rudder is mass overbalanced the control-surface damping characteristics should be satisfactory for all specified flight conditions.

#### E-2b. Static Lateral and Directional Stability.

##### E-2b(1). Yawing Moment Due to Sideslip.

(a) Sideslip Due to Aileron Deflection. A theoretical time history of the rolling, yawing, and sideslipping motions of the XP-77 airplane following an instantaneous abrupt full-right deflection of the ailerons is given in figure 3 for a low-speed, level-flight condition with flaps retracted. The angle of sideslip generated at the peak value of the rolling velocity is about  $4^\circ$ . If the airplane had been constrained in sideslip, the theoretical maximum rate of rolling for this condition would have been about  $71^\circ$  per second. Thus, it is seen that the effect of sideslip was to reduce the peak value of the rolling velocity from  $71^\circ$  per second ( $\frac{pb}{2v} = 0.097$ ) to about  $65^\circ$  per second ( $\frac{pb}{2v} = 0.088$ ) - a reduction of only  $6^\circ$  per second. It is realized that the results are theoretical and that the development of wing twist during the actual flight maneuver may result in a somewhat lower rolling velocity. The analysis does indicate, nevertheless, that the yawing moment due to sideslip will be adequate to restrict the yaw due to the ailerons to a point necessary to maintain good aileron control.

The theoretical maximum angle of sideslip developed during the maneuver is about  $28^\circ$  and exceeds the specified limit of  $20^\circ$ . The calculated value occurs after a considerable angle of bank has developed and is therefore considered to be rather large owing to approximations made concerning the relationship between the angles of bank and sideslip in the equations of motion from which the solutions in figure 2 were obtained (reference 4).

A similar analysis for the low-speed level-flight condition with flaps deflected  $55^\circ$  was not made inasmuch as the wind-tunnel tests revealed the airplane to be laterally unstable in this condition. (See paragraph E-2b(2)(a)).

(b) Sideslip Due to Rudder Deflection. The estimated steady-sideslip characteristics of the XP-77 airplane for four flight conditions are presented in figures 4 to 7. The flight conditions associated with figures 4 and 5 roughly correspond to the two low-speed conditions listed under this requirement in the specifications. The flight conditions associated with figures 6 and 7 correspond to the approximate attitudes for the speed for best climb with the flaps retracted (fig. 6) and with the flaps deflected  $20^\circ$  (fig. 7) except that level-flight power was used.

For each configuration investigated the sideslip due to rudder deflection is such that the rudder deflection from its trim position at zero sideslip produces sideslip in the correct direction for angles of sideslip up to  $15^\circ$ . Unfortunately test data are lacking at angles of sideslip above  $15^\circ$ .

Attention is called to the irregularities in the curve relating the angle of sideslip and rudder deflection at very small angles of sideslip. In view of the fact that the airplane may trim in this range of sideslip angles during normal flight, these irregularities may be reflected in an undesirably low degree of directional stability at these angles.

(c) Rudder Free Requirement. The pedal forces associated with the steady-sideslip computations (figs. 4-7) indicate that the rudder will tend to return to its initial trim position when it is released in steady sideslip maneuvers up to at least  $15^\circ$  sideslip. The force curves do not pass through zero at the angle of sideslip for zero angle of bank because the data from which these results were evaluated was obtained at a single trim tab setting. Inasmuch as test data are lacking at angles of sideslip beyond  $15^\circ$ , the conditions for rudder lock could not be investigated. Rudder lock will occur if the pedal force decreases to zero and changes direction. Under these conditions the airplane would be directionally unstable with the rudder free.

#### E-2b(2). Rolling Moment Due to Sideslip.

(a) Aileron Deflection and Control Force in Sideslip. As judged from the variation of aileron deflection and control force with angle of sideslip, the rolling

moment due to sideslip (aileron fixed and free) is destabilizing for the low-speed condition with the flaps deflected  $55^{\circ}$  (fig. 4). For this configuration the instability amounts to an effective negative dihedral angle of  $3^{\circ}$ .

For the configurations with flaps retracted (figs. 5 and 6) the rolling moment due to sideslip is stabilizing. The rolling stability for these conditions is equivalent to about  $3^{\circ}$  effective dihedral.

For the configuration with flaps deflected  $20^{\circ}$  (fig. 7) the rolling stability is sensibly neutral but it is believed that the use of normal-rated power in this condition would result in an unstable variation of rolling moment with angle of sideslip.

(b) Reversal of Rolling Velocity. This requirement is discussed under paragraph E-2c(2)(a).

E-2b(3). Side Force Due to Sideslip. It is evident from the steady sideslip characteristics presented in figures 4 through 7 that the variation of side force with sideslip angle is such that positive (right) bank accompanies positive (right) sideslip and vice versa.

E-2c. Static Lateral and Directional Control.

E-2c(1). Rudder control.

(a) Rudder to Overcome Adverse Aileron Yaw. As measured in static wind-tunnel tests the adverse yawing moment caused by the ailerons was small. The amount of rudder control required to neutralize the sideslip developed in rapid rolling maneuvers was not investigated.

(b) Rudder Control in Take-off and Landing. The critical condition for this requirement appears to be the take-off attitude just before the airplane becomes airborne. For the take-off configuration with the flaps deflected  $20^{\circ}$  and with the engine developing rated take-off power, it is estimated that about  $12^{\circ}$  of right rudder and a pedal force of 30 pounds are required to maintain a straight ground path while traveling at a speed of 103 miles per hour into a  $90^{\circ}$  cross wind of 18 miles per hour intensity.



(d) Rudder Control in Dives. An estimated pedal force of 121 pounds is required to maintain a straight course in a dive at a corrected indicated airspeed of 513 miles per hour when the entry is made from a high-speed level-flight condition in which the airplane is trimmed for zero pedal force. The level-flight condition used for the computation corresponded to flight with normal rated power at an indicated airspeed of 302 miles per hour.

It must be emphasized, however, that the hinge-moment data used for these computations was obtained at a very low Mach number and that the pedal force on the airplane under these conditions will probably be considerably different from that given here owing to unpredictable compressibility effects.

#### E-2c(2). Lateral Control.

(a) Maximum Rolling Velocity and Control Force. The estimated variation of  $pb/2V$  with total aileron deflection from trim is given in figures 8 to 10 for various indicated airspeeds together with the estimated variation of the accompanying stick forces for sea-level operation. Figure 8 gives the results of the computations for flaps retracted, and figures 9 and 10 give the results of the computations for flaps deflected  $55^\circ$  and  $20^\circ$ , respectively.

The values of  $pb/2V$  given in figures 8 to 10 were computed from the following formula:

$$\frac{pb}{2V} = 0.8 \frac{C_l}{C_{l_p}}$$

where

$$C_{l_p} = 0.463 \text{ (reference 5)}$$

The factor 0.8 is empirical and its use in previous investigations has been found to give reasonably good correlation between flight results and the estimated values of  $pb/2V$  as computed from wind-tunnel tests. The stick forces were computed using the curve of aileron deflections versus stick displacement (fig. 11).

As far as the present requirement is concerned it is apparent from an examination of figures 8 to 10 that the rolling velocity and aileron control force vary smoothly with aileron deflection and are sensibly proportional to the aileron deflection from trim. These results also indicate that no reversal of rolling velocity should occur.

(b) Aileron Control Force Gradients. Based on the criterion for maximum control friction as given under paragraph E-4d of reference 2, the allowable friction in the aileron control would be 0.8 pounds. For frictional forces of this magnitude it appears that the gradients of the aileron stick force versus aileron deflection in the steady-sideslip maneuvers (figs. 4 to 7) are not large enough to insure good self-centering characteristics.

(c) Peak Value of Rolling Acceleration. As the XP-77 airplane uses the conventional aileron-type lateral control no abnormalities should be experienced in the establishment of the peak value of rolling acceleration during aileron roll maneuvers. For instance, in the aileron roll maneuver depicted in figure 3, the maximum value of the rolling acceleration is attainable well within the allowable limit of 0.09 second.

(d) Minimum Rolling Requirements.

(3) Pursuit (Class III) Airplane. It is evident from an examination of figures 8 to 10 that the response of the airplane to the aileron control is rapid and varies uniformly with aileron deflection for all conditions. The stick forces increase uniformly and never appear excessive. It appears that the maximum values of  $pb/2V$  attainable will be less for most conditions than the value of 0.09 specified. Attention is called, however, to the fact that owing to the extremely short span of this airplane, moderate values of  $pb/2V$  are associated with exceptionally high values of the actual rate of roll  $p$ . For example, in the dive condition shown in figure 8 ( $\alpha = -1.2^\circ$ ,  $V_1 = 368$  miles per hour), the value of  $pb/2V$  for  $16^\circ$  total right aileron deflection is only 0.042 (sea-level value) whereas the actual rate of roll  $p$  is about  $94^\circ$  per second.

(4) Fighter-type airplane. The reduction in available rolling velocities at high speeds due to

wing twist, compressibility effects, etc., is beyond the scope of this report. These effects, however, have been partially compensated for by the use of the factor 0.8 in the expression for  $pb/2V$  in paragraph E-2c(2)(a).

(f) Aileron Control Forces in Dives. An estimated aileron stick force of 7.4 pounds is required to maintain a straight course in a dive at a corrected indicated airspeed of 513 miles per hour when the entry is made from a high-speed level-flight condition in which the airplane was trimmed for zero aileron stick force. The level-flight condition used in the computation corresponds to flight with normal rated power at an indicated airspeed of 302 miles per hour.

Again it is cautioned that the hinge-moment data used for these computations were obtained at a very low Mach number and that the stick force on the airplane may be considerably different from that given here.

E-2c(3). Rudder and Aileron Trimming Devices. No test data were available on the rudder and aileron trimming devices.

### CONCLUDING REMARKS

Based on tests conducted in the NACA full-scale tunnel, an estimate has been made of the lateral flying qualities of the Bell XP-77 airplane. The estimate indicates that the airplane will not meet the Army Air Forces specification that a maximum value of  $pb/2V$  of 0.09 be attainable. The rolling response in maneuvers involving abrupt deflections of the ailerons will be very rapid, however, due to the short span of the airplane. Apart from the low maximum values of  $pb/2V$ , it appears that the airplane will conform to the provisions for satisfactory lateral flying qualities as established by the U. S. Army Air Forces except in the following particulars:

1. In a low-speed, level-flight attitude with flaps deflected  $55^\circ$  the airplane manifests an undesirable variation of rolling moment with angle of sideslip amounting to an effective negative dihedral angle of about  $3^\circ$ .

2. It is believed that a small amount of negative dihedral effect will be exhibited in the climbing configuration with rated power and with flaps deflected  $20^{\circ}$ .

3. The aileron control-force gradients associated with the steady-sideslip maneuvers appear too small to insure good self-centering characteristics.

4. There is a possibility that an undesirably low degree of directional stability may be exhibited for some flight conditions over a very small range of sideslip angles.

Langley Memorial Aeronautical Laboratory  
National Advisory Committee for Aeronautics  
Langley Field, Va., June 16, 1944

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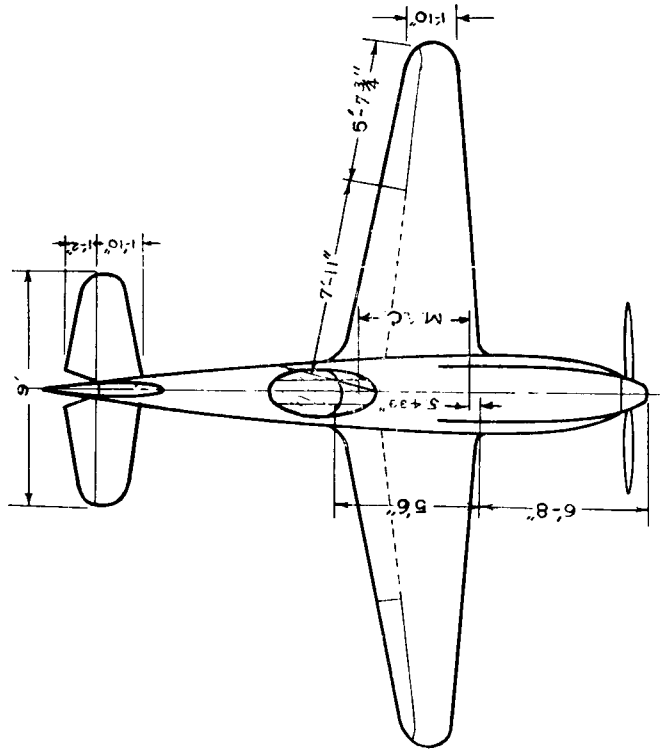
*Clinton H. Dearborn*

Clinton H. Dearborn  
Chief of Full-Scale Research Division

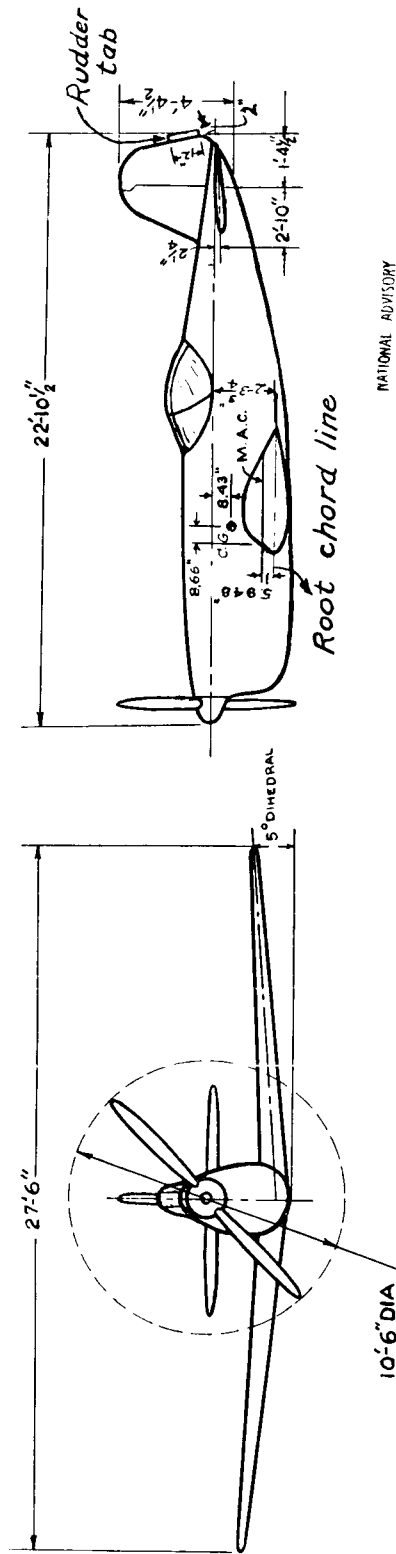
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2. Anon.: Stability and Control Requirements for U. S. Army Airplanes. AAF Specification, June 10, 1943.
3. Bryant, L. W., and Hopwood, M. L.: Calculation of the Damping of the Lateral Oscillation of an Aeroplane. 5559, S. & C. 1307, N.P.L. (British), Jan. 6, 1942.
4. Jones, Robert T.: A Simplified Application of the Method of Operators to the Calculation of Disturbed Motions of an Airplane. NACA Rep. No. 560, 1936.
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|                                      |                                    |
|--------------------------------------|------------------------------------|
| Gross weight                         | 3500 lbs                           |
| Areas                                | sq ft                              |
| Wing                                 | 100.00                             |
| Aileron (total)                      | 504                                |
| Horizontal tail (total)              | 18.90                              |
| Stabilizer                           | 12.50                              |
| Elevator, aft of hinge line          | 6.40                               |
| Vertical tail (total)                | 1237                               |
| Fin                                  | 738                                |
| Rudder, aft of hinge line            | 475                                |
| Flap, aft of hinge line (total)      | 8.58                               |
| Mean aerodynamic chord               | 3.99 ft                            |
| Airfoil section                      |                                    |
| Root                                 | Modified NACA-65(216)-017          |
| Tip                                  | Modified NACA-671-(13)15           |
| Angle of incidence (at root section) | 2°                                 |
| Washout                              | 1°30'                              |
| Propeller - Aero products            | A-15-132-two blades constant speed |

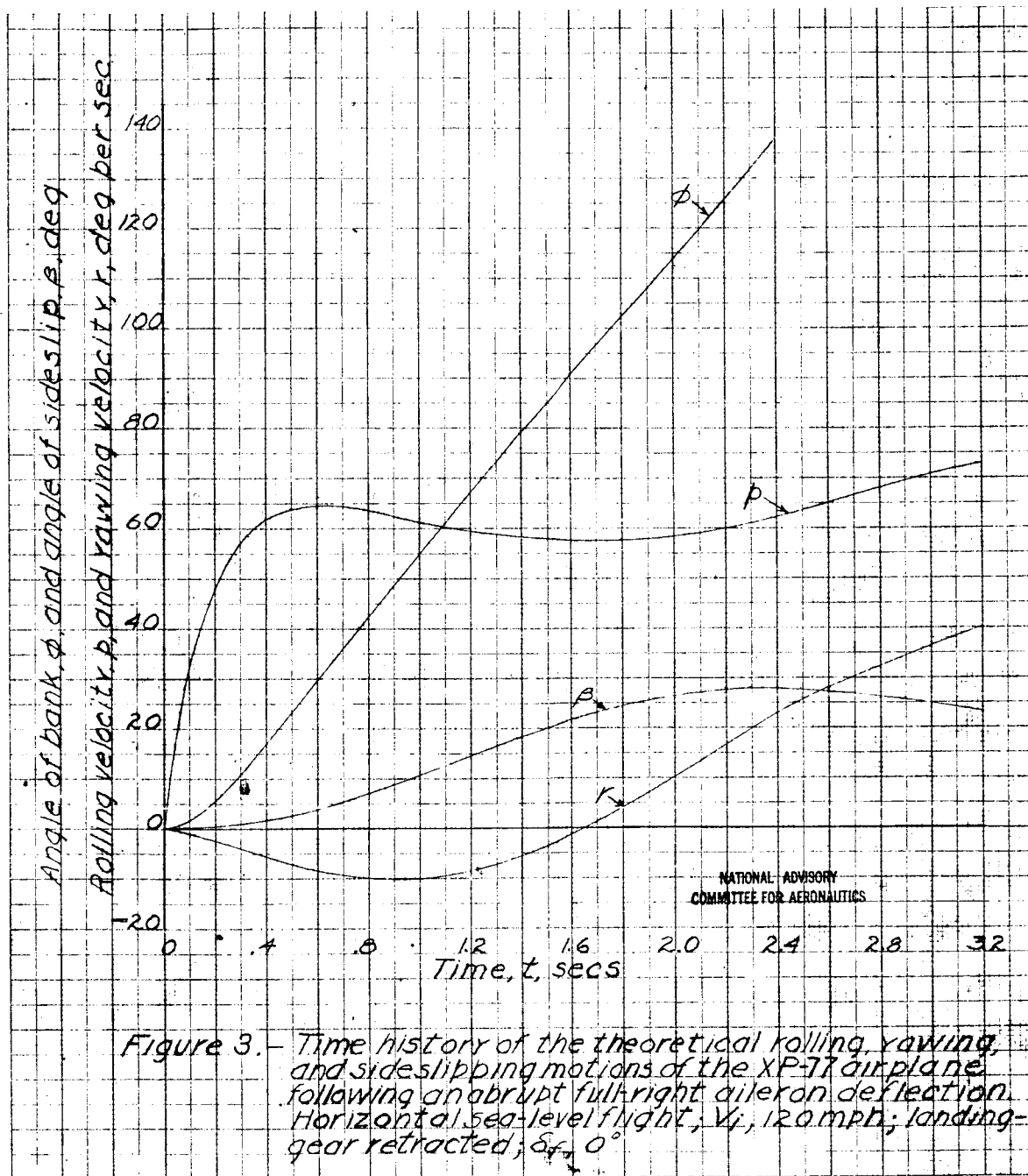


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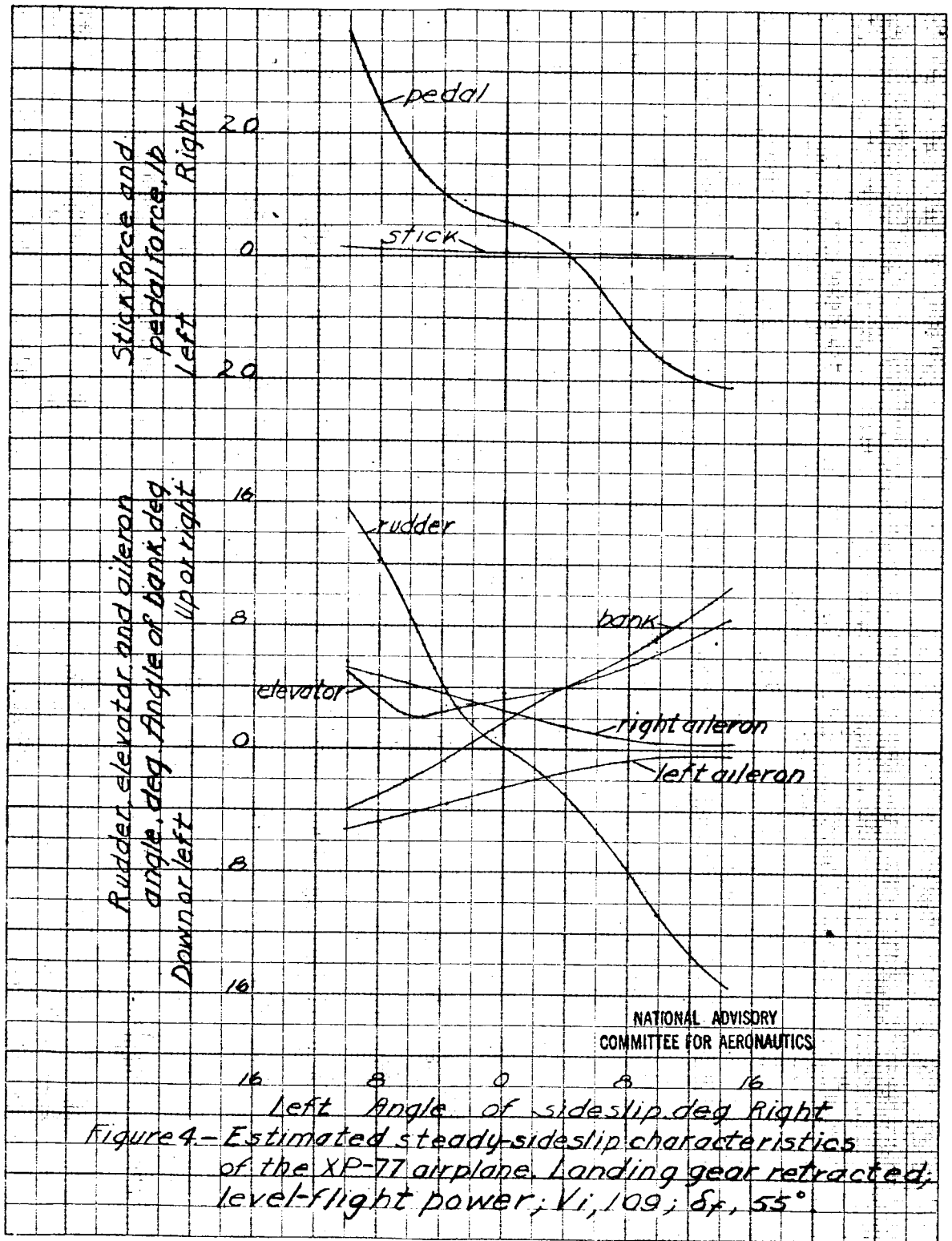
Figure 1.-Three-view drawing of the XP-77 airplane used in the full-scale tunnel.

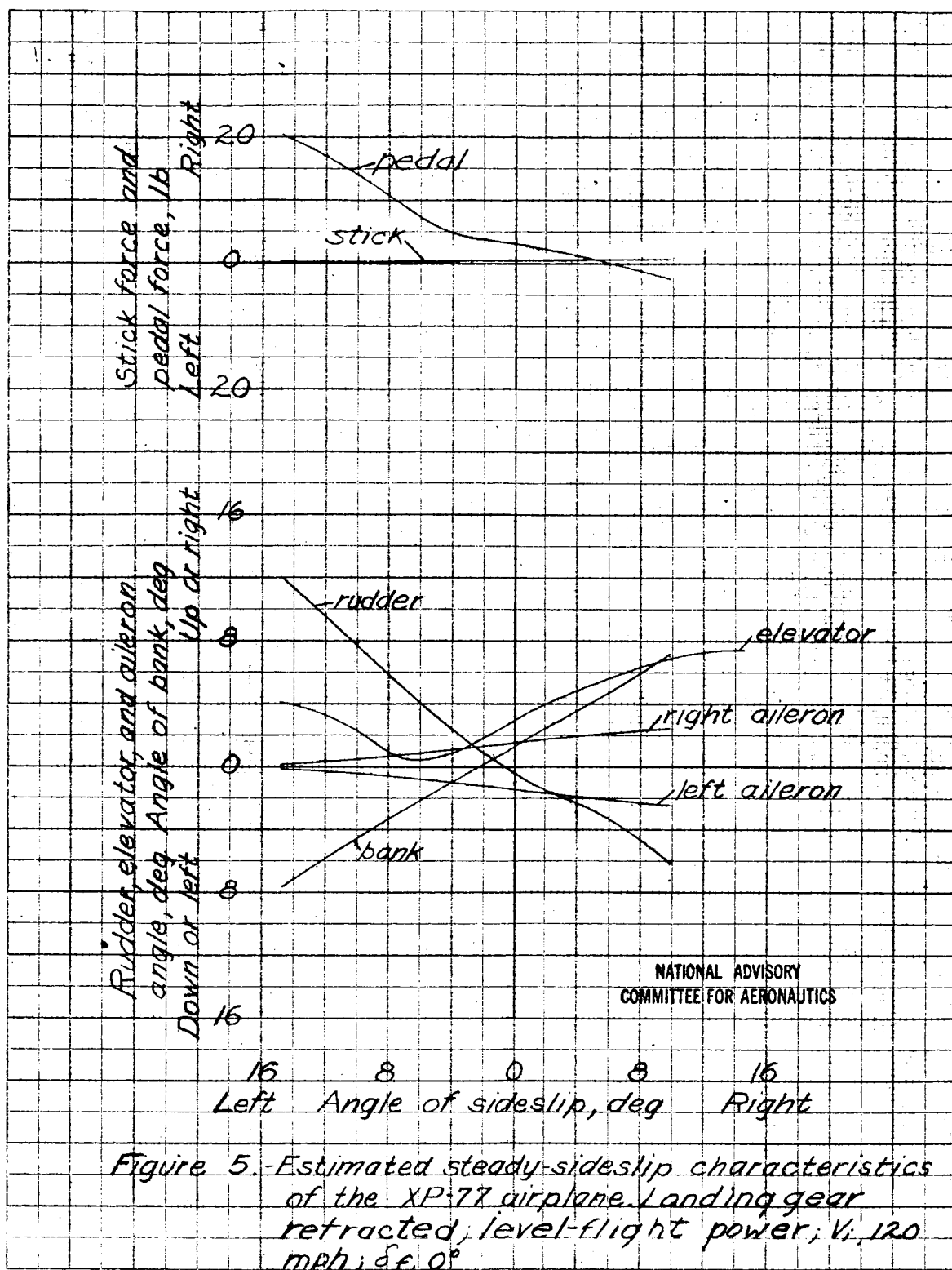


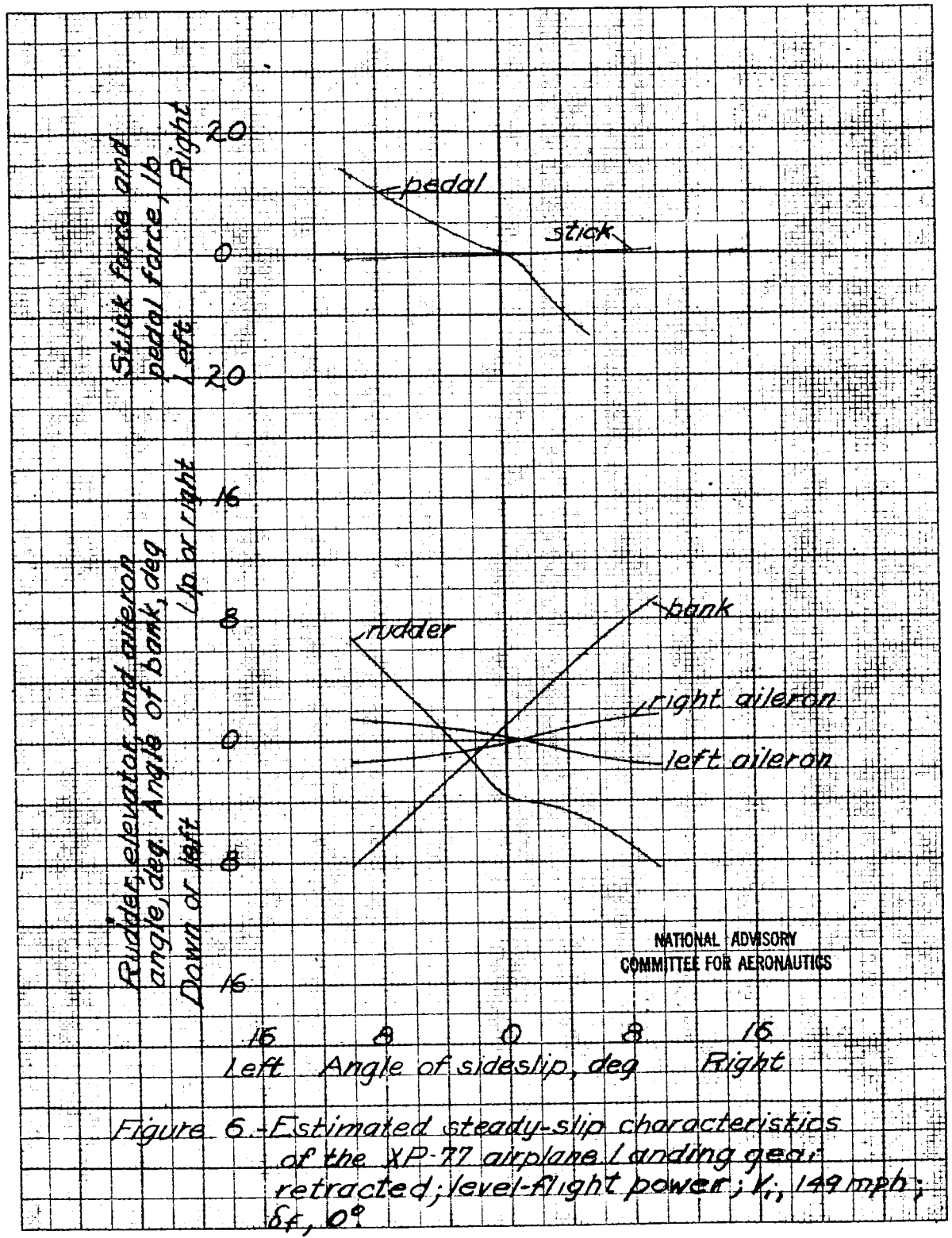
Figure 2.- The XP-77 airplane mounted in the full-scale tunnel.

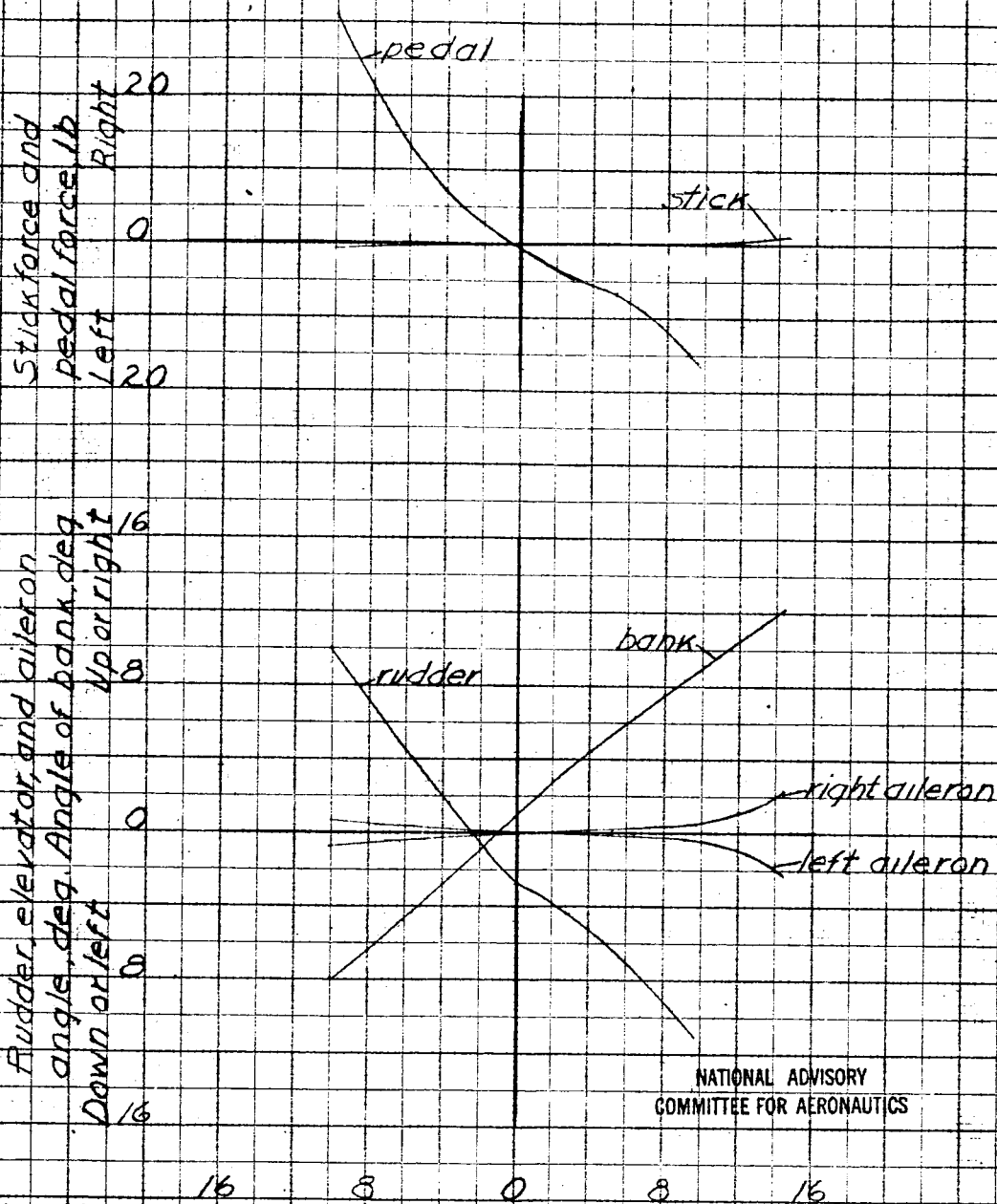






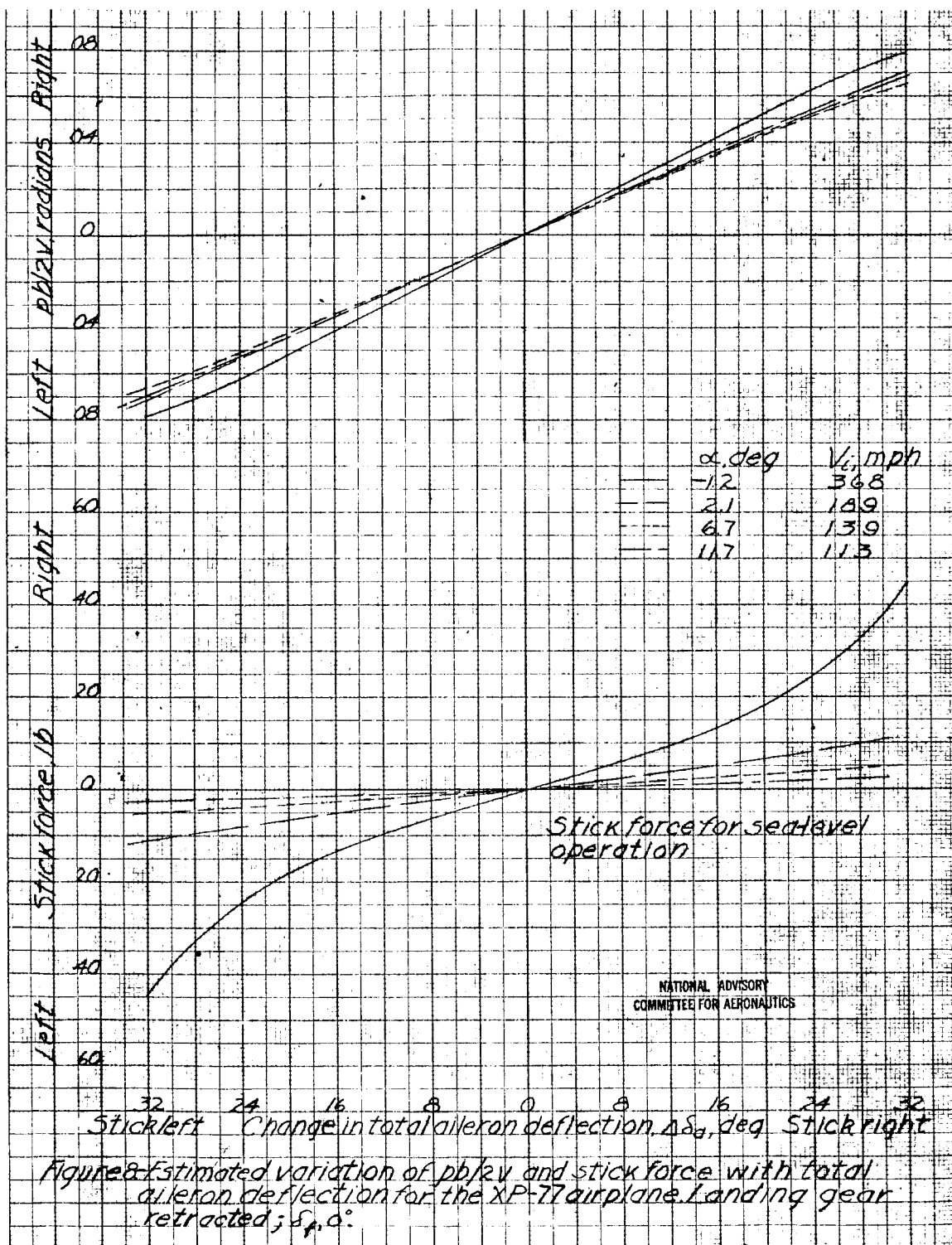






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Left Angle of sideslip, deg Right  
Figure 7- Estimated steady-sideslip characteristics  
of the XP-77 airplane. Landing gear retracted;  
level flight power;  $V_i, 143$ ;  $\delta_F, 20^\circ$



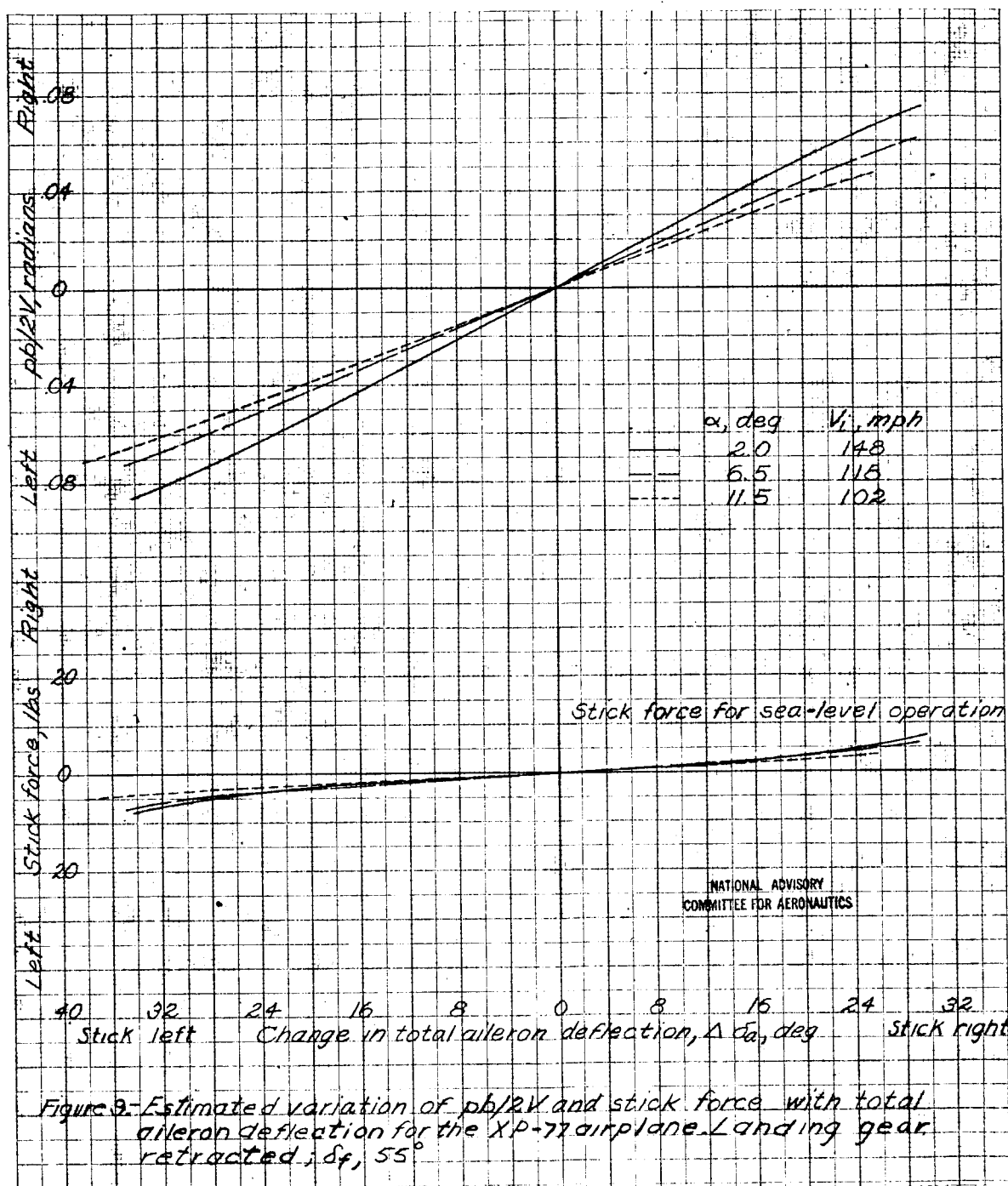
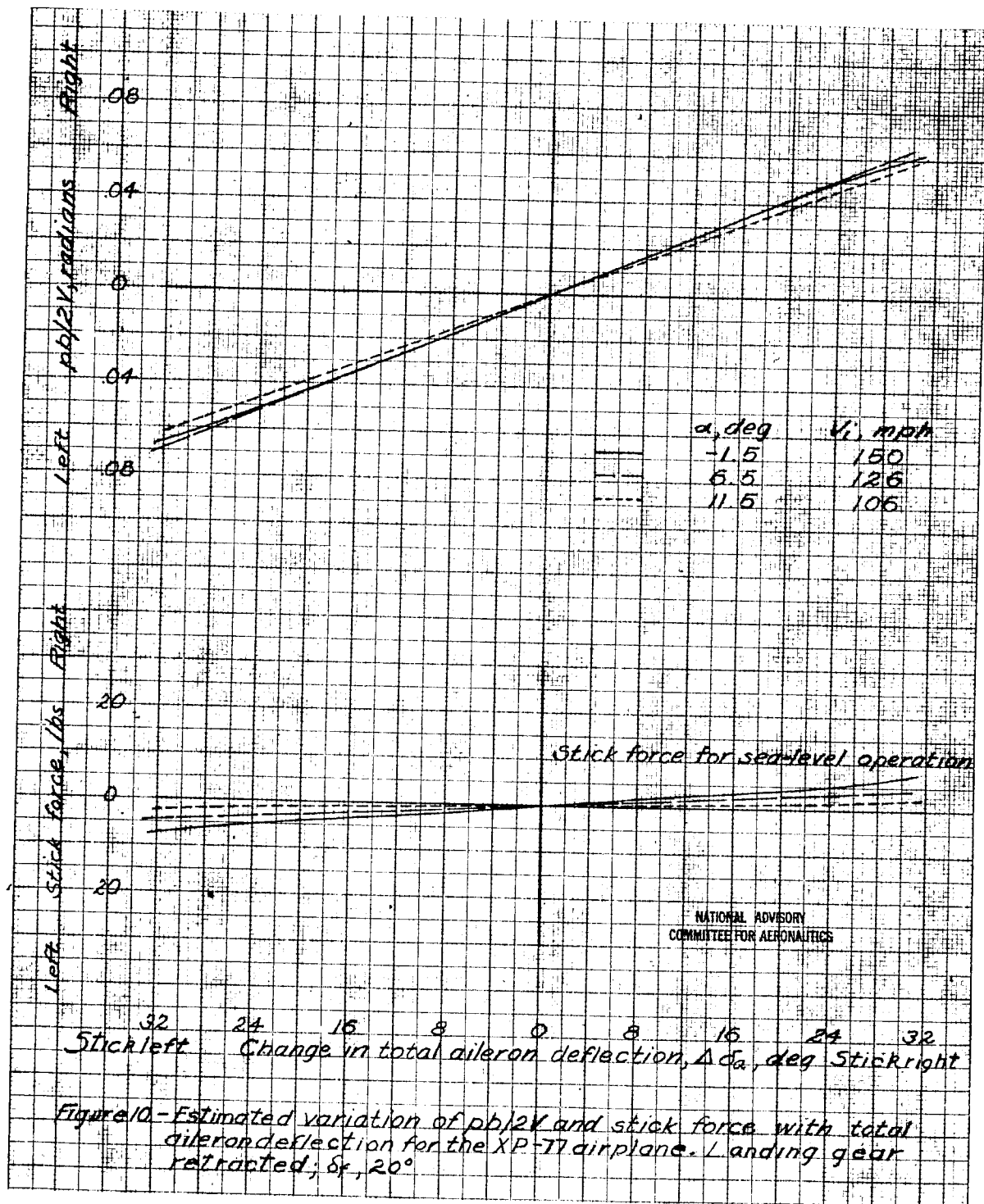
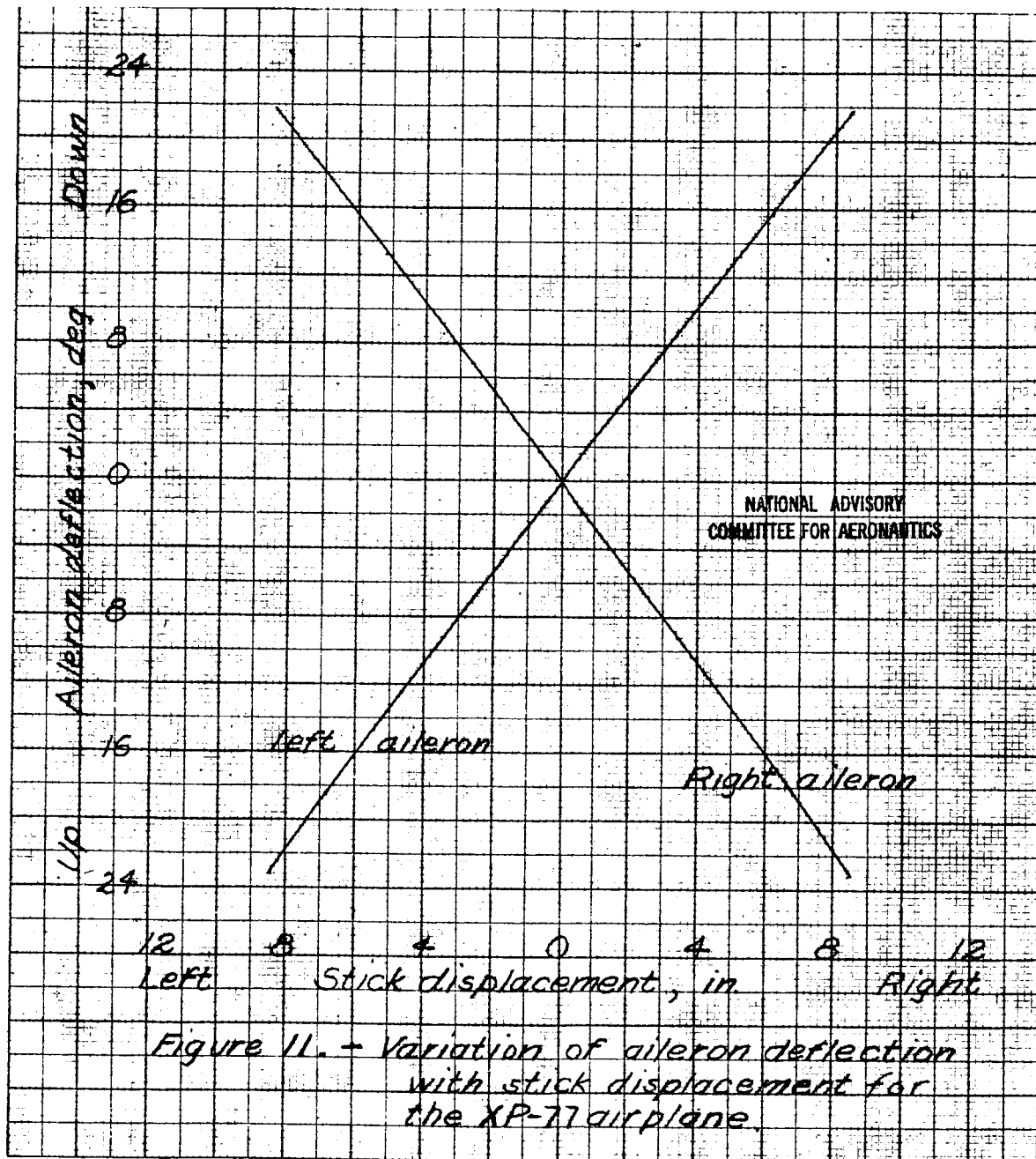


Figure 3- Estimated variation of  $pb/2V$  and stick force with total aileron deflection for the XP-77 airplane. Landing gear retracted;  $\delta_f$ ,  $55^\circ$







## REPORT No. 796

### AN INTERIM REPORT ON THE STABILITY AND CONTROL OF TAILLESS AIRPLANES

By LANGLEY STABILITY RESEARCH DIVISION

COMPILED by CHARLES J. DONLAN

#### SUMMARY

Problems relating to the stability and control of tailless airplanes are discussed in consideration of contemporary experience and practice. In the present state of the design of tailless airplanes, it appears that:

(1) Sweepback affords a method of supplying tail length for directional and longitudinal stability and control and allows the utilization of a high-lift flap but introduces undesirable tip stalling tendencies that must be overcome before the advantages of sweepback can be realized.

(2) The damping in pitching appears to have little effect on the longitudinal behavior of the airplane provided the static margin is never permitted to become negative.

(3) The directional stability must be as great as for conventional airplanes if the same requirements regarding satisfactory stability and control characteristics are to be adhered to.

(4) The influence of the lateral resistance and the damping in yawing on the flying qualities is somewhat obscure; however it is believed that these parameters will be of secondary importance if adequate directional stability is supplied.

(5) On account of the difficulties encountered in obtaining adequate stability and control with tailless airplanes, it appears that a thorough reevaluation of the relative performance to be expected from tailless and conventional designs should be made before proceeding further with stability and control studies.

#### INTRODUCTION

Much interest has been shown in tailless airplanes during the past few years. A number of tailless-airplane designs have appeared and prototypes of several of these designs have been flown extensively. It appears desirable at this time to amplify and expand an earlier work (reference 1) relating to the stability and control of tailless airplanes in the light of the recent flight experience acquired and the related studies that have accompanied the development of new designs.

It is the purpose of this paper to assemble and record some expressions of fact and opinion pertaining to numerous problems that have assumed significance in tailless-airplane design rather than to supply specific quantitative design data. The problems specifically discussed in this paper pertain to the requirements and attainment of longitudinal and lateral stability and control and to spinning, tumbling, and steadiness in flight as regards gunnery and bombing platform. A discussion is also included of some of the relative merits of tailless and conventional airplanes.

#### SYMBOLS

$C_L$  lift coefficient  
 $C_D$  drag coefficient  
 $C_l$  rolling-moment coefficient  
 $C_n$  yawing-moment coefficient  
 $C_m$  pitching-moment coefficient  
 $V$  airspeed  
 $r$  yawing angular velocity  
 $\rho$  density of air  
 $m$  mass of airplane  
 $q$  dynamic pressure ( $\frac{1}{2}\rho V^2$ ); also, pitching angular velocity

$T_c$  thrust coefficient ( $\frac{\text{Thrust}}{\rho D^2 V^2}$ )  
 $C_h$  hinge-moment coefficient  
 $\alpha$  angle of attack  
 $\beta$  angle of sideslip  
 $\Lambda$  angle of sweep  
 $\lambda$  taper ratio; ratio of tip chord to root chord  
 $S$  wing area, except as designated otherwise by subscript  
 $c$  wing chord, except as designated otherwise by subscript  
 $\bar{c}$  mean aerodynamic chord  
 $A$  aspect ratio  
 $x$  distance of aerodynamic center from center of gravity  
 $z$  vertical displacement of thrust axis from center of gravity (positive when thrust axis is below center of gravity)  
 $b$  wing span, except as designated otherwise by subscript  
 $D$  propeller diameter  
 $F_s$  stick force  
 $\phi$  trailing-edge angle (see fig. 11)  
 $\theta$  landing-gear angle (see fig. 13)  
 $\delta$  control-surface deflection

$$C_{L_\alpha} = \frac{\partial C_L}{\partial \alpha}$$

$$C_{L_\delta} = \frac{\partial C_L}{\partial \delta}$$

$$C_{n_\delta} = \frac{\partial C_n}{\partial \delta}$$

$$C_{l_\delta} = \frac{\partial C_l}{\partial \delta}$$

$$C_{r_r} = \frac{\partial C_r}{\partial \frac{rb}{2V}}$$

$$C_{m_e} = \frac{\partial C_m}{\partial \frac{qc}{2V}}$$

$$C_{l_r} = \frac{\partial C_l}{\partial \frac{rb}{2V}}$$

$$\mu \text{ relative density } \left( \frac{m}{\rho S b} \right)$$

$$C_{n_\beta} = \frac{\partial C_n}{\partial \beta}$$

$$C_{l_\beta} = \frac{\partial C_l}{\partial \beta}$$

Subscripts:

- f* flap; also, flipper  
*a* aileron  
*e* elevator  
*t* tab  
*r* rudder  
*c/4* about quarter point of mean aerodynamic chord

#### LONGITUDINAL STABILITY AND CONTROL

It was noted in reference 1 that a straight wing with a slight reflex camber and dihedral has all the necessary aerodynamic characteristics for both longitudinal and lateral stability. A straight wing employing a trailing-edge flap as a trimming control, however, suffers an undesirable loss in maximum lift, particularly if the static margin is large. In order to improve this condition, the installation of leading-edge slats has been considered. This solution has found little favor, however, because of the accompanying increase in profile drag and the unusually high attitude required for landing with leading-edge slats. At the present time the most practicable method of overcoming the deficiency in maximum lift appears to be to incorporate sweepback (or sometimes sweepforward) into the wing. The majority of the contemporary problems in longitudinal stability of tailless airplanes arise from the adoption of this solution.

#### EFFECTS OF SWEEP

**Advantages of sweep.**—Sweepback gives the wing an effective "tail length" and is therefore especially adaptable for tailless airplanes. This tail length is proportional to the product of one-half the span of the portion of the wing with sweep and the tangent of the sweep angle; consequently, (1) high-lift flaps can be located at the center of the wing where their lift increments produce only minor changes in the pitching moment about the center of gravity of the airplane, (2) flaps for longitudinal control can be located near the wing tips where only minor changes in lift are needed to produce the requisite pitching moments for trim, and (3) more leeway is permitted in locating the center of gravity inasmuch as the aerodynamic center of the wing can be controlled by the angle of sweepback.

If only high lift is considered, the results of an investigation relating to the use of various types of flap on swept-back

wings have indicated that trailing-edge split flaps are particularly suitable for swept-back wings because of the relatively small pitching-moment increment accompanying the production of a given lift increment (reference 2). The ratio of the pitching-moment increment to the lift increment produced by a flap depends, of course, on the position of the centroid of the flap load relative to the aerodynamic center of the wing. The centroid of the flap load has been observed to move forward along the wing chord as the hinge-line position of the flap is shifted forward, with the consequence that the ratio of the flap pitching-moment increment to the flap lift increment is reduced. The extent of the forward movement of the centroid of the incremental flap load accompanying a forward shift of the flap hinge line that may be expected for full-span trailing-edge split flaps is given in figure 1. It was noted in reference 3 that the ratio of the flap pitching-moment increment to the flap lift increment could be considerably reduced by moving the flap hinge line forward with only slight losses in the magnitude of the flap lift associated with a given flap deflection. It appears, therefore, that shifting the hinge line of the flap affords a promising means of minimizing the pitching moments caused by high-lift flaps, but more data on this effect are needed before specific recommendations can be made.

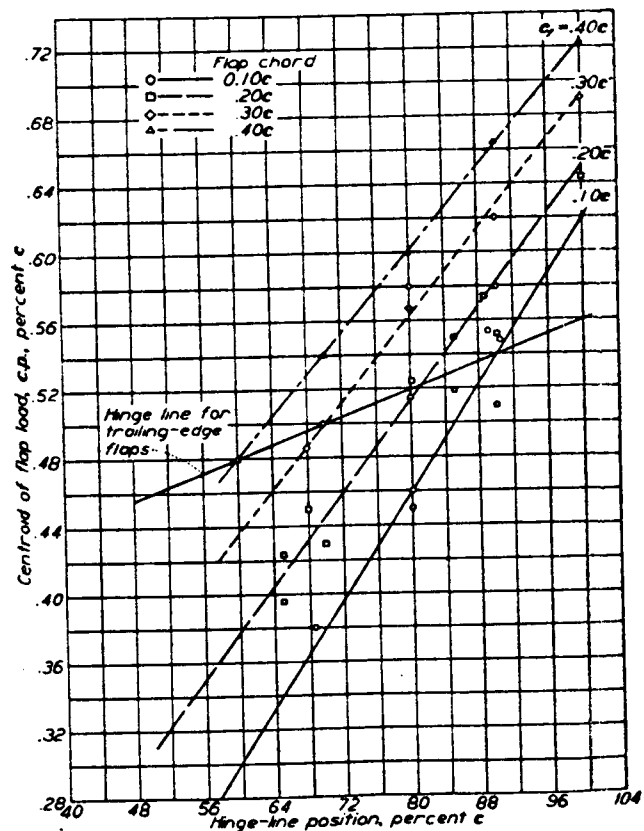


FIGURE 1.—Variation of centroid of incremental flap load with flap hinge-line position for full-span trailing-edge split flaps.

It is known that, for trailing-edge flaps, an increase in flap chord shifts the centroid of the incremental flap load forward and thus causes a reduction in the ratio of the flap pitching-moment increment to the flap lift increment. This effect can be observed in figure 1 by comparing the results for different flap chords. At the present time, the optimum combination of flap size and flap hinge-line position for specific designs must be determined by experiment.

The lift increments produced by flaps are governed also by the plan form of the basic wing design. The important factors are (1) the aspect ratio, (2) the taper ratio, and (3) the angle of sweep. Of particular interest for tailless airplanes is the so-called self-trimming flap, which is a flap arranged to produce zero pitching-moment increment about the aerodynamic center of the wing. The effect of aspect ratio on the lift-coefficient increment produced by a self-trimming trailing-edge split flap on a swept-back wing is shown in figure 2. The effect of taper ratio on the lift-coefficient increment produced by a flap is discussed in reference 4 and an indication of the effect to be expected can be obtained from figure 3. In general, a moderate taper ratio of the order of 2:1 is recommended. The effect of sweepback on the lift increment produced by a self-trimming trailing-edge split flap on a swept-back wing is shown in figure 4. The data in figures 2 to 4 were taken from an analytical investigation of self-trimming trailing-edge split flaps (reference 2).

Although trailing-edge split flaps have been found to be particularly beneficial on swept-back wings in producing high lift, it is cautioned that there are considerations other than high lift involved in the selection of a flap for a specific design. For example, consideration of the minimum drag of flaps for take-off, ground clearance, and the operation of a pusher propeller in the flap wake may lead to the adoption of some other flap even at some sacrifice in lift.

Increases in maximum lift can be expected with swept-

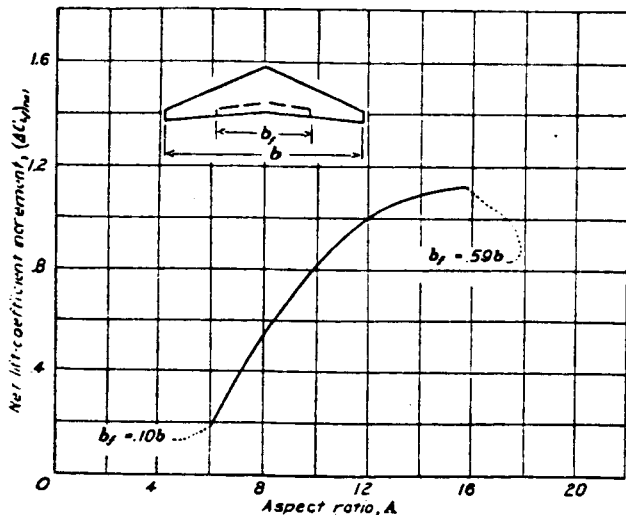


FIGURE 2.—Effect of aspect ratio on the increment in lift coefficient produced by trailing-edge split flap having zero pitching-moment increment about the aerodynamic center of the wing.  $\frac{1}{A}=4:1$ ;  $A=20^\circ$ ;  $c_f=0.30c$ ;  $\delta_f=60^\circ$ .

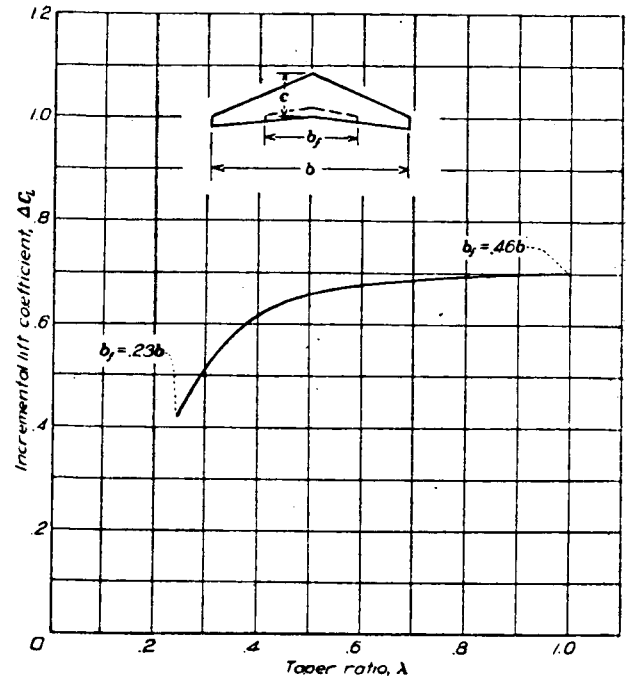


FIGURE 3.—Effect of taper ratio on the increment in lift coefficient produced by trailing-edge split flap having zero pitching-moment increment about the aerodynamic center of the wing.  $A=20^\circ$ ;  $A=7.3$ ;  $c_f=0.30c$ ;  $\delta_f=60^\circ$ .

forward wings, provided the high-lift flaps are placed on the outer portion of the wing span and the flap for longitudinal control is placed at the center of the wing.

**Disadvantage of sweep.**—A most disagreeable characteristic of a swept-back wing is the inherent tendency to stall prematurely at the tips, a phenomenon primarily associated with the lateral flow of the boundary layer. This characteristic is particularly undesirable because it occurs first

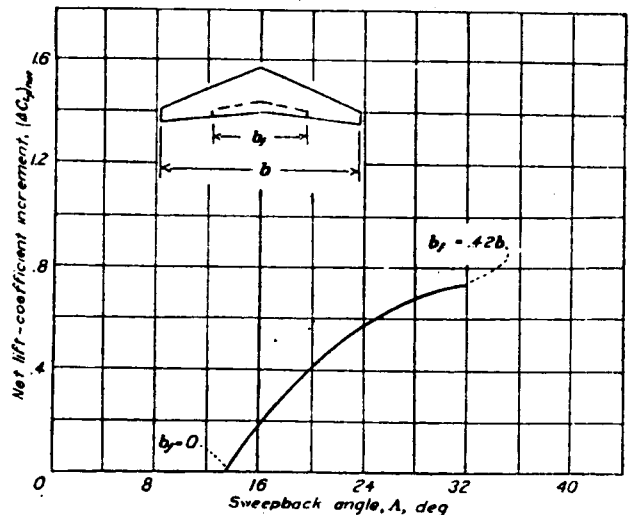


FIGURE 4.—Effect of sweepback on the increment in lift coefficient produced by trailing-edge split flap having zero pitching-moment increment about the aerodynamic center of the wing.  $\frac{1}{A}=4:1$ ;  $A=7.3$ ;  $c_f=0.30c$ ;  $\delta_f=60^\circ$ .

over the rear portion of the wing where the control surfaces are located. The tip stall is manifested as a pronounced pitching and rolling instability accompanied by a tendency of the elevators or ailerons to float upward. An example of the effect produced by the tip stall on the pitching moment of a swept-back wing is given in figure 5. The rapid increase in positive pitching-moment coefficient accompanying the tip stall is characteristic.

Swept-forward wings tend to stall first at the central part of the wing. Center-section stalling causes pitching instability but the rolling instability associated with the tip stall

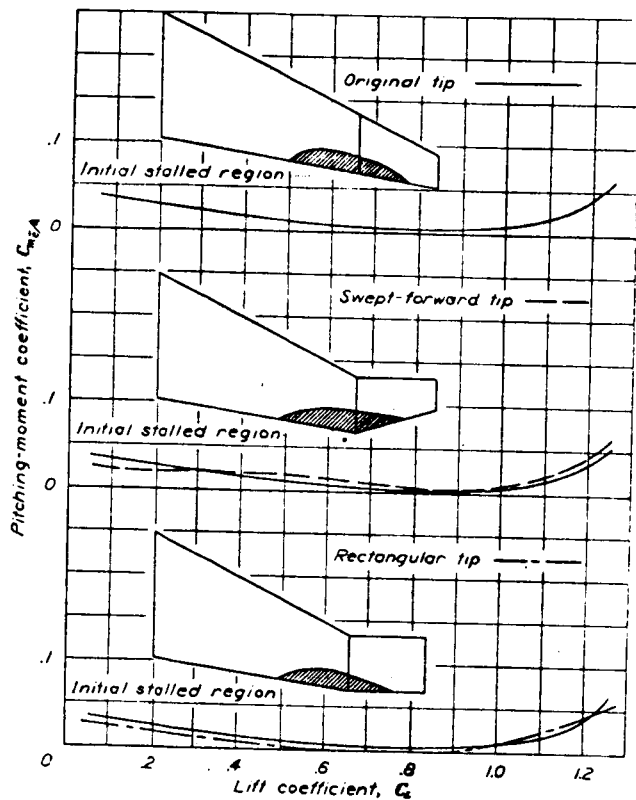


FIGURE 5.—Effect of change in tip shape on the pitching-moment characteristics of a swept-back wing.

of swept-back wings does not occur. This advantage of sweepforward, however, is partly offset by the difficulty created in obtaining adequate static balance on account of the forward shift in the aerodynamic center of the wing caused by sweepforward. With swept-forward wings, the fuselage or load-carrying element must be placed ahead of the wing in order that the center of gravity may be ahead of the aerodynamic center.

**Remedies for tip stalling.**—Before satisfactory flight behavior can be assured, provision must be made for delaying or eliminating the tip stall. Various schemes have been proposed for delaying or eliminating the tip stall and a number of such schemes are summarized as follows:

(1) *Wing twist.*—It has been proposed to wash out the wing tips, that is, to lower the angle of attack of the section near the tip. Reference 5 shows that the amount of wash-out required to benefit the tip stalling characteristics is sufficient to increase the drag of the wing seriously at low angles of attack. One method of avoiding the high drag is to have a portion of the wing tip rotatable in flight. The rotatable wing tips should be so proportioned with respect to the elevator that the airplane cannot be stalled until the tip angle has been sufficiently reduced to eliminate the tip stall.

(2) *Change in airfoil section.*—The initial stalling of the wing sections on the outer span of the wing can be controlled somewhat by increasing the thickness or changing the camber of the airfoil sections used. The results of reference 5 indicate that this method can appreciably increase the angle of stall of a wing without flaps or sweepback, particularly if a change in camber is used in conjunction with wing twist. The analysis in reference 5 does not consider the effects of sweep or flaps. Changing the wing sections, however, generally has the disadvantage of increasing the drag of the wing at low angles of attack.

(3) *Flat-plate separators.*—It has been suggested that the tip stall might be delayed by means of vertical flat plates or fins aligned with the wing chord at about one-half the distance to the wing tip and extending around the trailing edge of the wing and forward almost to the leading edge. The function of the plate is to prevent cross flow of the boundary layer by "separating" the fields of flow along the wing span. Experiments on swept-back wings with flat-plate separators installed have indicated that some increase in the angle of stall can be obtained by this method alone but that generally a new stall is induced just inboard of the plate itself. Better results might be obtained if the flat-plate separators are used in conjunction with changes in wing plan form, particularly in the vicinity of the wing tip.

(4) *Changes in plan form at tip.*—According to tests made in the Langley free-flight tunnel, a change in wing plan form at the tip alone has little effect on the tip stall (fig. 5), as evidenced by the instability manifested by the pitching-moment curves for all tip arrangements. It appears from associated tuft studies that flow separation always occurs at the junction between the tip and the inboard portion of the wing. In any event, the change in plan form should extend inboard of the original stalled regions.

(5) *Leading-edge slats.*—The use of tip slats has been found to be the most effective method of delaying the tip stall. Leading-edge slats may increase the angle of stall as much as  $10^\circ$  if judiciously located. Tests of models in the Langley free-flight tunnel have indicated the necessity of extending the slat at least over the portion of the wing affected by the stall. It has been found that slat spans of the order of 30 to 50 percent of the wing span are necessary to abolish completely the effects of the tip stall. Typical stalled areas behind a swept-back wing with various slat arrangements are shown in figure 6.

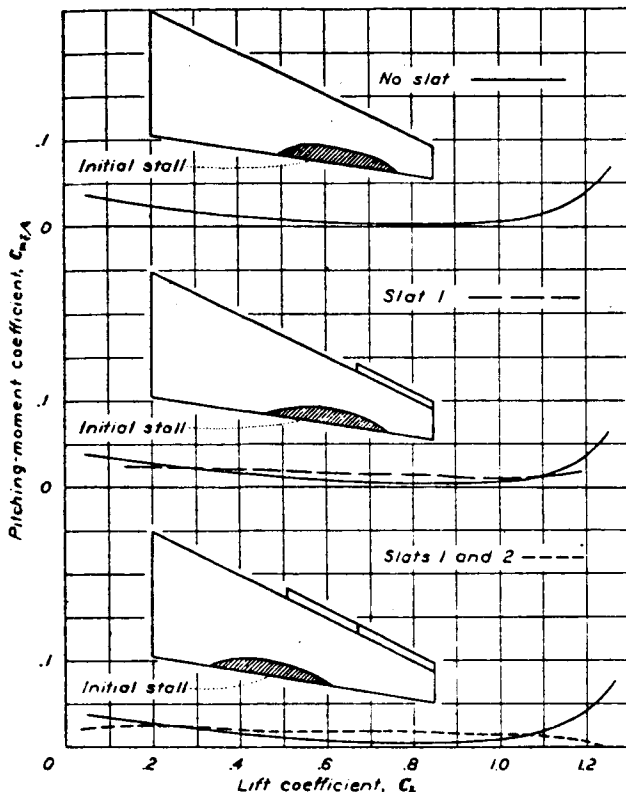


FIGURE 6.—Effect of slat on the pitching-moment characteristics of a swept-back wing.

If fixed slats are used, an undesirable increase in drag may result at low angles of attack. It may be possible, however, to build retractable slats that have only minor effects on the over-all drag of the wing at low angles of attack after more research and work on the development of retractable slats have been done.

(6) *Taper*.—Part of the stalling of swept-back wings can be attributed to high taper. The use of highly tapered swept-back wings should be avoided, therefore, inasmuch as data on tapered wings indicate that the beneficial effects of sweepback can be obtained with moderate taper ratios of the order of 2:1 (reference 5).

#### LONGITUDINAL STABILITY

As with a conventional airplane, a tailless airplane is statically stable if the center of gravity is ahead of the aerodynamic center. The position of the aerodynamic center is appreciably affected by (1) the addition of a fuselage or a streamline nacelle, (2) sweepback, and (3) power. The extent of the forward shift of the aerodynamic center produced by a fuselage or nacelle has been discussed in reference 6. The basic procedures for calculating the aerodynamic center of wings of various plan forms are given in reference 4. Applications of lifting-surface theory to the determination of the span loading of swept-back wings can be found in reference 7. The effects of power on longitudinal stability are discussed in the following paragraphs.

**Effects of power.**—The analysis of the effects of power on the longitudinal stability is somewhat simpler for tailless airplanes than for the conventional airplane on account of the absence of the horizontal tail. For convenience, the effects are divided into three parts:

- (1) Effects associated with normal force and direct thrust of propellers
- (2) Effects associated with slipstream velocity and downwash behind propellers
- (3) Effects associated with dynamic action of jets

The effect of the propeller normal force is small for the conventional arrangements of propellers and is usually a fixed factor for a given design. Methods of estimating the effect are available in reference 8.

As with conventional airplanes, the effect of the thrust on stability is directly proportional to the product of the thrust and the perpendicular distance from the center of gravity of the airplane to the thrust line. This effect is controlled, of course, by the vertical location of the propeller and the inclination of the thrust line. The farther above the center of gravity the thrust line passes, the greater is the stabilizing effect produced by a given thrust; and the farther below the center of gravity the thrust line passes, the greater is the instability produced by a given thrust. In any case, the farther from the center of gravity the thrust line passes, the greater are the changes in trim due to the thrust that accompany changes in power. This effect is illustrated in figure 7. The effects of power were small when the thrust-line axis passed close to the center of gravity of the airplane. When the thrust line was 0.0488 below the center of gravity, however, the stability decreased appreciably.

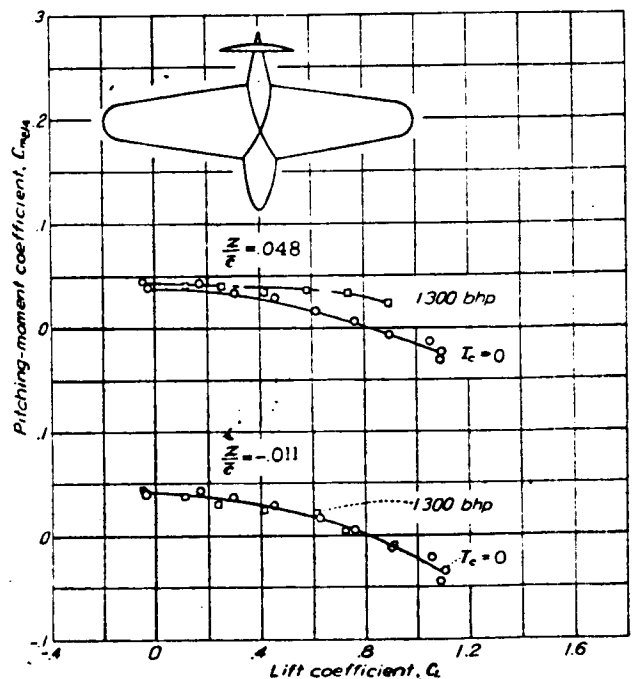


FIGURE 7.—Effect of power on the longitudinal stability of a pusher-type tailless airplane.

At a lift coefficient of 0.8, the static margin decreased from 0.04 to 0.012 and the unbalanced pitching moment introduced by the thrust required about  $10^\circ$  of down elevator to trim the airplane.

The propeller slipstream is an important contributing item to the longitudinal stability characteristics of the airplane—particularly for tractor arrangements. The controlling factor is the location of the aerodynamic center of the portion of the wing immersed in the slipstream. If the aerodynamic center of this portion of the wing is behind the center of gravity of the airplane, the slipstream produces a stabilizing effect; if the aerodynamic center of this portion of the wing is ahead of the center of gravity of the airplane, the slipstream produces a destabilizing effect. Design parameters affecting the contribution of the propeller slipstream are (1) the loca-

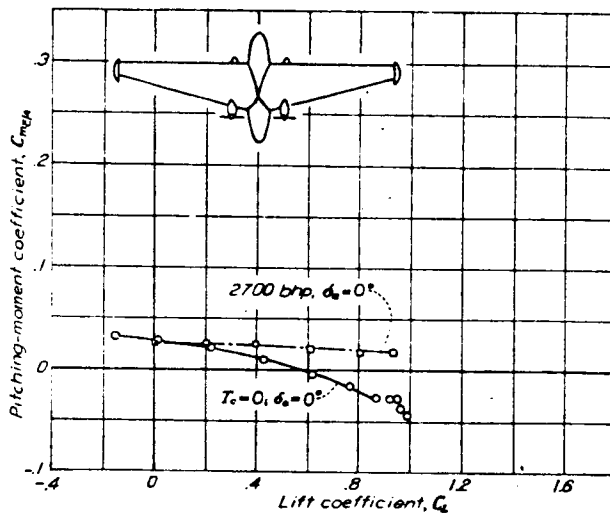


FIGURE 8.—Effect of power on the longitudinal stability of a tractor-type tailless airplane.

$$\frac{z}{c} = 0.$$

tion of the section aerodynamic centers, (2) the spanwise location of the propellers, and (3) the inclination of the propeller axis. The basic moment of the immersed wing sections also has an effect. Figure 8 indicates the magnitude of some of the power effects to be expected.

For the tractor-type tailless airplane shown in figure 8, the thrust line passes near the center of gravity so that the effect of the thrust is negligible. The aerodynamic centers of the wing sections immersed in the slipstream are ahead of the center of gravity, however, and the slipstream therefore produces a destabilizing effect.

From consideration of changes in static margin and trim, it appears desirable on tailless airplanes of the pusher type to locate the thrust line close to the center of gravity of the airplane ( $\frac{z}{c} < 0.01$  is recommended) and, if feasible, to locate the propeller so that the aerodynamic centers of the wing sections affected by the inflow to the propeller are either on

or slightly behind the lateral axis through the center of gravity of the airplane.

For jet-propelled airplanes, the location and inclination of the jet axis exercises an effect on the stability characteristics of the airplane similar to the effect produced by the thrust of a propeller. At the present time, it appears that the location of the jet axis should be governed by the same factors which were considered in the discussion concerning the location of the thrust line of conventionally powered airplanes.

**Damping in pitch.**—As pointed out in reference 1, the low value of  $C_{m_q}$  associated with tailless airplanes is no serious disadvantage so far as the damping of the oscillations is concerned if the airplane has a positive static margin. It appears that damping is introduced by the particular coupling of the modes of motion as affected by the low value of  $C_{m_q}$  and by the reduced radius of gyration in pitch as shown in reference 1 and figure 9. The results of tests in the Langley free-flight tunnel (reference 9) indicated that changes in the rotational damping in pitch have little effect on the longitudinal steadiness for values of the static margin greater than 0.03.

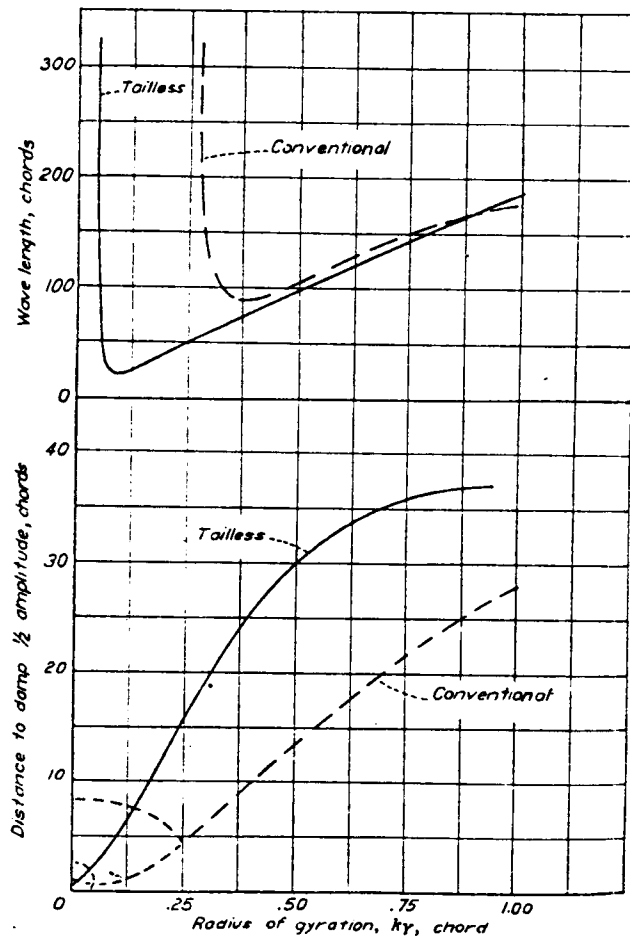


FIGURE 9.—Effect of radius of gyration in pitch on the short-period longitudinal oscillation of a tailless and a conventional airplane. Static margin = 0.075;  $\mu = 12$ .

It was pointed out in reference 1 that the reduced damping in pitch of a tailless airplane might result in an uncontrollable motion of the airplane if the static margin is allowed to become negative. This contention has been supported by subsequent tests in the Langley free-flight tunnel (reference 9). The tests indicated that a serious form of instability may develop when the static margin of a tailless airplane becomes negative. As a result of this danger of uncontrollable motions with negative static margins, it is recommended that the center of gravity of a tailless airplane never be permitted, under any conditions, to reach a position behind the aerodynamic center.

**Tumbling.**—A form of dynamic instability of tailless airplanes may be manifested as tumbling. Tumbling consists of a continuous pitching rotation about the lateral axis of the airplane. The maneuver is extremely violent and imposes severe accelerations on parts of the airplane. So far as is known, there are no authenticated instances of the occurrence of tumbling in flight. Models of tailless airplanes have been made to tumble in the Langley 20-foot free-spinning tunnel, however, by forcing the model to simulate a whip stall. At the present time, however, little is known about the mechanics of the tumbling motion. Tests conducted in the Langley 20-foot free-spinning tunnel have shown that the position of the center of gravity has a pronounced effect on the motion. It appears that provision of a large static margin prevents tumbling but that a stable tumbling condition may exist if the static margin is slight. Tests have shown also that once the tumbling motion has started the normal flying controls are relatively ineffective for recovery from this stable tumbling condition.

In view of the severity of the tumbling maneuvers, it is recommended that tumbling tests be required of models of all fighter tailless airplanes.

#### LONGITUDINAL CONTROL

One of the difficult problems in the design of tailless airplanes is the provision of adequate longitudinal control. The type of longitudinal control usually employed consists of an elevator (or flap) placed at the trailing edge of the wing. With this type of control, the loss in lift caused by the flap deflection required to trim the airplane can be appreciable, particularly for a tailless airplane with a large static margin. The computed loss in lift that results from trimming the airplane at various values of static margin is shown in figure 10. It is evident from figure 10 that the loss in lift caused by the longitudinal control can be minimized by placing the control surfaces at the tips of highly swept-back wings of high aspect ratio. When the longitudinal control is placed near the wing tips, the elevator can be combined with the aileron in an arrangement to be discussed later in the section entitled "Aileron Control."

**Design requirement.**—It is to be expected that the elevator stick-force requirements for tailless airplanes should be the same as for conventional airplanes of the same class. The balance requirements for tailless airplanes, however, are more severe than for conventional airplanes. For the same static margin, the elevator of a tailless airplane usually must

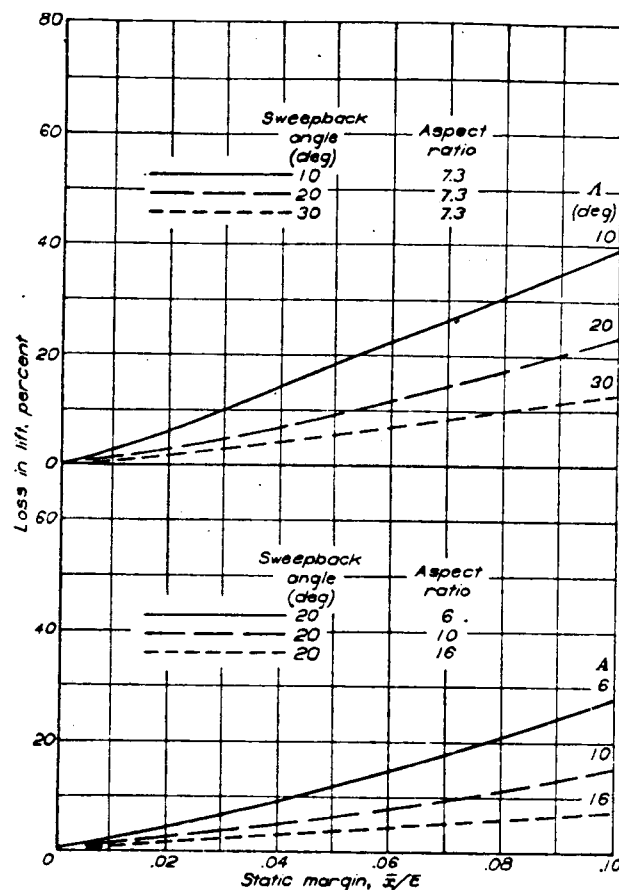


FIGURE 10.—Percent loss in lift caused by longitudinal control.  $c_a=0.30c$ ;  $\delta_a=21^\circ$ ;  $\frac{1}{\lambda}=4:1$ . Control consists of extending elevator span inboard from tip until the resulting pitching moment about the center of gravity is zero.

be deflected considerably more than that of a conventional airplane in order to produce the same changes in trim lift coefficient in flight. The elevator on tailless airplanes, being an integral part of the wing, must also operate at all angles of attack of the wing up to the stall. The elevator must therefore be balanced over a large range of angle of attack and deflection.

In order that push forces may be required to increase the airplane speed (from trim speed) and that pull forces may be required to reduce the airplane speed, the inherent up-floating tendencies of the elevator with increasing angle of attack must be reduced. The critical case for stick-force reversal (called elevator snatch) is that for neutral longitudinal stability (or zero static margin). If there is to be no stick-force reversal for this case, the variation of the elevator hinge moment with angle of attack must be zero or positive at all angles of attack throughout the flight range. When this condition is fulfilled, the elevator either remains stationary or floats down as the angle of attack is increased. Further discussion of this point may be found in reference 10.

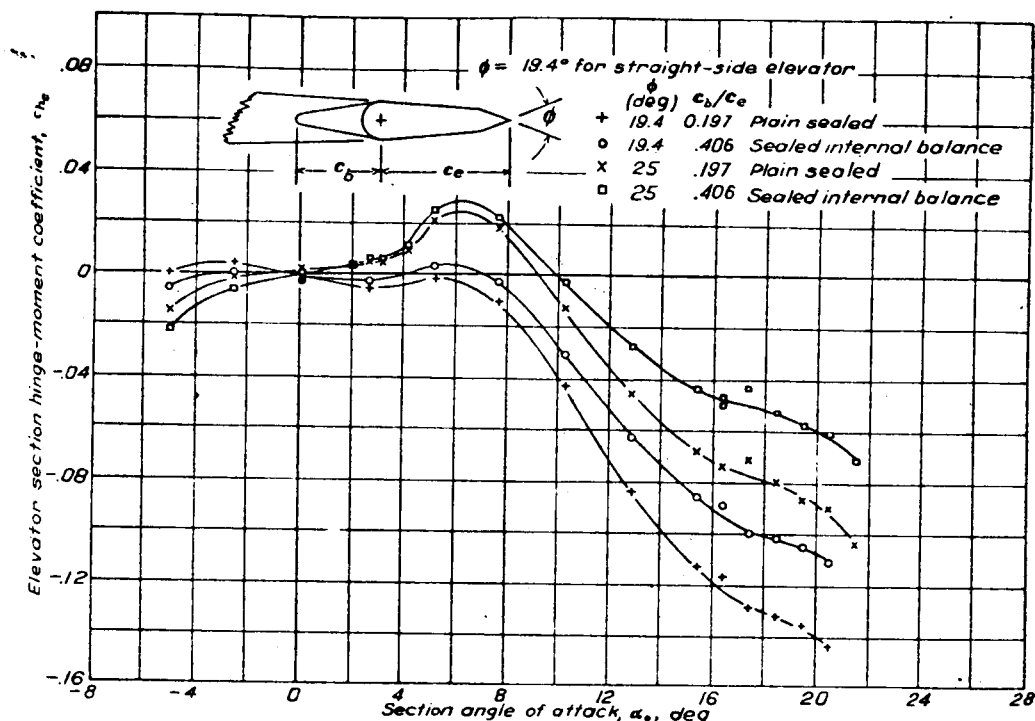


FIGURE 11.—Variation of section hinge-moment coefficient with angle of attack for 0.18c elevators on a modified NACA 65,3-018 airfoil.  $\delta_s = 0^\circ$ ;  $q = 59.4$  pounds per square foot.

**Types of control.**—A plain flap is unsuitable for use as an elevator on a tailless airplane mainly because it floats upward as the angle of attack of the wing is increased. In figure 11, the upfloating tendency of the flap is manifested by the increasingly negative flap hinge moments that are developed as the angle of attack is increased. Various balancing schemes have been proposed for reducing or eliminating the upfloating tendency of plain flaps but no aerodynamic balances are yet known that completely satisfy the design requirements. Several balance arrangements, however, show promise of being satisfactory in two-dimensional tests but have received no experimental verification in three-dimensional tests. A few of the proposals are discussed in the following paragraphs:

(1) *Bevels.*—Figure 11 presents the variation of elevator section hinge-moment coefficient with angle of attack at zero elevator deflection for straight-side and beveled elevators with and without internal balance vented at the hinge line. The curves indicate that the desired hinge-moment variation with angle of attack cannot be obtained with these arrangements of bevel and internal balance. Since the slopes for all elevators are nearly parallel at large angles of attack, it is not to be expected that favorable curves can be obtained either by further increasing the trailing-edge angle or by increasing the length of the internal balance vented at the hinge line.

Beveled elevators also affect the location of the wing aerodynamic center. The magnitude of the effect depends on the chord and span of the elevator. In general, the stick-

fixed aerodynamic center of the wing moves forward with an increase in trailing-edge angle, and the stick-free aerodynamic center of the wing moves backward with an increase in trailing-edge angle.

(2) *Special venting.*—It has been suggested that an internal balance be used which has a vent near the airfoil leading edge. Analysis of available data indicates that, at large angles of attack, however, this arrangement would have the same unfavorable characteristics as the internal-balance arrangements vented at the hinge line.

An analysis of pressure-distribution data indicates that an internal balance vented near the trailing edge of the airfoil would give the desired hinge-moment variation with angle of attack. The fact that the pressure changes in this region of the airfoil are small, however, appears to demand an internal balance of such length as to be impracticable.

(3) *Slots ahead of elevators.*—As the upfloating tendency inherent in all control surfaces at large angles of attack is caused by air-flow separation over the control surfaces, it has been proposed that slots be placed in the wing ahead of the elevator as one means of suppressing this effect. Very little research has been done on this particular scheme however and, at the present time, all that can be said is that it might be advantageous.

(4) *Automatically controlled tabs.*—Several rather mechanically complex types of balance have been proposed to prevent elevator-force reversals. Because a tab is normally a powerful means of changing elevator hinge moments, it has



been proposed to place a tab on the elevator and cause the tab to deflect upward in such a manner that the elevator floats down when the angle of attack is increased. The deflection of the tab would be controlled either by linking it to an internal balance, suitably vented, or by linking it to a free-floating spanwise portion of the elevator called a flipper. The flipper should be located along the span in a region where the stall is first manifested over the control surface. Two-dimensional characteristics of several such flipper-tab arrangements have been computed from section data, and the results are presented in figure 12. Some of the configurations result in hinge-moment slopes that are either zero or positive at all angles of attack. If similar characteristics could be obtained in three-dimensional flow, no stick-force reversal would occur for these combinations. The stick force could be controlled by adapting a spring either to the same tab or to an auxiliary tab.

(5) *Spoilers*.—The possibility of using a spoiler as an elevator has been suggested as a means of avoiding stick-force reversals. The loss in lift accompanying the production of a given pitching moment is greater with the spoiler control, however, than with the elevator control. Unpublished tests of rearwardly located spoilers on two different models confirm the fact that spoiler projections of less than  $0.01c$  produce negligible changes in lift. Such a spoiler is undesirable for longitudinal control because a small stick movement produces no change in trim, whereas a larger movement of the stick may produce large changes in trim and normal acceleration. The characteristics of spoilers can be controlled somewhat by adjusting the spoiler span and by incorporating special venting to the spoiler.

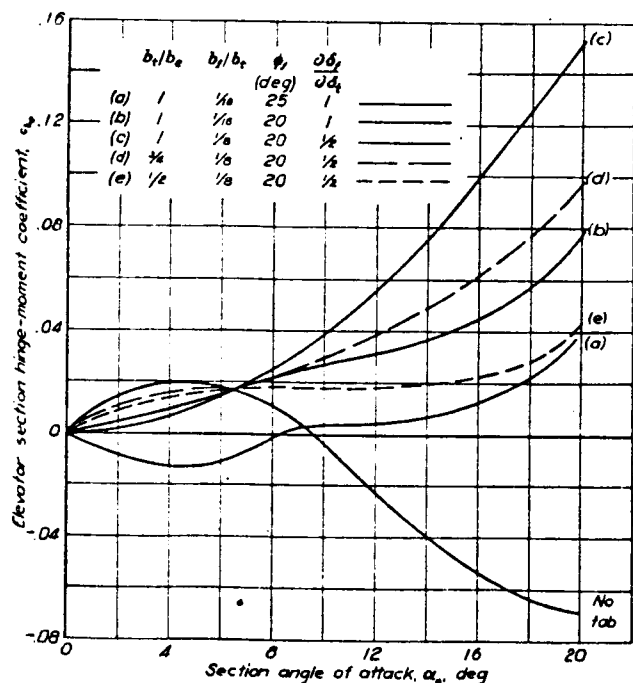


FIGURE 12.—Variation of  $c_{h_e}$  with  $\alpha$  at  $\delta = 0^\circ$  for various flipper-tab arrangements on a beveled elevator. Section data.

Inasmuch as the spoiler may be located ahead of an aileron, an upward deflection of the spoiler would cause the aileron to have an upfloating tendency and at the same time cause the ailerons to be less nearly balanced.

**Control for take-off.**—Under take-off conditions, the longitudinal control, besides supplying a pitching moment large enough to trim the wing at the lift coefficient corresponding to the ground angle of the airplane, may be required to supply the "additional" pitching moment necessary to counteract (1) the pitching moment of the weight of the airplane about the point of contact with the ground, (2) the pitching moment created by the friction force on the wheels, and (3) pitching moments arising from interference caused by the proximity of the airplane to the ground (references 11 and 12). In order to make certain that the airplane has adequate longitudinal control to compensate for these additional pitching moments arising during the take-off, the Army requirements for an airplane equipped with a tricycle landing gear state that the longitudinal control shall be powerful enough to pull the nose wheel off the ground at 80 percent of the take-off speed during operation off terrain where the coefficient of friction is  $1/10$  (reference 13). An idea of the magnitude of the "additional" pitching moment that the longitudinal control must supply to compensate for the extraneous effects associated with take-off may be obtained from figure 13.

Because of the short moment arm associated with the elevator of a tailless airplane, it is extremely difficult to

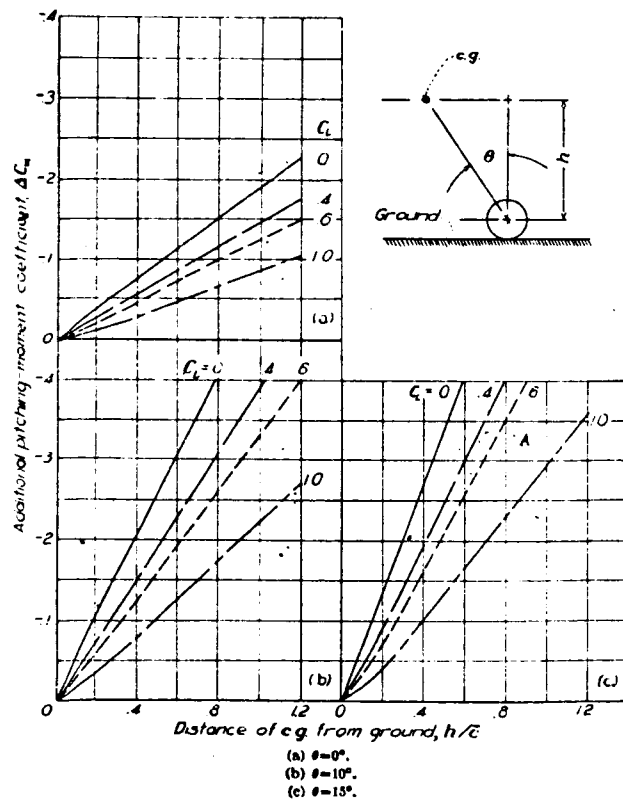


FIGURE 13.—Additional pitching-moment coefficient required to pull the nose wheel of a tailless airplane off the ground at 80 percent of the take-off speed. Assumed take-off  $C_L = 1.16$ .

design an elevator that can alone supply the pitching moments necessary to meet the Army take-off requirements; for example, point A spotted on figure 13(c) was computed for a typical tailless airplane. For this case, a pitching-moment coefficient of  $-0.335$  is needed to raise the nose wheel off the ground. The elevator effectiveness  $C_{m_\epsilon}$  for this airplane is only  $-0.003$  per degree, and thus the elevator cannot raise the nose wheel for take-off. In order to remedy this situation, it has been proposed to utilize the nose wheel as a jack to adjust the ground angle during the take-off run. If some scheme of this type is not provided, it appears likely that tailless airplanes may experience difficulty in raising the nose wheel off the ground at take-off if the landing-gear angle  $\theta$  is large, particularly with large static margins.

**Center-of-gravity range.**—On the basis of the longitudinal stability and control problems which have been discussed, it appears that the permissible range of center-of-gravity position compatible with satisfactory flight behavior is more critical for tailless airplanes than for conventional airplanes. If the static margin becomes negative, there is danger of encountering longitudinal instability either as a divergence from straight flight or as tumbling. If the static margin is too great, the elevator control may not be powerful enough to raise the nose wheel off the ground at take-off. Furthermore, if the static margin is large, the elevator deflection required to trim the airplane in level flight may seriously impair the efficiency of the wing with a consequent loss in performance of the airplane. At the present time, a range of ultimate static margin from  $0.02$  to  $0.08$  appears to be reasonable for tailless airplanes.

## LATERAL STABILITY AND CONTROL

### DIRECTIONAL STABILITY

Since the publication of reference 1, several models of tailless airplanes have been tested in the Langley free-flight tunnel. It has been verified from these tests that the amount of directional stability possessed by tailless airplanes should be as great as required on conventional airplanes if the same requirements regarding satisfactory flying qualities are to be adhered to. The value of the directional-stability parameter  $C_{n_\beta}$  recommended for conventional airplanes, is usually greater than  $0.001$  per degree. As evidenced from figure 14, however, models have been flown in the Langley free-flight tunnel and with a value of  $C_{n_\beta}$  of only one-third this amount although the best flying qualities of these models were obtained with values of  $C_{n_\beta}$  in excess of  $0.001$ .

The inherent aerodynamic characteristics of the wing alone have sometimes been tried as the source for directional stability. The amount of stability contributed by the wing depends on the wing plan form and the lift coefficient. The effect of the wing plan form does not appear large but more data are needed on this subject. The directional stability of the wing alone increases somewhat with lift coefficient. The directional stability at low angles of attack for the wing alone has generally been found to be inadequate although adequate stability may sometimes exist at high angles of attack.

A pusher propeller usually contributes a small degree of directional stability because of the stabilizing normal pro-

PELLER force. If, in addition, the pusher propeller is mounted behind a vertical tail surface, an additional increment in directional stability is realized from the vertical tail surface because of the effects produced by the inflow of air into the propeller.

The destabilizing effect of a fuselage or streamline nacelle on the directional stability has been discussed in reference 1. The destabilizing effect of the fuselage and nacelle of tailless airplanes is usually at least as great as the stabilizing effects contributed by the wing alone. It is therefore necessary on tailless airplanes to provide some method of supplying directional stability.

The provision of adequate directional stability for tailless airplanes is more difficult than for conventional airplanes because of the short longitudinal moment arm. A variety

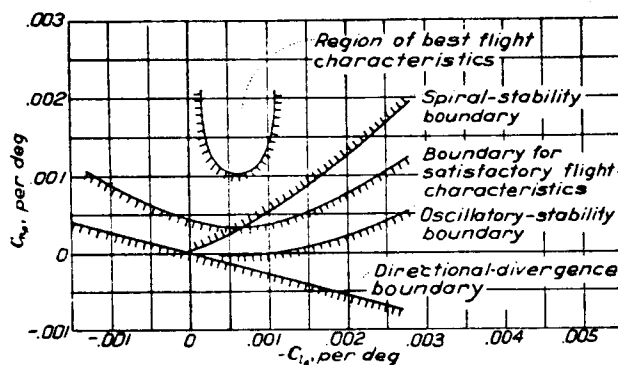


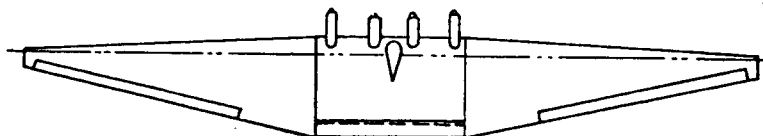
FIGURE 14.—Effect of directional stability  $C_{n_\beta}$  and dihedral effect  $C_{l_\beta}$  on flight characteristics as determined by tests of a model in Langley free-flight tunnel.  $C_L=1.0$ .

of fin arrangements and end plates of the type discussed in reference 1 has been tested on models of tailless airplanes in the Langley free-flight tunnel in an effort to improve the directional stability of specific models. A résumé of some of the more pertinent considerations that have evolved from these tests is given in the subsequent discussions.

(1) *Fins.*—It has been found that, for a tailless airplane having a straight wing, adequate directional stability can be provided by vertical tail surfaces located at the center section of the wing near the trailing edge (or on the fuselage if one is available). The size and number of vertical tails necessary for a specific design of course depends primarily on the degree of directional stability required. When multiple tails are used, it appears to be preferable to use as few tails of as high aspect ratio as possible because (a) fins of high aspect ratio are more effective than fins of low aspect ratio, (b) the interference effects between adjacent vertical fins are minimized, and (c) much of the fin is outside the relatively thick boundary layer on the upper rear surface of the wing.

If the tailless airplane has a swept-back wing, the usual practice is to place the vertical tail surfaces at the tips rather than at the center section in order to take advantage of the longer moment arm available. When vertical fins are placed at the wing tip extremities, however, the moment arm associated with the drag of the tip fin is so large (one-half the span) that the drag characteristics as well as the lift characteristics of the tip fins exert an influence on the directional stability. The relative contribution of the lift and drag of

Vertical tails tested on a tailless-airplane model



Plan view of tailless-airplane model

| Tail                | A                                                                                 | B                                                                                 |
|---------------------|-----------------------------------------------------------------------------------|-----------------------------------------------------------------------------------|
|                     |  |  |
| Area (percent $S$ ) | 5                                                                                 | 10                                                                                |
| Aspect ratio        | 4                                                                                 | 0.5                                                                               |
| $C_{D0}$            | 0.00010                                                                           | 0.00012                                                                           |

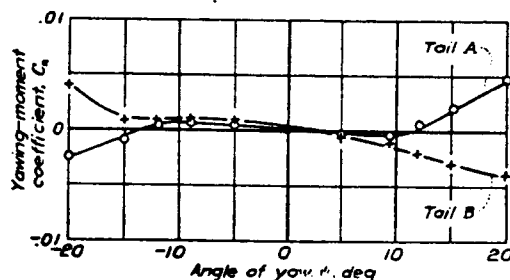


FIGURE 15—Directional stability characteristics of a tailless airplane equipped with toed-in and toed-out tip fins.

the tip fins to the directional stability of the airplane of course depends on their inherent aerodynamic characteristics. Some attention must accordingly be devoted to setting the initial angle of the tip fins.

If directional stability is to be obtained with tip fins of low aspect ratio (less than about 2), the tip fin must be set with some initial toe-in because of the large induced drag associated with lifting surfaces of low aspect ratio. When the airplane is yawed, the stabilizing moments generated by tip fins are produced by the large induced drag of the forward wing tip. If, on the other hand, directional stability is to be obtained with tip fins of moderate or high aspect ratio, the tip fins must be set with some initial toe-out. With toed-out tip fins, the stabilizing moments are generated by the outwardly directed lift as explained in reference 1. The stalling characteristics of the tip fins, moreover, are an important design consideration. When an airplane with toed-out tip fins is yawed to an angle sufficient to stall the rear tip fin, a large destabilizing moment is generated by the increased drag of the rear tip fin. On the other hand, when an airplane with toed-in tip fins is yawed to an angle sufficient to stall the forward tip fin, a large stabilizing moment is produced. The manner in which the stalling of toed-in and toed-out tip fins affects the directional stability of the airplane is illustrated in figure 15.

It has been suggested that the effectiveness of drag tip fins can be augmented by employing an airfoil section possessing aerodynamic characteristics similar to those shown in figure 16 for the NACA 4306 airfoil. In practice, the tip

fins are set at the correct angle of toe-in for zero lift in straight flight. When the airplane sideslips, the angle of attack of the leading tip fin is made more negative and thus causes a large increase in the profile-drag coefficient due to flow separation; whereas, at the same time, the angle of attack of the trailing tip fin is increased positively and thus causes only a relatively small increase in its profile-drag coefficient. Drag fins of this type have not been tested in flight. A lateral oscillation may possibly develop as a result of drag hysteresis, although such an effect has not been observed in tests of small-scale models.

The most effective tip fins tested in the Langley free-flight tunnel have been based on the profile-drag principle. Tip fins based on induced-drag principles have been somewhat less effective. The tip fins based on lift principles have been the least effective tested because of the short moment arm associated with the lift tip fins. The moment arm, however, is controlled by the angle of sweep so that, for wings with a large amount of sweepback, it may be feasible to design an effective lift tip fin. Central fins have generally been satisfactory, particularly if mounted on the end of a fuselage.

(2) *Turned-down wing tips*.—The amount of inherent directional stability possessed by a wing may be increased by turning down the wing tips; thus, in effect, the wing tips are made to function somewhat as lower-surface tip fins and the increased directional stability is manifested through the outward lift developed on the wing tips. The incorporation of positive dihedral angle on the wing, however, results in a decrease in directional stability because the lift of the wing

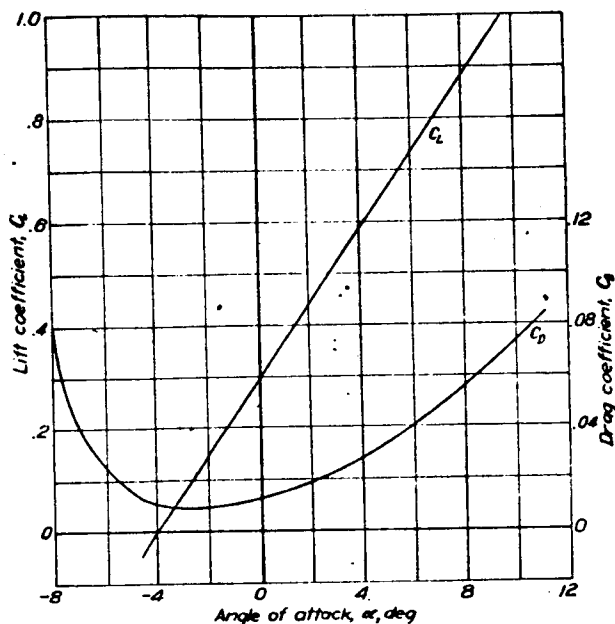


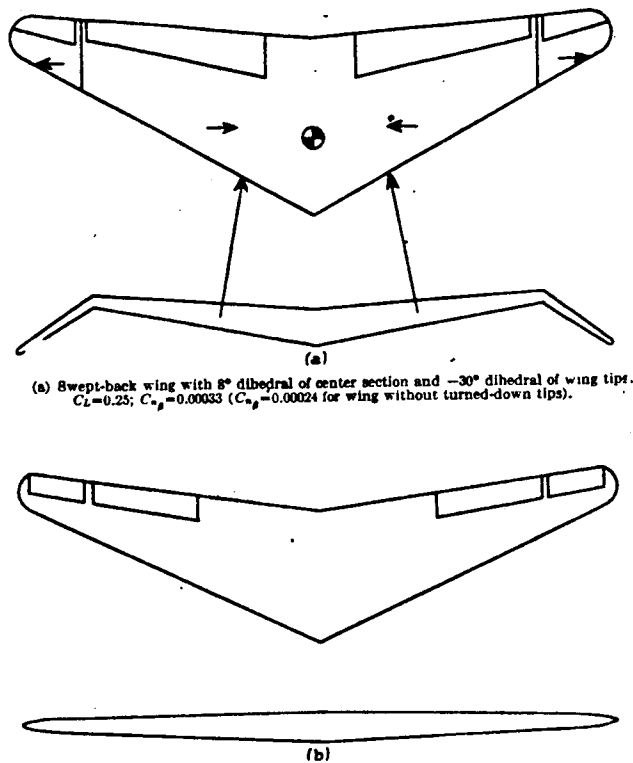
FIGURE 16.—Aerodynamic characteristics of NACA 4308 airfoil section.

itself is directed inward rather than outward (fig. 17). The destabilizing influence of a positive dihedral angle must be taken into account in computing the directional stability required of the wing tips. An examination of figure 17 indicates that the effects of the dihedral can practically nullify the effects of the turned-down wing tips. Turned-down wing tips are believed to be less satisfactory for securing directional stability than fins of the types previously discussed.

(3) *Automatic control.*—It has been suggested that a tailless airplane of very low directional stability with fixed controls could be flown satisfactorily if an automatic pilot were geared to the directional control in such a manner that when the airplane sideslipped the amount of directional control supplied would be sufficient to increase the effective value of  $C_{n_p}$ . Reference 1 includes the suggestion that the directional control could be linked with the aileron control in order to minimize the effects of adverse aileron y. v. It is believed that satisfactory flight behavior could be obtained with such automatic-stabilizing schemes although, at the present time, no flight investigations of such applications have been reported.

#### DIRECTIONAL CONTROL

The requirements of rudder control for tailless airplanes are essentially the same as for conventional airplanes. Rudder control is necessary to counteract the adverse yaw occurring during rolling maneuvers and to provide sufficient directional control to trim the airplane directionally at operation under asymmetric power conditions. At the



(a) Swept-back wing with 8° dihedral of center section and -30° dihedral of wing tips.  $C_L=0.25$ ;  $C_{n_p}=0.00033$  ( $C_{n_p}=0.00024$  for wing without turned-down tips).

(b) Swept-back wing with 0° dihedral.  $C_{n_p}=0.00027$  ( $C_L=0.3$ ) to 0.00055 ( $C_L=1.0$ ).

FIGURE 17.—Comparison of directional stability of two tapered swept-back wings

present time, the solution to the problem of creating adequate directional control rests primarily in reducing the yawing moment that such a control must overcome; thus, it is of particular advantage on tailless airplanes to locate the propellers as close as possible to the center line and to provide ailerons that create favorable yawing moments when deflected.

The provision of adequate directional control on a tailless airplane with rudders based on lift principles is difficult because of the small moment arm available for control. Computations have indicated that rudders based on lift principles alone generally are not able to counteract the yawing moments generated by severe asymmetric thrust conditions even if mounted at the tip of a swept-back wing. Lift rudders must also develop an appreciable side force because of the short moment arm. In order to compensate for this side force, the tailless airplane must be sideslipped or banked an appreciable amount because of its low lateral resistance. Some of the flight difficulties that may arise as a result of these circumstances are discussed in reference 14. Some use has been made, therefore, of directional control that is dependent upon drag characteristics because of the large moment arm which can be obtained by locating the drag directional control at the wing tip.

It appears possible to design a rudder based on drag principles utilizing a double split flap (brake flap) that could trim the yawing moments caused by asymmetric thrust conditions (fig. 18). It is cautioned, however, that split-flap rudders may generate undesirable rolling moments along with the yawing moments produced. This type of rudder may also affect the performance of the airplane if the drag increments necessary for control are very large. At the present time, specific designs of rudders of this type should be developed experimentally.

The use of propellers mounted in the wing tips has been proposed as a method for supplying directional stability and control. Such a system could, of course, be used easily

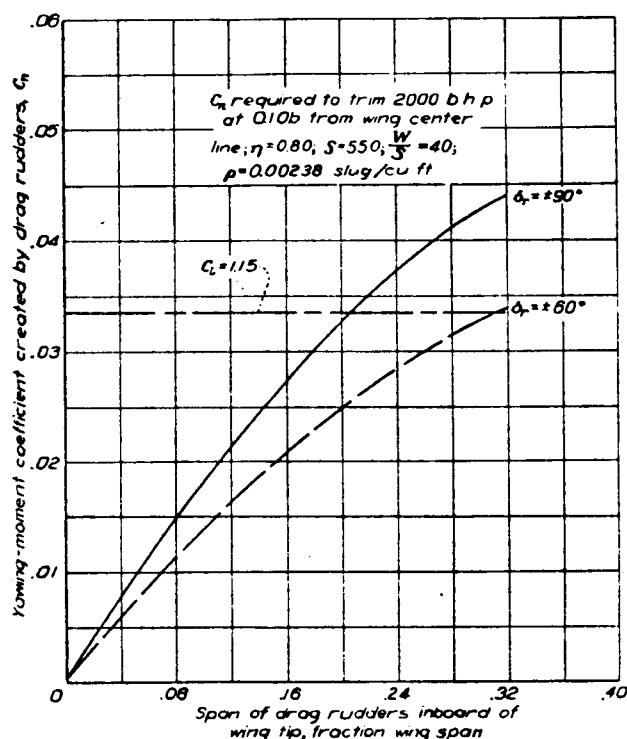


FIGURE 18.—Yawing moments created by double split perforated drag rudders mounted at wing tips.  $\eta = 0.80$ ;  $\eta$ , propulsive efficiency;  $\frac{W}{S}$ , wing loading, pounds per square foot.

with an automatic pilot. It is believed, however, that structural considerations may make such an arrangement impracticable at the present time.

#### DIHEDRAL

The requirements of dihedral for stability are essentially the same for a tailless airplane as for a conventional airplane. Computations of the type presented in references 15 and 16 and investigations conducted in the Langley free-flight tunnel (fig. 14 and reference 17) have indicated that, in the interest of lateral control and steadiness in gusty air, it is desirable to keep the effective dihedral angle small. The results of these investigations have indicated that, for satisfactory lateral stability, the effective dihedral angle should not exceed a value corresponding to  $-C_{l\beta} = 0.001$  per degree. This value of  $C_{l\beta}$  corresponds to a geometric dihedral angle of about  $5^\circ$

on a plain wing with no sweepback. It is noted that for a wing with no sweepback  $C_{l\beta}$  is practically independent of lift coefficient.

**Effect of sweepback.**—Systematic investigations to determine the effect of sweepback and taper on  $C_{l\beta}$  are being conducted. The limited data available at the present time indicate that the effective dihedral of a swept-back wing increases with angle of attack; it is thus advisable to use a geometric dihedral angle of about  $0^\circ$  in order that, at the higher lift coefficients, the effective dihedral does not exceed  $3^\circ$  or  $4^\circ$ . The increase in  $C_{l\beta}$  with angle of attack for a swept-back wing is not so detrimental as might first be supposed, however, because of the accompanying increase in weathercock stability. An empirical formula for estimating the effect of sweep on  $C_{l\beta}$  is discussed in reference 18.

**Effect of sweepforward.**—The effective dihedral of a swept-forward wing decreases as the angle of attack is increased. Some idea of the magnitude of the effect to be expected is given in reference 18. There is an indication also that the weathercock stability of a swept-forward wing may decrease with increase in angle of attack. This effect would make the attainment of lateral stability over a large range of angle of attack difficult. More information on swept-forward wings is needed, however, in order to evaluate these effects.

#### AILERON CONTROL

The aileron control of a tailless airplane presents no problems greatly different from those for conventional airplanes. An effort should be made, however, to avoid adverse aileron yawing moments, particularly if the directional stability is low, in order to minimize the sideslip developed during rolling maneuvers. Adverse aileron yawing moments can be minimized by uprigging both ailerons or by utilizing rotatable wing tips of the type previously described. In order to overcome the effects of adverse aileron yaw, it may be of advantage to employ a spring connection between the aileron and rudder control in a manner described in the section entitled "Tactical maneuvers." It is desirable also that no pitching moments be produced by the deflection of the ailerons because the ailerons have nearly the same moment arm as the elevators. It is necessary therefore to use ailerons with an equal up and down deflection.

**Spoiler control.**—The use of spoilers for ailerons on tailless airplanes has been advocated from time to time. If only upgoing spoiler projections are used, the pitching moments developed are prohibitive. A spoiler arrangement employing equal up and down projections would improve this condition but the data available are insufficient for evaluating conclusively the merits of such a system.

**Elevon control.**—For some tailless airplanes utilizing a swept-back wing, ailerons placed near the wing tips have been made to act also as elevators because the most effective position for both controls is near the wing tips and because larger-span lift flaps can be employed if the two controls are combined. Such an arrangement, called elevons, combines the design requirements of both aileron and elevator in one control and introduces additional problems.

The total effective deflection range for an elevon must be the sum of the ranges required for the aileron and elevator. The fact that the neutral position of the elevon may be at some upward deflection when it is functioning as an aileron can be utilized to a certain extent in reducing the aileron stick forces. With a large static margin, however, the full aileron deflection used with the large upward elevator deflection required at low speed may produce large pitching moments and small rolling moments because the upgoing elevon may stall. In order to improve this condition in some designs, the use of an auxiliary longitudinal trimming device called a pitch flap has been proposed. The pitch flap is located outboard of the aileron. With such a device, the lateral control could be obtained at low speeds by supplying most of the trim with the pitch flaps and thereby minimizing the upward deflection of the elevons. The elevons then would be deflected as ailerons over a greater linear range of the curve of rolling moment against deflection.

The conditions regulating the balance of an elevon for a typical large tailless airplane are indicated in figure 19. The ranges of values of  $C_{h_1}$  and  $C_{h_2}$  that satisfy the stipulated elevator and aileron requirements independently were evaluated by the methods given in references 10 and 19. The crosshatched region includes all values of  $C_{h_1}$  and  $C_{h_2}$  that satisfy simultaneously the stipulated elevator and aileron requirements. The elevon must be balanced over a much larger deflection range than either the elevator or aileron alone and, because of the increased deflection range required, greater physical limitations are imposed concerning the length of the internal balance that can be used. The considerations that have already been discussed in regard to controlling the upfloating tendency of the elevator with angle of attack also apply to the elevons.

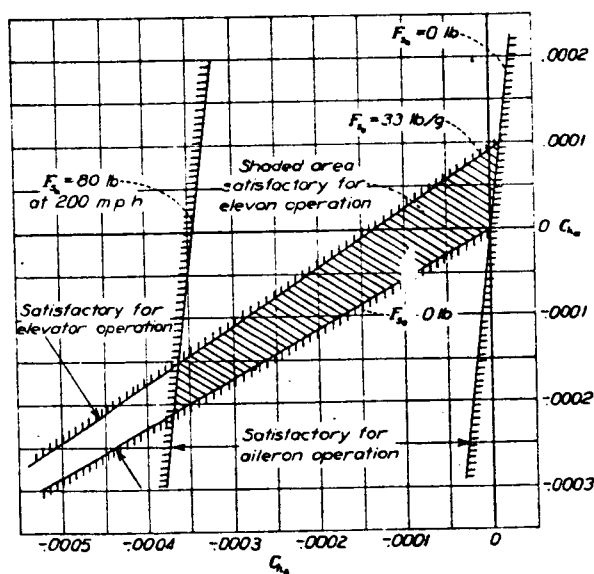


FIGURE 19.—Values of  $C_{h_1}$  and  $C_{h_2}$  required of elevons for a large tailless airplane.  $\frac{3}{2} = 0.03$ .

Trimming-tab operation of elevons differs from that for ailerons alone in that the tab must trim the hinge moment of each elevon to zero when it is desired to trim the airplane in roll in order to prevent the development of elevator stick forces. For ailerons alone, it is essential only that the tab cause one aileron hinge moment to balance that of the other aileron.

Section data from unpublished tests of an internally balanced, beveled, 0.18c elevon with 0.25c tab indicate that for angles of attack up to the stall a full-elevon-span tab deflected  $\pm 20^\circ$  could trim to zero the hinge moment of an elevon deflected  $\pm 25^\circ$ . The same data, however, indicate that little if any additional rolling moment can be produced by deflecting the elevon upward beyond  $25^\circ$  at large angles of attack.

#### DYNAMIC STABILITY

**Damping in yawing.**—For tailless airplanes, the rotational damping is invariably low on account of the reduction of the tail length. A comparison of the measured damping-moment coefficient due to yawing at a lift coefficient of 0.60 for various tailless airplanes and a conventional airplane is given in figure 20. The values were obtained by the free-oscillation method described in reference 20. The portion of  $C_{n_r}$  contributed by the wings can be estimated from the data in reference 21. It was pointed out in reference 1, however, that within the usual limits of dihedral and directional stability the damping of the lateral oscillations is generally greater than would be indicated from only the damping due to yawing velocity. Subsequent experience in flying tailless models in the Langley free-flight tunnel has substantiated this statement, and it appears that the small values of the damping parameter  $C_{n_r}$  associated with tailless airplanes will not be excessively detrimental to the flying qualities provided the directional stability of the airplane is adequate. The damping of the lateral oscillations is likely to be critical in the high-speed conditions because both  $C_{n_r}$  and the coupling between the yawing and rolling motion tend to diminish at the low angles of attack.

On account of the low values of  $C_{n_r}$  associated with tailless airplanes, some apprehension has existed concerning the large angles of sideslip that may be developed when the airplane is subjected to a disturbance of the type produced by asymmetric loss of thrust. There appear to be no data pertaining to the direct effect of  $C_{n_r}$  on a sideslipping motion of this type. The experience acquired in flying tailless-airplane models in the Langley free-flight tunnel has indicated that the effect of  $C_{n_r}$  is probably secondary to other parameters. The results presented in reference 15 indicate that the maximum amplitude of the sideslip oscillation is influenced markedly by the rolling moment due to the sideslip  $C_{l_b}$  and particularly by the yawing moment due to the sideslip  $C_{n_b}$ . Increasing either the directional stability or the dihedral reduces the magnitude of the sideslip generated by a yawing moment but the greatest reduction in sideslip appears to result from increasing the directional stability.

Power off:  $C_{n\dot{\beta}}$   $C_{n\dot{\gamma}}$   
 0.00070 -0.105  
 Power on — —0.290



Typical conventional airplane

$C_{n\dot{\beta}}$   $C_{n\dot{\gamma}}$   
 0.00072 -0.044



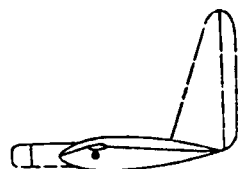
Straight-wing tailless fighter airplane

Tail toe-in  
 angle, deg  $C_{n\dot{\beta}}$   $C_{n\dot{\gamma}}$   
 5 0.00031 -0.014  
 10 .00033 -0.018  
 15 .00045 -0.025



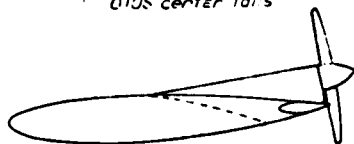
Tailless all-wing airplane with  
 0.05 tip tails

$C_{n\dot{\beta}}$   $C_{n\dot{\gamma}}$   
 0.00063 -0.017



Tailless all-wing airplane with  
 0.05 center tails

$C_{n\dot{\beta}}$   $C_{n\dot{\gamma}}$   
 0.00038 -0.017



Tailless all-wing airplane with  
 sweepback (no tails)

FIGURE 20.—Values of damping-in-yaw derivative  $C_{n\dot{\beta}}$  for a conventional airplane and various tailless airplanes.  $C_L = 0.60$ .

**Spinning.**—Tests conducted in the Langley 20-foot free-spinning tunnel have indicated that the steady-spin characteristics of tailless airplanes are essentially the same as for conventional airplanes. The control manipulations required for recovery from a steady spin, however, have been found to depend on the type and location of the control surface employed.

For tailless airplanes that have a vertical tail mounted at the rear of a fuselage, the application of rudder control would probably affect the spin in a manner similar to that for a conventional airplane because the vertical tail is not blanketed by the wing. If the vertical tail is located on the rear upper surface of the wing, however, the rudder control is likely to be ineffective because of the blanketing effect of the wing.

For tailless airplanes that have vertical tails at the wing tips, the application of rudder control would probably be effective for spin recovery, particularly if the rudder extends below the wing. For tailless-airplane designs without a fuselage, spin recovery has been found to be expedited by application of rolling moments against the spin. The ailerons therefore should be moved against the spin for best

recovery. The moments produced by trailing-edge drag rudders in the stalled range of angle of attack may be considerably different from those in the unstalled range. Some types of drag rudder have been found to produce appreciable pro-spin rolling moments when applied against the spin and therefore are not effective for recovery. It is recommended, therefore, that the aerodynamic characteristics in yaw for different rudder deflections of tailless-airplane designs that have drag rudders be obtained at angles of attack beyond the stall if the possibility of a spin appears likely. The results of these tests would facilitate the evaluation of the relative merits of alternative rudder designs. For a complete investigation of the recovery characteristics, spin tests of the model are usually required.

**Tactical maneuvers.**—The suitability of tailless airplanes for performing tactical maneuvers of the nature required for formation flying, bombing, and aerial combat has been the subject of frequent discussion. From considerations previously discussed, it appears that adequate directional stability is a necessary requirement for steadiness and ease of control. The fact that the lateral resistance associated with tailless airplanes is low may preclude the possibility of making flat turns with the rudder alone. At the present time, however, little information is available concerning the influence of side area on the lateral flying qualities of tailless airplanes. More research is needed on this subject, particularly in regard to the effects produced by the different directional-control devices mentioned in this paper.

The argument has been advanced that a pilot flying a fighter tailless airplane will experience difficulty in keeping his gunsight aligned with the target. It is believed however that, if the tailless airplane possesses the same directional stability and dihedral characteristics as are demanded for conventional airplanes, the controlled motions during the normal accelerated maneuvers should not differ appreciably from those of the conventional airplane.

In view of the likelihood that the successful tailless-airplane design may yet have lower directional stability than conventional airplanes, the effect of adverse aileron yaw on the pilot's aim may be more pronounced and in such cases a spring connection between the aileron and a trimming tab on the rudder may be necessary in order to satisfy the following criterion:

$$\frac{C_{n\dot{\beta}}}{C_{l\dot{\beta}}} > \frac{C_{n\dot{\gamma}}}{C_{l\dot{\gamma}}} < \frac{C_{n\dot{\gamma}}}{C_{l\dot{\gamma}}}$$

Such an arrangement should improve the steadiness of flight.

#### GENERAL CONSIDERATIONS OF TAILLESS AND CONVENTIONAL AIRPLANES

In recent years opinion has been divided as regards the relative adaptability of tailless and conventional airplanes for both fighter and bomber airplanes as evidenced by the variety of designs that have appeared. Some observations concerning the relative merits of tailless and conventional designs are offered here from consideration of the stability and control problems that have been discussed.

**Small airplanes.**—On account of the thin wing sections required for high speed, the volume enclosed by the wings of a small airplane is not large enough to carry all the load; consequently, it is necessary on small airplanes of either the tailless or conventional type to incorporate a fuselage or some other load-carrying element. It appears also that a vertical tail is necessary for directional stability. The difference between a small tailless airplane and a small conventional airplane, therefore, is essentially due to the suppression of a horizontal tail as a means of obtaining longitudinal stability and control. If the conventional airplane were permitted a reduction in maximum lift comparable with that tolerated on tailless airplanes, the tail size could be reduced considerably. With the small horizontal tail then allowable, the conventional airplane might have a performance comparable with that usually claimed for tailless airplanes without the restrictions attached to the longitudinal control.

**Large airplanes.**—For large airplanes having spans of 150 to 500 feet, the volume of the wing alone may be sufficient to enclose bulk or weight of an appreciable magnitude even with the thin wing sections required for high speed. There is little reason to suspect that conventional airplanes of equal span will have any less wing space available for cargo purposes than tailless airplanes. It appears, therefore, that the suppression of the fuselage as a load-carrying element is primarily a matter of airplane size rather than of type.

In spite of the suppression of the fuselage, however, a vertical tail may be necessary on any large airplane, particularly on bombers, if optimum directional stability and control are to be obtained. Some method must also be provided for obtaining longitudinal control. Whether the longitudinal control is obtained by elevons or by a horizontal tail located on a tail boom would seem to have a secondary influence on the ultimate performance to be expected. On the basis of the present knowledge of the stability and control characteristics of tailless airplanes, it appears desirable to make a comprehensive study of the comparative performance to be expected from tailless and conventional airplanes before proceeding further with stability and control studies.

LANGLEY MEMORIAL AERONAUTICAL LABORATORY,  
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,  
LANGLEY FIELD, VA., August 19, 1944.

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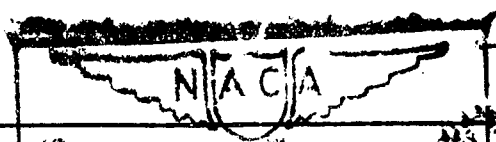
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# MEMORANDUM REPORT

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Report No. L6D12

WIND-TUNNEL TESTS OF A 1/4-SCALE MODEL  
OF THE BELL XS-1 TRANSONIC AIRPLANE  
(ARMY PROJECT MX-653)

I - LONGITUDINAL STABILITY AND CONTROL  
CHARACTERISTICS

By C. J. Donlan, W. B. Kemp,  
and E. C. Polhamus

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**NATIONAL ADVISORY COMMITTEE  
FOR  
AERONAUTICS**

1946

NACA LANGLEY MEMORIAL AERONAUTICAL LABORATORY

MEMORANDUM REPORT

for the

Air Materiel Command, Army Air Forces

~~CONFIDENTIAL~~ MR No. L6D12

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SUMMARY

Tests were made in the Langley 300 mile-per-hour 7- by 10-foot tunnel of a 1/4-scale model of the Bell XS-1 transonic airplane (Army Project MX-653) to determine its low-speed longitudinal stability and control characteristics. Pertinent longitudinal flying qualities expected of the XS-1 research airplane have been estimated from the results of these tests including estimated effects of compressibility likely to be encountered at speeds below the force break.

It appears that the static longitudinal stability and elevator-control power will be adequate, but that the elevator control-force gradient in steady flight will be undesirably low for all configurations. The control-force gradient in maneuvers will be below the required minimum of 3 pounds per g for all wing loadings below 90 pounds per square foot.

It is suggested that a centering spring be incorporated in the elevator control system of the airplane in order to increase the control-force gradient in steady flight and in maneuvers.

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## INTRODUCTION

This report contains the results of a longitudinal stability and control investigation conducted at the request of the Air Technical Service Command, Army Air Forces, on an unpowered 1/4-scale model of the Bell XS-1 transonic airplane. Results of a lateral stability and control investigation on the same model will be presented in a subsequent report.

The results pertain only to the tests of the basic configuration without simulation of the rocket propulsive system contemplated for this airplane.

For the evaluation of the longitudinal stability and control characteristics the investigation included stabilizer- and elevator-effectiveness tests, and elevator hinge-moment tests over an angle-of-attack range for the cruising and landing configurations. Tests were also made to determine the spoiler characteristics in pitch for the landing and cruising configurations. Included also are results for the wing alone, the wing-fuselage combination without empennage, stall patterns obtained by means of silk tufts attached to the surface of the model, and pressure distributions.

An analysis of the pertinent longitudinal flying qualities for steady flight and maneuvers is also presented.

## SYMBOLS

The system of axes used for the presentation of the data together with an indication of the sense of the positive forces and moments is presented in figure 1. Pertinent symbols are defined as follows:

|       |                                                 |
|-------|-------------------------------------------------|
| $C_L$ | lift coefficient ( $Lift/qS$ )                  |
| $c_l$ | section lift coefficient ( $Section\ lift/qS$ ) |
| $C_D$ | drag coefficient ( $Drag/qS$ )                  |
| $C_m$ | pitching-moment coefficient ( $M_{cg}/qSc'$ )   |

|                         |                                                                                                     |
|-------------------------|-----------------------------------------------------------------------------------------------------|
| $C_h$                   | hinge-moment coefficient ( $H/b'\bar{c}^2$ )                                                        |
| $C_R$                   | resultant force coefficient (Resultant force/ $qS$ )                                                |
| $M_{cg}$                | pitching moment about center of gravity                                                             |
| $H$                     | hinge moment                                                                                        |
| $q$                     | dynamic pressure, pounds per square foot ( $\rho v^2/2$ )                                           |
| $\rho$                  | mass density of air, slugs per cubic foot                                                           |
| $V$                     | free-stream velocity, feet per second                                                               |
| $S$                     | wing area, square feet                                                                              |
| $c$                     | section chord, feet                                                                                 |
| $c'$                    | wing mean aerodynamic chord, feet                                                                   |
| $b$                     | wing span, feet                                                                                     |
| $b'$                    | control-surface span along hinge line, feet                                                         |
| $\bar{c}$               | root-mean-square chord of elevator back of hinge line, feet                                         |
| $\alpha$                | angle of attack of fuselage center line, degrees                                                    |
| $\epsilon$              | angle of downwash, degrees                                                                          |
| $i_t$                   | stabilizer setting with reference to fuselage center line, degrees                                  |
| $\delta$                | control-surface deflection with reference to fixed surface, degrees                                 |
| $\delta_s$              | angle between the chord of spoiler and upper surface of wing                                        |
| $i_{t_0}, \delta_{e_0}$ | stabilizer and elevator angles, respectively, corresponding to neutral position of centering spring |
| $\epsilon_0$            | blocking correction factor                                                                          |

$\beta$  compressibility factor,  $\sqrt{1 - M_o^2}$   
 $\gamma$  angle of climb, degrees  
 $M$  Mach number  
 $M_o$  free-stream Mach number in tunnel  
 $P$  pressure coefficient,  $(\bar{p} - p_o)/q$   
 $p$  local static pressure, pounds per square foot  
 $p_o$  free-stream static pressure, pounds per square foot  
 $F_n$  elevator control-force gradient in maneuvers, pounds per g  
 $n$  normal acceleration of airplane in terms of g  
 $g$  acceleration due to gravity, 32.2 feet per second<sup>2</sup>  
 $W$  weight, pounds

## Subscripts:

$e$  elevator

$f$  flap

$t$  stabilizer

$s$  spoiler

$cr$  critical

$M$  measured value

$cor$  corrected value

$\delta_e$  } denote partial derivative of a coefficient with  
 $i_t$  } respect to  $\delta_e$ ,  $i_t$ ,  $\alpha$ . Example:  $C_{m i_t} = \frac{\partial C_m}{\partial i_t}$   
 $\alpha$  }

## MODEL AND APPARATUS

The Bell XS-1 airplane is a rocket-propelled airplane having the physical characteristics tabulated in table I. The operation of this airplane is characterized by extreme variations in speed, loading, and altitude. The airplane employs an irreversible power-driven stabilizer which is controlled by the pilot.

The 1/4-scale model is shown mounted in the Langley 300-mile-per-hour 7- by 10-foot tunnel in figures 2 and 3 and a three-view drawing of the model as tested is presented in figure 4. Details of the horizontal-tail assembly are presented in figure 5.

The Bell Aircraft Corporation supplied the model which was constructed of wood attached to metal reinforcing members with cycle-weld cement except for the all-metal control surfaces. The elevator, rudder, and flaps were 20-percent plain flaps, and the ailerons were 15-percent plain flaps. All controls were flat-sided from the hinge line to the trailing edge. The spoiler was located at the 30-percent-chord line of the wing, hinged at its leading edge, and had a chord of 3/4 inch.

Specific model configurations referred to in the text are as follows:

- (a) High-speed configuration
  - Flaps retracted
  - Landing gear retracted
- (b) Landing configuration
  - Flaps deflected 60°
  - Landing gear extended

Other configurations tested are described where encountered.

For all tests all flap and control-surface gaps were sealed with tape. For the hinge-moment measurements the elevator gaps were sealed by attaching the tape to the stabilizer on the positive pressure side and allowing the pressure to force the tape against the elevator, thereby eliminating flow through the gap. Since the airplane is to be a research airplane, it may or may not have sealed gaps for any given flight.

The strain-gage equipment used for measuring hinge moments was supplied and installed by the NACA. The pressure orifices used in the pressure-distribution tests were installed by the Bell Aircraft Corporation.

The tests were conducted in the Langley 300-mile-per-hour 7- by 10-foot tunnel. This is a closed rectangular tunnel of the return-flow type with a contraction ratio of 14 and is powered by a 1600-horsepower synchronous motor.

## TESTS AND RESULTS

### Test Conditions

Tests in the high-speed configuration were run at dynamic pressures of 88.4 and 165.6 pounds per square foot. Tests in the landing configuration with and without empennage were run at a dynamic pressure of 33.5 pounds per square foot. Pressure distributions were obtained at a dynamic pressure of 55.6 pounds per square foot. The corresponding approximate values of Mach number and Reynolds number (based on a chord of 1.202 feet) were as follows:

| q     | Mach number | Reynolds number, millions |
|-------|-------------|---------------------------|
| 33.5  | 0.15        | 1.18                      |
| 55.6  | .20         | 1.52                      |
| 88.4  | .25         | 1.94                      |
| 165.6 | .35         | 2.69                      |

The Reynolds number was computed using a turbulence factor of unity. The degree of turbulence of the tunnel is not known quantitatively but is believed to be small because of the high contraction ratio.

### Corrections

All data have been corrected for tare caused by the model-support struts. Jet-boundary corrections were computed as follows (reference 1):

$$\alpha = \alpha_M + 0.88C_{LM}$$

$$C_D = C_{DM} + 0.0128C_{LM}^2$$

For flaps undeflected:

$$C_m = C_{mM} + 0.0203C_{LM}$$

For flaps deflected:

$$C_m = C_{mM} + 0.0214C_{LM}$$

All force and moment coefficients were corrected for blocking by the following equation obtained from reference 2.

$$C_{cor} = C_M [1 - \epsilon_o (2 - M_o^2)]$$

where

$$\epsilon_o = \epsilon_{owing} + \epsilon_{ofuselage}$$

$$\epsilon_{owing} = \frac{0.00113}{\beta^3}$$

$$\epsilon_{ofuselage} = \frac{0.00625}{\beta^4}$$

An increment in drag coefficient of 0.00275 has been added to account for the horizontal buoyancy effected by the longitudinal static pressure gradient in the tunnel.

#### Presentation of Results

An outline of the figures presenting the results is given below:



| I. Aerodynamic characteristics            | Figure no. |
|-------------------------------------------|------------|
| A. Stabilizer tests                       | 6 to 8     |
| B. Elevator tests                         | 9 to 10    |
| C. Effect of flap deflection              | 11         |
| D. Stalling characteristics               | 12 to 14   |
| E. Downwash at tail                       | 15         |
| F. Spoiler tests                          | 16 to 20   |
| G. Pressure distribution                  | 21 to 27   |
| H. Critical Mach numbers                  | 28         |
| II. Stability and control characteristics |            |
| A. Static longitudinal stability          | 29 to 33   |
| B. Control forces in accelerated flight   | 34 to 38   |

## DISCUSSION

## Aerodynamic Characteristics

Lift and stalling characteristics.- The lift characteristics presented in figures 6 to 11 are summarized in the following table:

|                                 | $\delta_f = 0^\circ$ | $\delta_f = 30^\circ$ | $\delta_f = 60^\circ$ |
|---------------------------------|----------------------|-----------------------|-----------------------|
| $C_{L_{max}}$ (trimmed)         | 0.88                 | 1.10                  | 1.14                  |
| $C_{L_\alpha}$                  | .089                 | .081                  | .081                  |
| $\Delta C_L$ (due to flaps)     |                      |                       |                       |
| at trimmed $C_{L_{max}}$        | -----                | .22                   | .26                   |
| at untrimmed $\alpha = 0^\circ$ | -----                | .38                   | .49                   |
| Reynolds number, millions       | 1.94                 | 1.94                  | 1.18                  |

With flaps neutral, the model exhibited a center-section stall which progressed outward gradually and symmetrically as the angle of attack was increased (fig. 13). A similar stall progression was observed on the isolated wing (fig. 12). The gradual stall progression for this configuration is characterized by the "flap-top"

lift curve obtained in the flaps-neutral configuration. With flaps deflected, the initial stall occurred over the inboard sections but the stall spreading was somewhat more rapid than with flaps neutral (fig. 14). With the stabilizer set at  $0^\circ$ , the horizontal tail remained unstalled, with or without flaps deflected, up to an angle of attack of about  $11^\circ$ .

Horizontal-tail characteristics.— Mean values describing the effectiveness of the stabilizer and elevator are tabulated below:

| Parameter<br>Condition | $\frac{\partial C_m}{\partial i_t}$ | $\frac{\partial C_m}{\partial \delta_e}$ | $\frac{\partial a_t}{\partial \delta_e}$ | $\frac{\partial C_h}{\partial a_t}$ | $\frac{\partial C_h}{\partial \delta_e}$ |
|------------------------|-------------------------------------|------------------------------------------|------------------------------------------|-------------------------------------|------------------------------------------|
| $\delta_f = 0^\circ$   | -0.035                              | -0.0170                                  | 0.486                                    | -0.0024                             | -0.0104                                  |
| $\delta_f = 60^\circ$  | -.033                               | -.0175                                   | .530                                     | -.0026                              | -.0106                                   |

The values quoted for  $Ch_{a_t}$  were obtained from figure 18 and probably are associated with the conditions for which transition from a laminar to a turbulent boundary layer occurs at some rearward position along the chord. No fixed transition tests were made to substantiate this contention but it would appear, for example, that the sudden increase in elevator hinge-moment coefficient in figure 6 for  $i_t = 5^\circ$  at  $C_L = 0.3$ , or the corresponding increase at  $a_t = 2^\circ$  in figure 8 is typical of the effects produced on the hinge-moment characteristics of a control surface by a sudden forward shift in the transition point. Unfortunately, hinge-moment data were not taken at enough lift coefficients to establish similar regimes of critical hinge-moment behavior for other stabilizer settings.

Downwash at tail.— The average downwash angles at the tail have been determined from the stabilizer tests and are presented in figure 15 for  $\delta_f = 0^\circ$  and  $60^\circ$  together with the estimated downwash angles for the wing alone as estimated from the charts of reference 3 and corrected for wing twist. At the fuselage angle of attack corresponding to zero lift, flaps up, the downwash angle determined from the stabilizer tests is over  $2^\circ$ . The

estimated values for the wing alone indicate that there is only  $0.2^\circ$  downwash at zero lift. Therefore, the  $2^\circ$  determined from stabilizer tests ( $C_L = 0$ ) may be caused by the fuselage. The potential flow about a streamline body of revolution resembling the fuselage of the XS-1 is given in reference 4. By superimposing on the body of revolution a horizontal tail, similar to that on the XS-1, the average downwash, weighted on the basis of chord, was calculated to be about  $1.8^\circ$  at zero angle of attack. On the basis of this result it appears, therefore, that the downwash angle of  $2^\circ$  found at zero lift on the model with flaps up was very likely caused by the flow around the fuselage. The agreement was not quite so good for the flaps-down condition, probably because no account was taken of wing-fuselage interference on the flow around the after portion of the fuselage.

Spoiler characteristics.- Inasmuch as the spoilers on this airplane are intended to control the glide-path angle, the effect of the spoilers on the lift-drag ratio for the landing configuration has been evaluated and is presented in figure 20. These data are for untrimmed conditions but the relative significance of the data should be unaltered. It appears that an appreciable reduction in the lift-drag ratio is obtained only for large spoiler deflections ( $60^\circ$  or more) at lift coefficients above 0.75. For this case the maximum reduction in lift-drag ratio corresponds to a steepening of the glide-path angle from  $10.5^\circ$  to  $13.5^\circ$ .

It appeared from visual observations of tufts that, for small spoiler deflections ( $30^\circ$ ) at all angles of attack and for large spoiler deflections at small angles of attack, the spoilers caused a smoother flow over the flap than was observed without spoilers. At large spoiler deflections, increasing the lift coefficient above 0.75 caused a sudden separation of the flow over the flap, thereby giving rise to the rather irregular variation of lift coefficient and pitching moments associated with large spoiler deflections in the landing configuration.

Pressure distribution.- The locations of the pressure orifices are shown in figure 21. The fuselage pressure distributions are presented in figures 22 through 26, and the wing pressure distributions are presented in figure 27. These pressure distributions were run primarily to provide data for structural design and, although they may be sufficient for that purpose, insufficient points were taken to

predict accurately the critical Mach number of the wing. According to the data, the maximum low-speed pressure coefficient, at an angle of attack of  $0.1^\circ$  ( $C_L = 0.19$ ), corresponds to a critical Mach number of 0.67 which is rather low compared to the anticipated section value of 0.74. Part of this difference in critical Mach number may possibly be attributed to the fact that, when the wing lift coefficient is 0.19, the section lift coefficient near the root is greater than 0.19 because of the wing washout and wing-fuselage interference. For these reasons, it appeared desirable to prepare figure 28 in which are presented the theoretical critical Mach number curve as evaluated from reference 5, and the force break curve as estimated from actual lift force break data as presented in reference 6. The estimated force break curve presented in figure 28 represents the approximate limit to which the methods of correcting low Mach number wind-tunnel data can be applied. This estimate agrees qualitatively with the data in reference 7 in that the limiting curve is in the region where the first large changes in trim lift coefficient with Mach number were observed. mod  
larger  
than 1

### Stability and Control Characteristics

Static longitudinal stability.-- The stick-fixed and stick-free neutral points for both the cruising and landing configurations are presented in figure 29 and, on the basis of low Mach number tests, it appears that the static longitudinal stability is adequate. The average static margins for these conditions are presented in the following table:

|                       | Static margin, percent M.A.C. |            |
|-----------------------|-------------------------------|------------|
|                       | Stick fixed                   | Stick free |
| $\delta_f = 0^\circ$  | 9                             | 6          |
| $\delta_f = 60^\circ$ | 10                            | 7          |

Although  $\frac{dC_m}{dC_L}$  defines the static margin in level flight, a more general form that holds for any flight

attitude is  $\frac{dC_m}{dC_R}$ . Since  $\frac{dC_m}{dC_R}$  equals  $\frac{dC_m}{dC_L} \times \cos \gamma$ , the static margin for any given angle of climb may be obtained by multiplying the static margins  $\left(\frac{dC_m}{dC_L}\right)$  presented in the preceding table by  $\cos \gamma$  if low speeds ( $M = 0.35$ ) and negligible power effects are assumed.

Inasmuch as climb angles of  $45^\circ$  are expected for this airplane, the reduction in stability due to climb may be appreciable and should not be overlooked.

Elevator control in steady flight.- The variation of elevator control force with speed for different wing loadings and flap configurations is presented in figure 30. No allowance was made in these particular computations for compressibility effects, but it is believed that the magnitude indicated will reflect the characteristics of the airplane up to the force break (fig. 28). The influence of the ground was estimated by the method of reference 8. It appears that the static longitudinal stability and elevator-control power are adequate, but that the elevator control-force gradient in steady flight is undesirably low for all configurations. Inasmuch as an unbalanced elevator is employed, the low control forces can be attributed to the unusually small size of the elevator. The low control forces are particularly objectionable in the landing condition (fig. 30(c)) where even a moderate amount of friction in the control system would probably mask all sense of control feel. Inasmuch as the stick travel required to trim the airplane throughout the speed range is also small (approx. 1 in.), it would appear very desirable to increase the elevator control-force gradient in steady flight.

Elevator control in steady flight with centering spring.- It is suggested that a sense of control feel could be provided by incorporating in the elevator control system a centering spring. Such a spring would supply a hinge moment proportional to the elevator deflection from its neutral position with spring attached, and thus would effectively increase the magnitude of the restoring parameter,  $Ch_\delta$ . (However, inasmuch as the spring supplies a constant hinge moment for a given control deflection, the hinge-moment coefficient will decrease with increasing speed.) The spring could be attached either to the stabilizer or to the fuselage, as illustrated schematically

in figure 31. The effect of such a spring on the control-force gradient in steady flight is illustrated in figure 32 for the high wing loading condition with flaps retracted. For this application (fig. 32) (1) the neutral position of the elevator with a spring attached to the stabilizer was adjusted to provide zero stick force at the same speed for which zero stick force had been attained for the elevator without the spring, and (2) the spring stiffness was dictated by the requirement that control forces be of the order of 15 to 20 pounds at the elevator deflection required for minimum speed. The latter requirement was satisfied by a spring of a stiffness such as to require a force of about 1.8 pounds per degree stick deflection. The effect of this spring on the control forces in the landing configuration is illustrated in figure 30(c). Here the spring was adjusted to provide trim at some arbitrary speed in order to compare results for the same stabilizer setting. However, inasmuch as this airplane has an adjustable stabilizer, the spring need only be adjusted for trim in high-speed flight, and the stabilizer adjusted to provide trim throughout the speed range. The variation of stabilizer angle and elevator angle necessary to produce zero stick force and trim at various lift coefficients for two wing loadings for either of the arrangements shown in figure 31 is given in figure 33. For both arrangements, the stabilizer was adjusted to produce trim condition with zero elevator deflection at  $CL = 0.124$ . Inasmuch as the minimum variation of stabilizer and elevator angles is obtained with the spring attached to the stabilizer (fig. 31(a)), this arrangement is preferable.

Elevator control in accelerated flight. - The effect of the elevator hinge-moment parameters on the elevator control-force gradient in maneuvers is illustrated in figure 34 for flight at 35,000 feet with a wing loading of 64.7 pounds per square foot. Experimental values of the parameters from the wind-tunnel tests are indicated on the diagram along with the estimated effect of the small mass-balancing horns on the actual elevator. Included in figure 34 are estimates of the effects of Mach number on the stick-force gradients, as based on the theoretical Mach number effects given in figure 35 (determined by use of reference 9) and on an assumed

forward shift of the neutral point of 3 percent mean aerodynamic chord which could be caused by the effects of compressibility on the wing and tail lift-curve slope, and on the downwash behind the wing. Although the theoretical effect of compressibility on the slope of the lift curve agrees well with available experimental data, the experimental variation of hinge-moment parameters with Mach number is apparently affected in a manner not known at present, and hence by factors not considered in the present theory. Experimental effects of Mach number have shown the variation to be smaller and sometimes in the opposite direction from that predicted by theory, and it is believed, therefore, that the variation used here indicates a somewhat larger reduction in stick-force gradient with speed than may be experienced in flight. It is pertinent, however, to note that variations in hinge-moment parameters have very little effect on the stick-force gradients for this design (fig. 34). It is recognized that the forward movement of the neutral point with Mach number in flight (below the force break speed) may be less than that assumed in this calculation, but the present computations probably represent the minimum stick forces that would be encountered at this Mach number. The estimated values of the hinge-moment parameters for a Mach number of 0.75 are also indicated on the curves. The influence of wing loading and altitude on the elevator control-force gradient is illustrated in figure 36, and the effect of center-of-gravity position is shown in figure 37. The extremely low control-force gradient - of the order of 1 pound per g - with a wing loading of 35 pounds per square foot should be noted. The subsonic compressibility effects observed in figure 35 would result in an even smaller gradient at high speeds. This condition is particularly objectionable because of the small amount of elevator required to produce high accelerations at high speeds (fig. 37).

Elevator control in accelerated flight with centering spring. - The effect on the control-force gradient, in maneuvers, of the centering spring previously discussed (fig. 31(a)) depends on dynamic pressure in the manner illustrated in figure 38. For flight at a specific dynamic pressure, the values of  $F_n$  given in this figure can be added directly to the control-force gradients presented in figure 36. As an example, the effect of the spring on the control-force gradient at sea level and at 35,000 feet has been computed for a speed of 572 miles per hour, and the results presented in

figure 36. In order to show the magnitude of the variation of the control-force gradient with speed, the variation has been computed for flight at sea level for a wing loading of 104 pounds per square foot, and the results are given in figure 32. The large values of the control-force gradient in maneuvers at the low speeds need not be troublesome inasmuch as the maximum load factor that can be developed at these speeds is small. On the other hand, the increments in the control-force gradient provided at the high speeds are very desirable. It will be noted, however, that at the lowest wing loadings apt to be encountered, the control-force gradient at high speeds, even with a spring in the system, will probably be of the order of the minimum acceptable value of 3 pounds per g, so that it is essential that friction on the control system be kept to a minimum.

### CONCLUSIONS

Based on low-speed wind-tunnel tests of a 1/4-scale model the following conclusions regarding the longitudinal stability and control of the Bell XS-1 transonic airplane have been reached.

1. The static margin is expected to be adequate for all level-flight conditions.
2. The elevator control-force gradient in steady flight will be undesirably low for all configurations.
3. The control-force gradient in maneuvers will be below the required minimum of 3 pounds per g for all wing loadings below 90 pounds per square foot.
4. It is suggested that a centering spring be incorporated in the elevator control system of the airplane in order to increase the control-force gradient in steady flight and in maneuvers.



5. The spoilers will be ineffective in controlling the glide path angle for spoiler deflections less than  $60^\circ$ .

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TABLE I

## PHYSICAL CHARACTERISTICS OF THE XS-1 RESEARCH AIRPLANE

## Power:

Four rocket units grouped in rear of fuselage, each capable of delivering 1500 pounds thrust.

## Wing loading:

|                              |     |
|------------------------------|-----|
| Take-off, lb/sq ft . . . . . | 103 |
| Landing, lb/sq ft . . . . .  | 40  |

## Center-of-gravity position (all conditions),

|                        |    |
|------------------------|----|
| percent M.A.C. . . . . | 25 |
|------------------------|----|

## Wing:

|                                                    |                   |
|----------------------------------------------------|-------------------|
| Area, sq ft . . . . .                              | 130               |
| Span, ft . . . . .                                 | 28                |
| Mean aerodynamic chord, in. . . . .                | 57.71             |
| Aspect ratio . . . . .                             | 6                 |
| Root and tip sections . . . . .                    | 651-110 (a = 1.0) |
| Setting (root chord to thrust line), deg . . . . . | 2.5               |
| Setting (tip chord to thrust line), deg . . . . .  | 1.5               |
| Flap chord, percent wing chord . . . . .           | 20                |

## Aileron:

|                                                   |       |
|---------------------------------------------------|-------|
| Area (one aileron) behind hinge line, sq ft . . . | 3.15  |
| Span, ft . . . . .                                | 5.8   |
| Chord, percent wing chord . . . . .               | 15    |
| Root-mean-square chord, ft . . . . .              | 0.565 |

## Horizontal tail:

|                                                |       |
|------------------------------------------------|-------|
| Total area, sq ft . . . . .                    | 26.0  |
| Span, ft . . . . .                             | 11.4  |
| Aspect ratio . . . . .                         | 5     |
| Root-mean-square chord of elevator, ft . . . . | 0.464 |

## Vertical tail:

|                                                |       |
|------------------------------------------------|-------|
| Total area (not including dorsal), sq ft . . . | 25.6  |
| Rudder span, ft . . . . .                      | 6.58  |
| Root-mean-square chord of rudder, ft . . . .   | 0.798 |

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## TABLE I - Concluded

## PHYSICAL CHARACTERISTICS - Concluded

## Control movement:

|                                                   |                   |
|---------------------------------------------------|-------------------|
| Total stick movement (fore and aft), ft . . . . . | 1.5               |
| Elevator movement, deg . . . . .                  | 15 up and 10 down |
| Length of stick, ft . . . . .                     | 1.8               |
| Rudder pedal movement, in. . . . .                | 3.25 fore and aft |
| Rudder movement, deg . . . . .                    | 15 left and right |
| Wheel movement, deg . . . . .                     | 60 left and right |
| Wheel diameter, in. . . . .                       | 13                |
| Aileron movement, deg . . . . .                   | 12 up and down    |
| Stabilizer movement, deg . . . . .                | 5 up and 10 down  |

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## FIGURE LEGENDS

Figure 1.- System of axes and control-surface hinge moments and deflections. Positive values of forces, moments, and angles are indicated by arrows. Positive values of tab hinge moments and deflections are in the same directions as the positive values for the control surfaces to which the tabs are attached.

Figure 2.- 1/4-scale model of the Bell XS-1 airplane mounted in the 300 mph 7- by 10-foot wind tunnel, high speed configuration.

Figure 3.- 1/4-scale model of the Bell XS-1 airplane mounted in the 300 mph 7- by 10-foot wind tunnel, landing configuration.

Figure 4.- Three view drawing of 1/4-scale model of the Bell XS-1 airplane.

Figure 5.- Drawing of the horizontal tail of the 1/4-scale model.

Figure 6.- Effect of stabilizer setting on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane. Landing gear retracted,  $\delta_f = 0^\circ$ ,  $\delta_e = 0^\circ$ ,  $\psi = 0^\circ$ ,  $q = 165.6$  lb/sq ft.

Figure 6.- Concluded.

Figure 7.- Effect of stabilizer setting on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane. Landing gear extended,  $\delta_e = 0^\circ$ ,  $\delta_f = 60^\circ$ ,  $q = 33.5$  lb/sq ft.

Figure 7.- Concluded.

Figure 8.- Variation of elevator hinge-moment coefficient with tail angle of attack for the 1/4-scale model of the XS-1 airplane,  $\delta_e = 0^\circ$ .

Figure 9.- Effect of elevator deflection on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane;  $\delta_f = 0^\circ$ ;  $i_t = 0^\circ$ ;  $\psi = 0^\circ$ ;  $q = 165.6$  lb/sq ft.

Figure 9.- Concluded.

## FIGURE LEGENDS - Continued

Figure 10.- Effect of elevator deflection on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane,  $\delta_f = 60^\circ$ ,  $i_t = 0^\circ$ ,  $\psi = 0^\circ$ ;  $q = 33.5$  lb/sq ft.

Figure 10.- Concluded.

Figure 11.- Effect of flap deflection on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane,  $\delta_e = 0^\circ$ ,  $i_t = 0^\circ$ ,  $\psi = 0^\circ$ .

Figure 11.- Concluded.

Figure 12.- Tuft studies of the 1/4-scale model of the XS-1 isolated wing;  $\delta_f = 0^\circ$ ;  $\psi = 0^\circ$ ;  $q = 88.4$  lb/sq ft.

Figure 13.- Tuft studies of the 1/4-scale model of the XS-1 airplane;  $\delta_f = 0^\circ$ ;  $\psi = 0^\circ$ ;  $q = 88.4$  lb/sq ft.

Figure 14.- Tuft studies of the 1/4-scale model of the XS-1 airplane;  $\delta_f = 60^\circ$ ;  $\psi = 0^\circ$ ;  $q = 33.5$  lb/sq ft.

Figure 15.- Average downwash angles at the horizontal tail of the 1/4-scale model of the Bell XS-1 airplane.

(a)  $\delta_f = 0^\circ$ ; landing gear retracted.

Figure 15.- Concluded.

(b)  $\delta_f = 60^\circ$ ; landing gear extended.

Figure 16.- Effect of stabilizer setting on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane landing gear extended;  $\delta_f = 60^\circ$ ;  $\delta_s = 30^\circ$ ;  $\delta_e = 0^\circ$ ;  $q = 33.5$  lb/sq ft;  $\psi = 0^\circ$ .

Figure 16.- Concluded.

Figure 17.- Effect of stabilizer setting on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane; landing gear extended;  $\delta_f = 60^\circ$ ;  $\delta_e = 0^\circ$ ;  $\delta_s = 60^\circ$ ;  $q = 33.5$  lb/sq ft;  $\psi = 0^\circ$ .

Figure 17.- Concluded.

## FIGURE LEGENDS - Continued

Figure 18.- Effect of stabilizer setting on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane; landing gear extended;  $\delta_f = 60^\circ$ ;  $\delta_e = 0^\circ$ ;  $\delta_s = 90^\circ$ ;  $q = 33.5$  lb/sq ft;  $\psi = 0^\circ$ .

Figure 18.- Concluded.

Figure 19.- Effect of spoiler deflection on the aerodynamic characteristics in pitch of the 1/4-scale model of the XS-1 airplane; landing gear retracted;  $\delta_f = 0^\circ$ ;  $i_t = 0^\circ$ ;  $\delta_e = 0^\circ$ ;  $q = 88.4$  lb/sq ft;  $\psi = 0^\circ$ .

Figure 19.- Concluded.

Figure 20.- Effect of spoiler deflection on the lift-drag ratio of the XS-1 airplane;  $\delta_f = 60^\circ$ ;  $\delta_e = 0^\circ$ ;  $i_t = 0^\circ$ ;  $q = 33.5$  lb/sq ft.

Figure 21.- Location of pressure orifices on the 1/4-scale model of the XS-1 airplane.

Figure 22.- Effect of angle of attack on the pressure distribution on the top of the fuselage of the 1/4-scale model of the XS-1 airplane,  $\psi = 0^\circ$ ,  $q = 55.6$  lb/sq ft;  $M = 0.20$ .

Figure 23.- Effect of angle of attack on the pressure distribution around the fuselage at station 11.75 of the 1/4-scale model of the XS-1 airplane;  $\psi = 0^\circ$ ;  $q = 55.6$  lb/sq ft;  $M = 0.20$ .

Figure 24.- Effect of angle of attack on the pressure distribution around the fuselage at station 46.2 of the 1/4-scale model of the XS-1 airplane;  $\psi = 0^\circ$ ;  $q = 55.6$  lb/sq ft;  $M = 0.20$ .

Figure 25.- Effect of angle of attack on the pressure distribution along row A of the fuselage of the 1/4-scale model of the XS-1 airplane;  $\psi = 0^\circ$ ;  $\delta_f = 0^\circ$ ;  $q = 55.6$  lb/sq ft;  $M = 0.20$ .

Figure 26.- Effect of angle of attack on the pressure distribution along row B of the fuselage of the 1/4-scale model of the XS-1 airplane;  $\delta_f = 0^\circ$ ;  $\psi = 0^\circ$ ;  $q = 55.6$  lb/sq ft;  $M = 0.20$ .

## FIGURE LEGENDS - Continued

Figure 27.- Effect of angle of attack on the pressure distribution over the wing of the 1/4-scale model of the XS-1 airplane;  $\delta_f = 0^\circ$ ;  $\psi = 0^\circ$ ;  $q = 55.6$  lb/sq ft;  $M = 0.20$ .

(a) Station 7.75.

Figure 27.- Concluded.

(b) Station 27.19.

Figure 28.- Variation of critical, force break and flight Mach numbers with lift coefficient.

Figure 29.- Variation of neutral points with lift coefficient for the 1/4-scale model of the XS-1 airplane.

(a) Stick-fixed.

Figure 29.- Concluded.

(b) Stick-free.

Figure 30.- Variation of elevator stick force and position for trim with speed at sea level.

(a)  $W/S = 104$  lb/sq ft.

Figure 30.- Continued.

(b)  $W/S = 64.7$  lb/sq ft.

Figure 30.- Concluded.

(c)  $W/S = 40$  lb/sq ft.

Figure 31.- Schematic arrangements for spring loaded elevators.

Figure 32.- Variation of elevator forces and gradients with speed for the XS-1 airplane;  $W/S = 104$  lb/sq ft;  $\delta_f = 0^\circ$ ;  $i_t = 2.74^\circ$ ; altitude = sea level;  $\delta_{e_0} = 0.6^\circ$ .



## FIGURE LEGENDS - Concluded

Figure 33.- Stabilizer and elevator angles required for zero stick force and trim for various lift coefficients;  $\delta e_0 = -0.1^\circ$ ;  $i_{t_0} = 2.92^\circ$ .

(a)  $\delta_f = 0^\circ$ ;  $W/S = 104$  lb/sq ft.

Figure 33.- Concluded.

(b)  $\delta_f = 60^\circ$ ;  $W/S = 40$  lb/sq ft.

Figure 34.- Variation of elevator control force gradient with hinge moment parameters and speed;  $W/S = 64.7$  lb/sq ft; altitude = 35,000 feet.

Figure 35.- Effect of Mach number on the aerodynamic characteristics of the XS-1 airplane, determined by method of reference 9.

Figure 36.- Variation of elevator control force gradient with wing loading and altitude.

Figure 37.- Variation of elevator control force and deflection gradients with center of gravity position,  $W/S = 104$  lb/sq ft;  $i_t = 0^\circ$ ;  $C_L = 0.1$ ; load factor = 6; altitude = sea level.

Figure 38.- Variation of  $\Delta F_n$  due to the spring with dynamic pressure.

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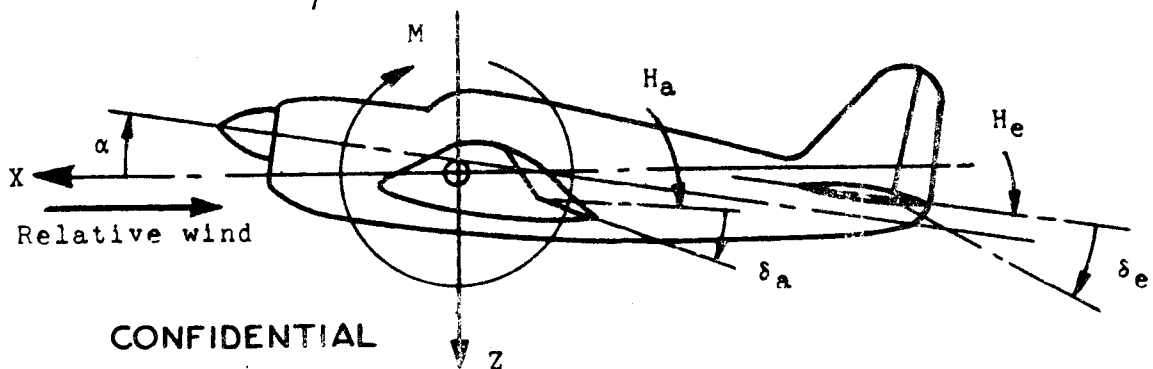
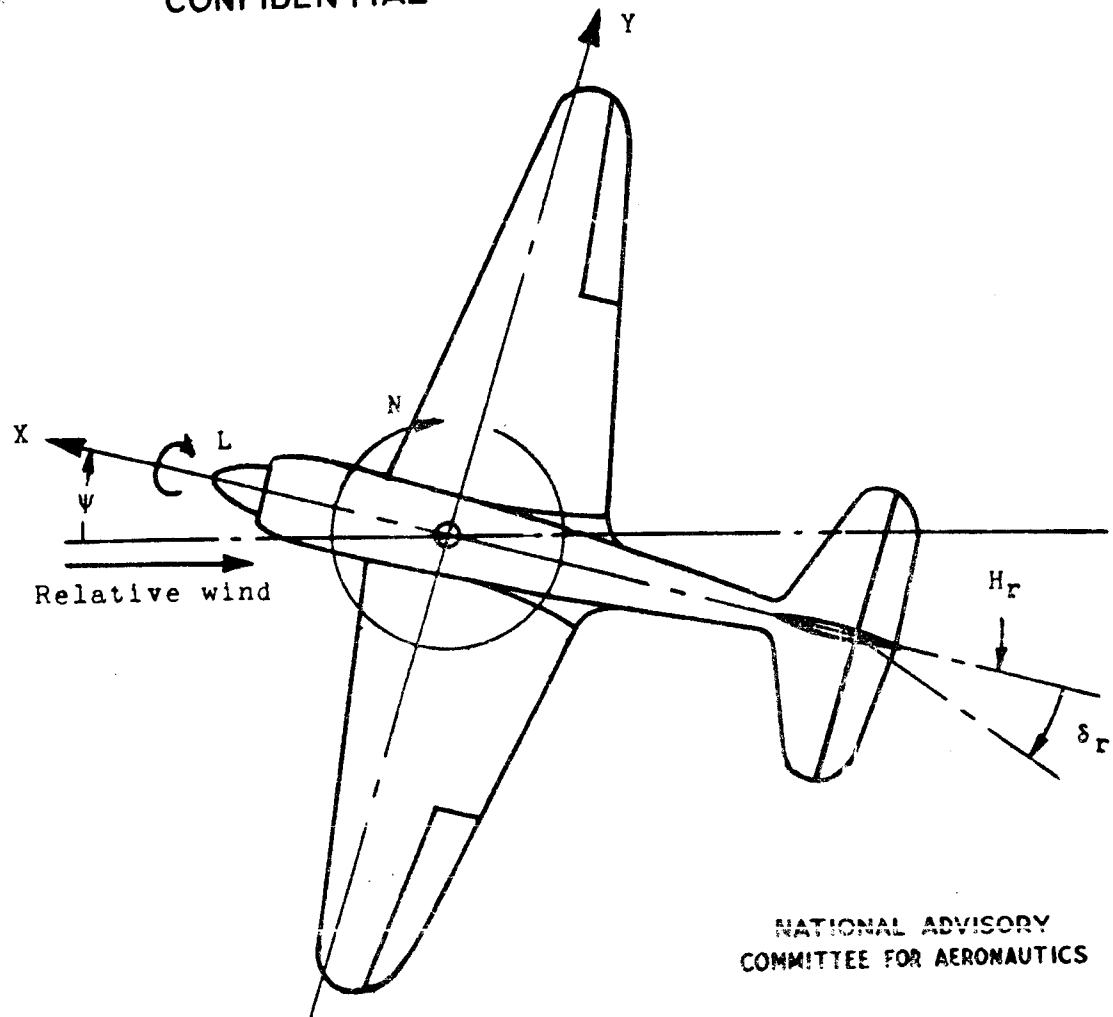


Figure 1.- System of axes and control-surface hinge moments and deflections. Positive values of forces, moments, and angles are indicated by arrows. Positive values of tab hinge moments and deflections are in the same directions as the positive values for the control surfaces to which the tabs are attached.

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Figure 2.-  $\frac{1}{4}$ -scale model of the Bell XS-1 airplane mounted in the 300 mph 7- by 10-foot wind tunnel, high speed configuration.

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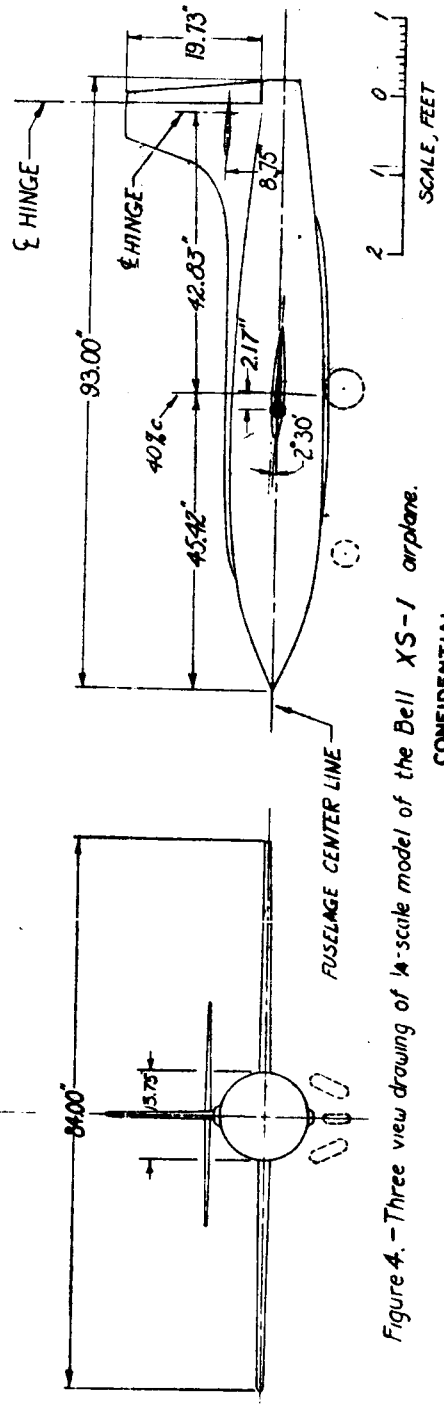
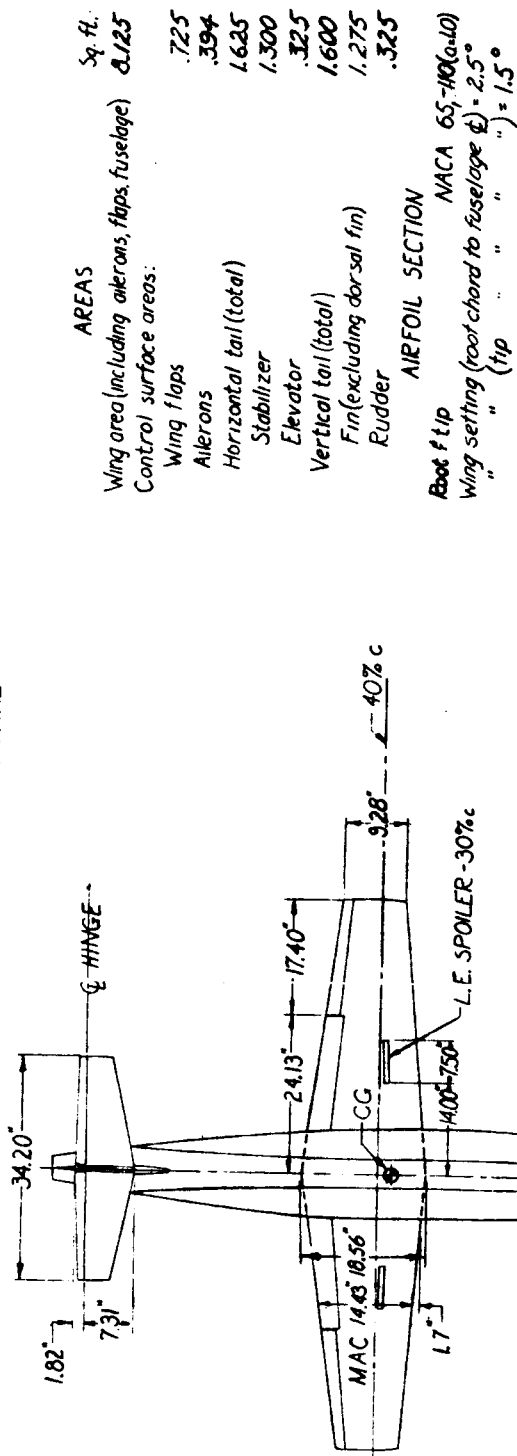
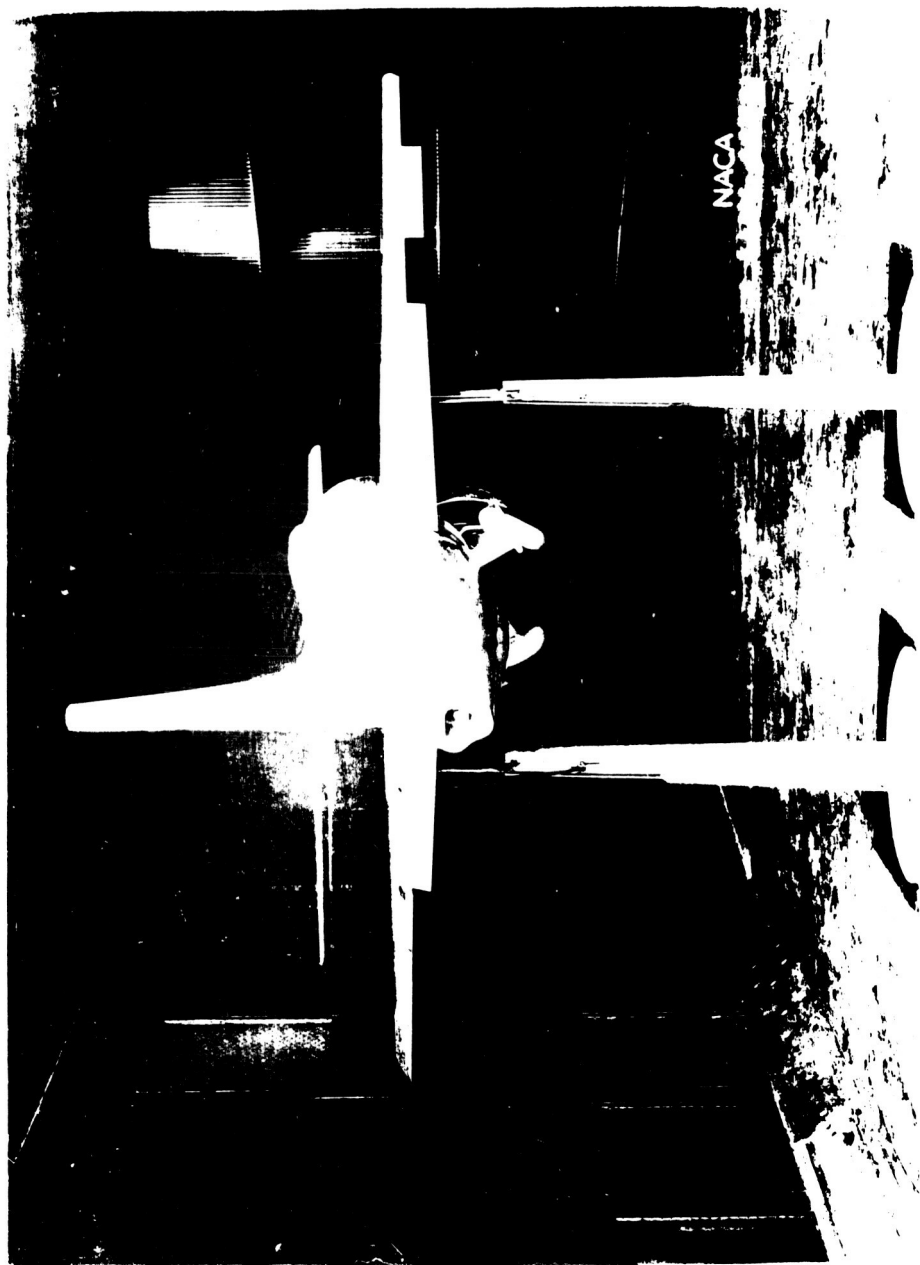


Figure 4. - Three view drawing of 1/4-scale model of the Bell XS-1 airplane.

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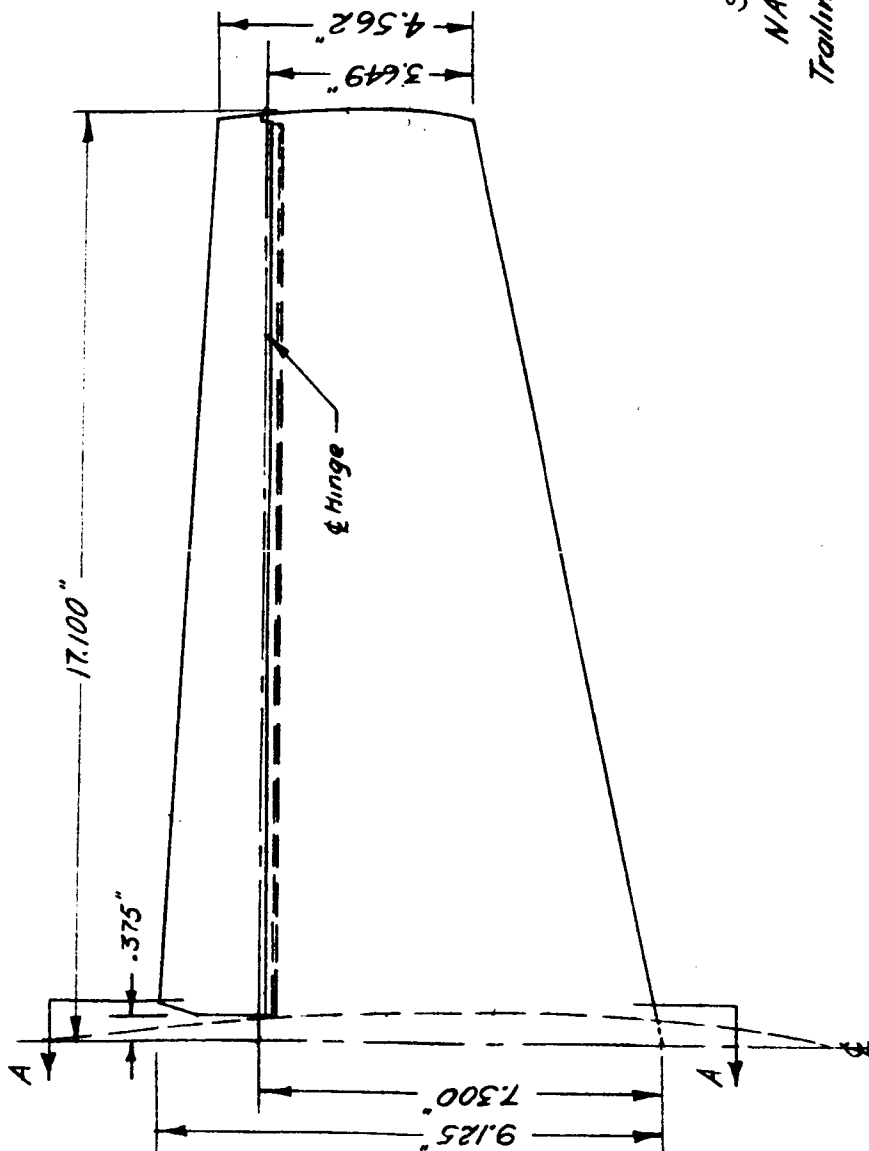
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Figure 3.-  $\frac{1}{4}$ -scale model of the Bell XS-1 airplane mounted in the 300 mph  
7- by 10-foot wind tunnel, landing configuration.

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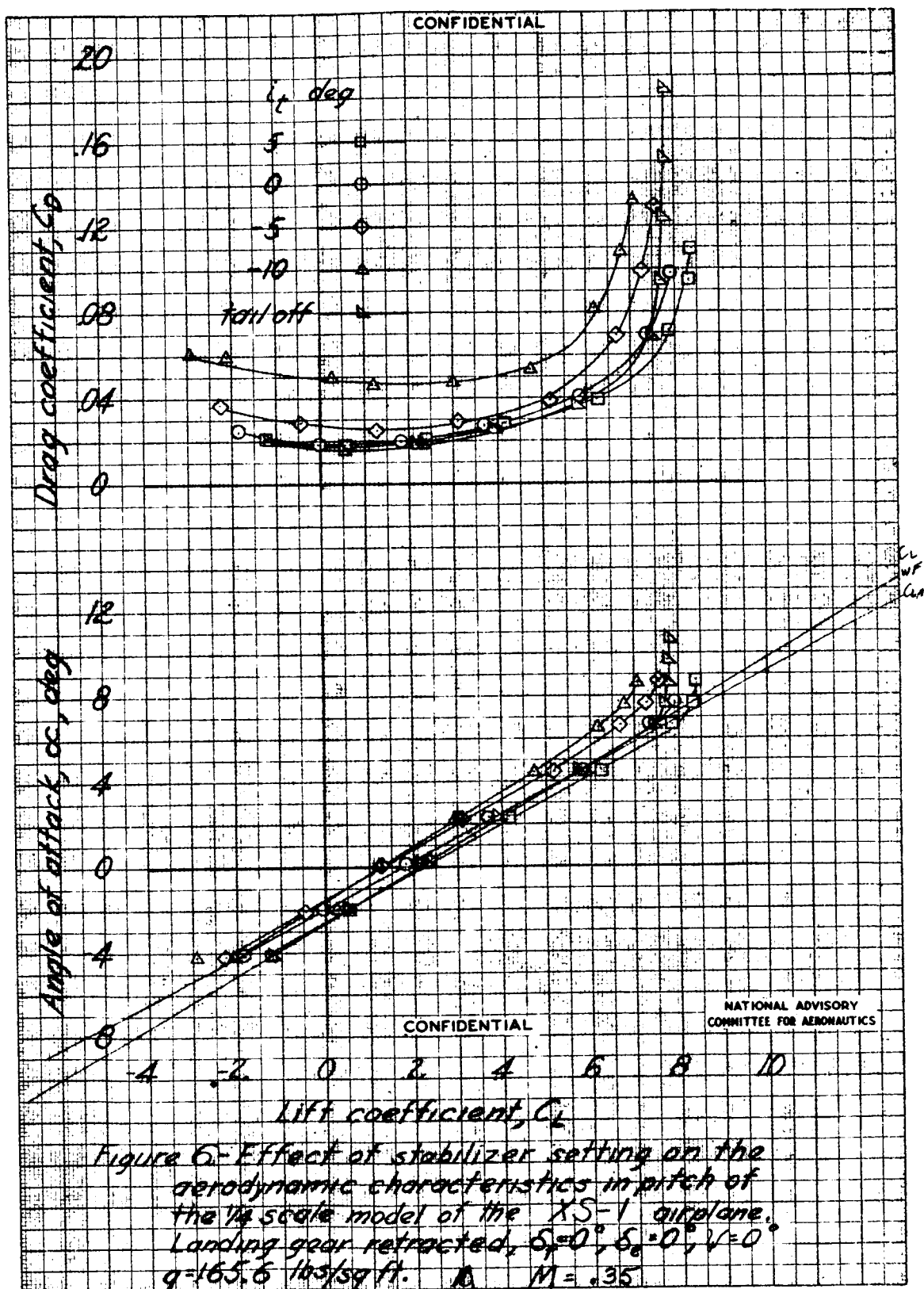
Section A-A  
NACA 65,-008  
Trailing edge angle =  $10^{\circ}$

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Figure 5.- Drawing of the horizontal tail of the 1/4 scale model



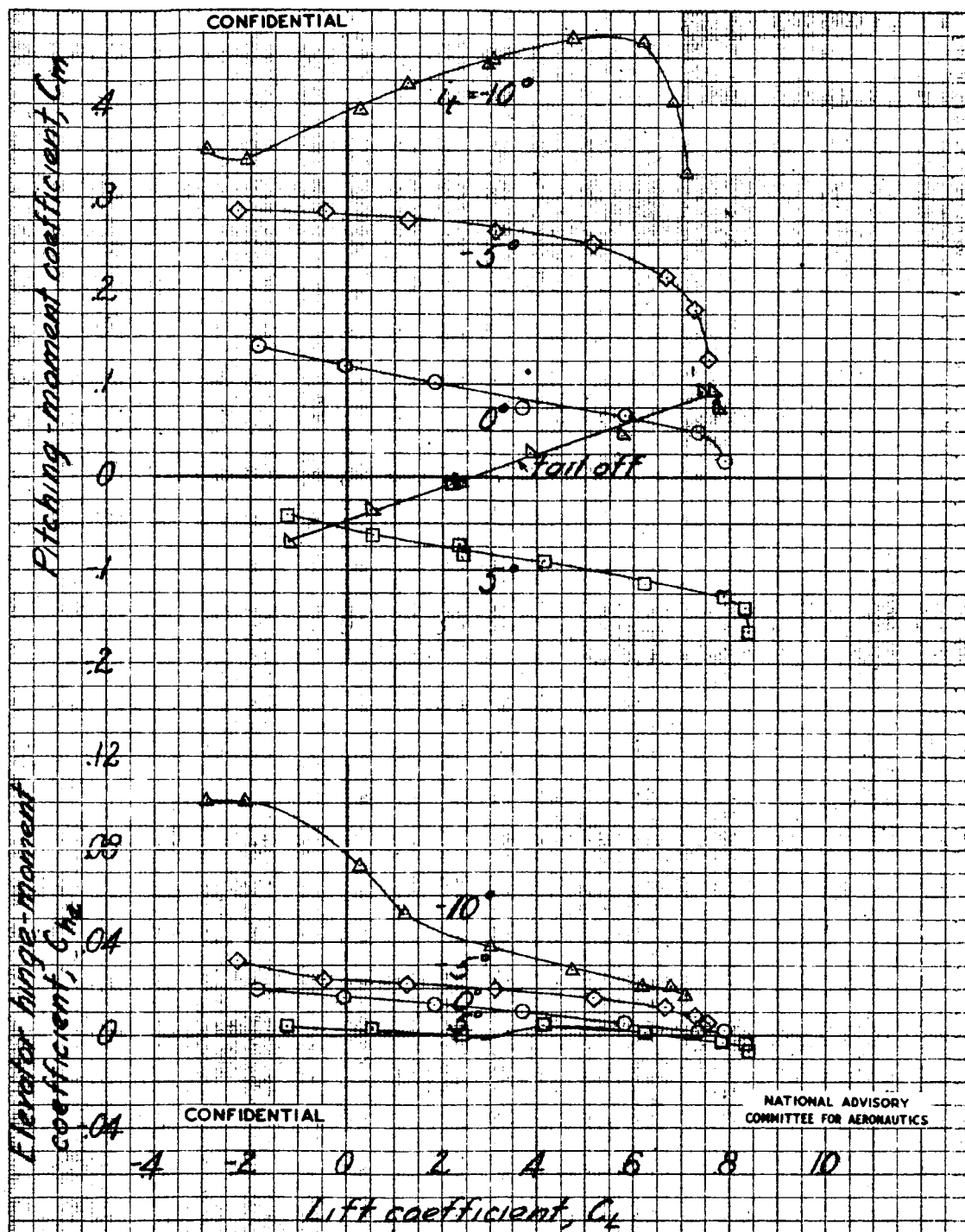


Figure 6.- Concluded.



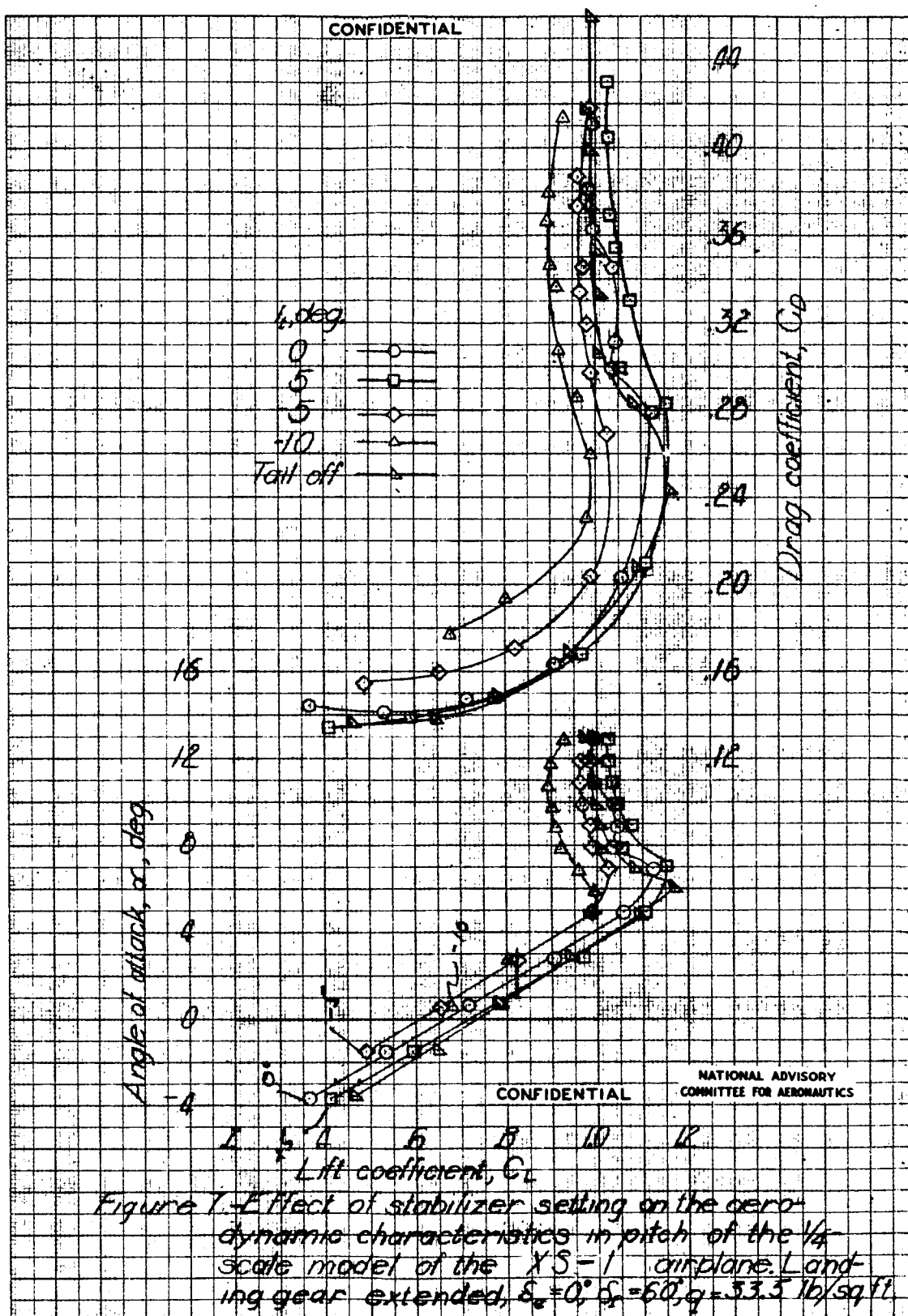


Figure 1. Effect of stabilizer setting on the aerodynamic characteristics in pitch of the  $1/4$ -scale model of the X-5-1 airplane. Landing gear extended,  $\delta_e = 0^\circ$ ,  $\delta_r = 60^\circ$ ,  $q = 33.3 \text{ lb/sq ft}$

$M = .15$

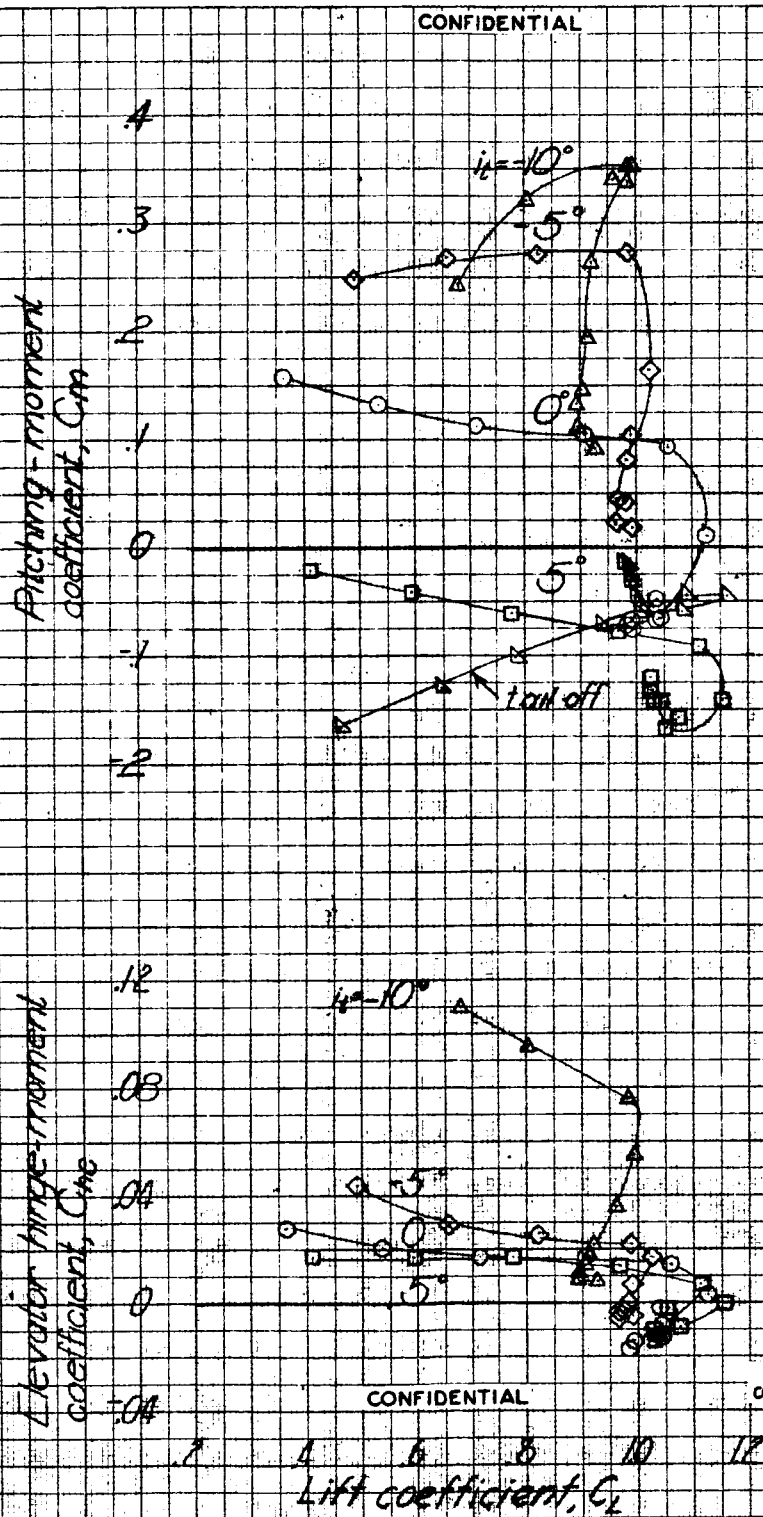
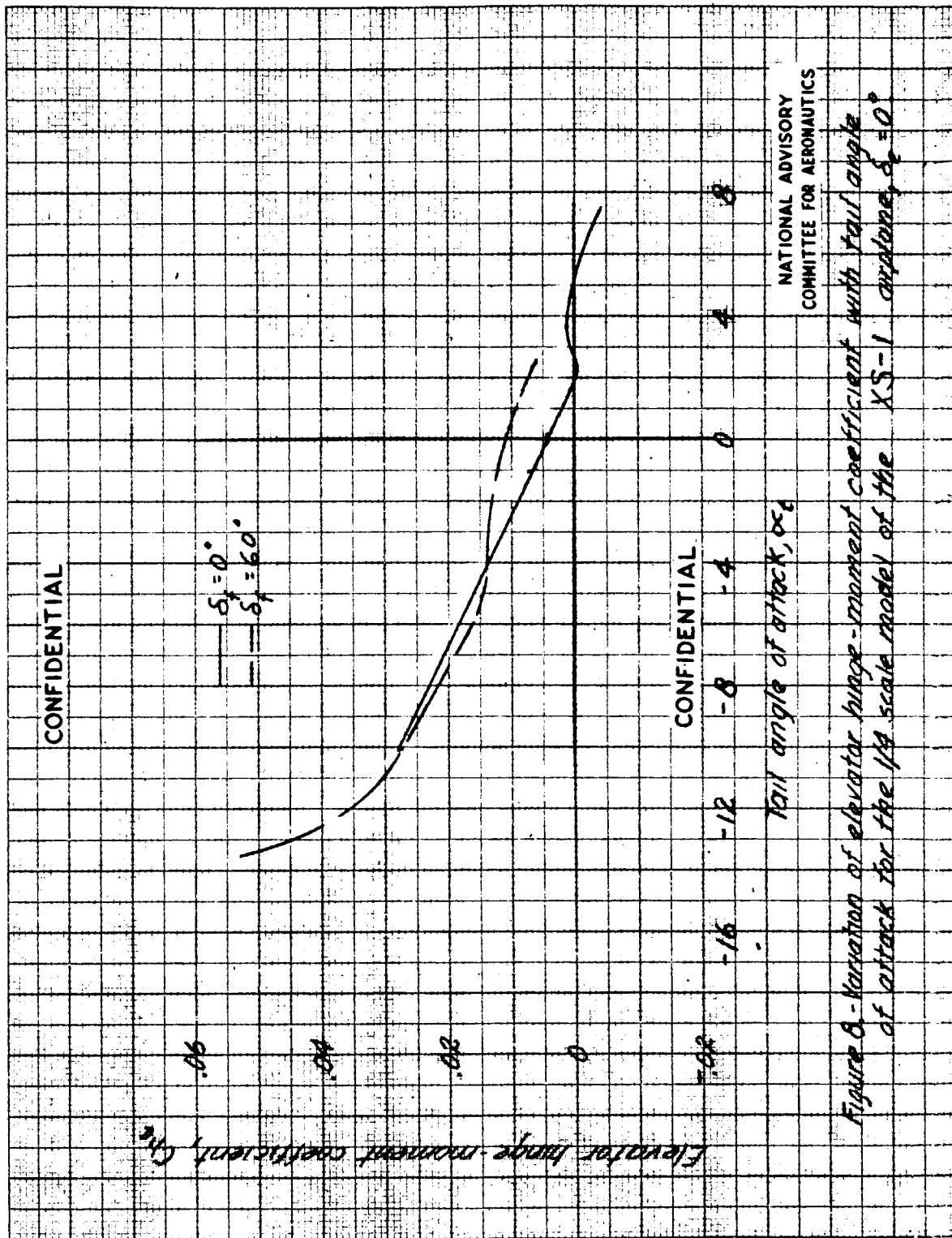
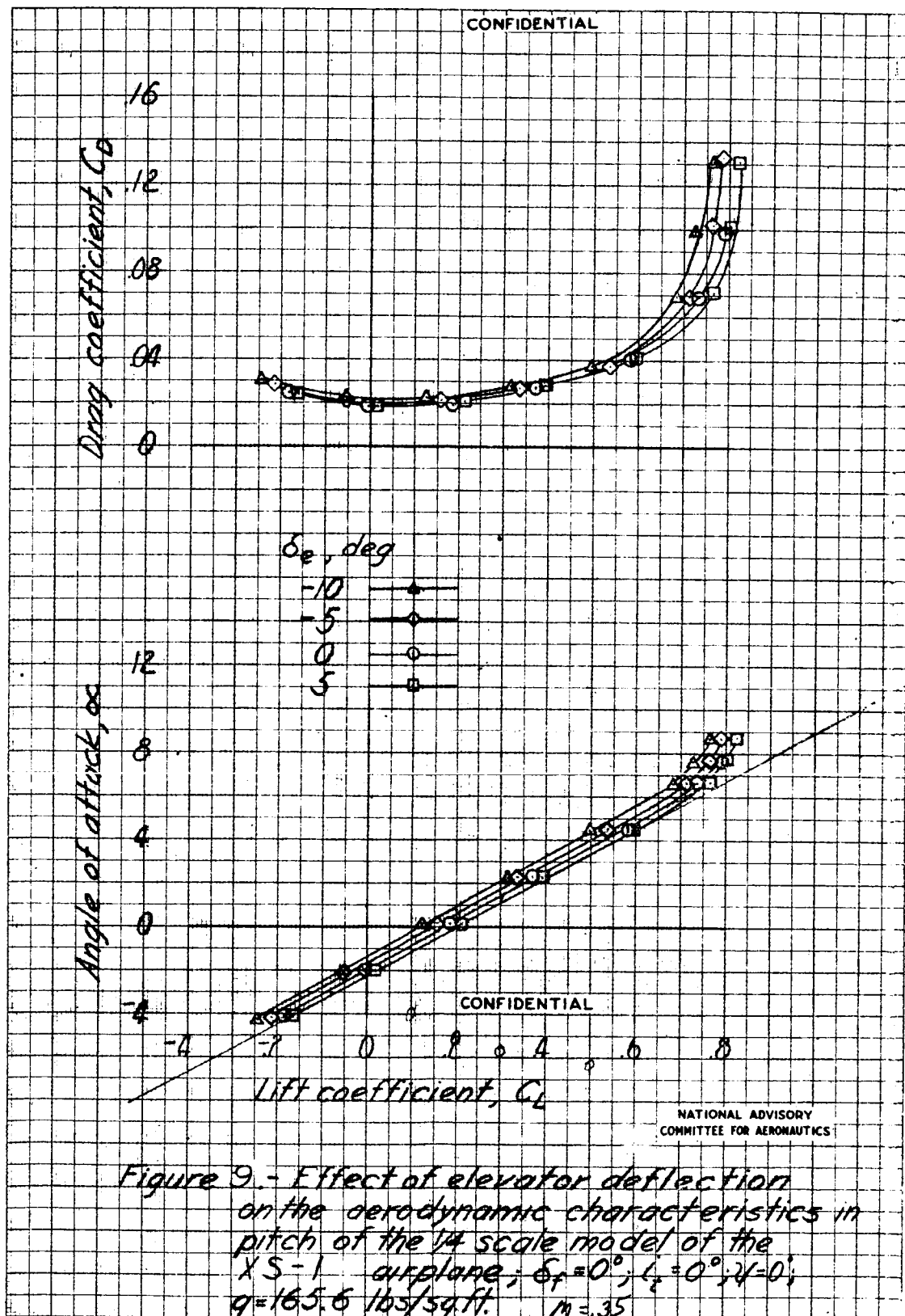


Figure 7- Concluded

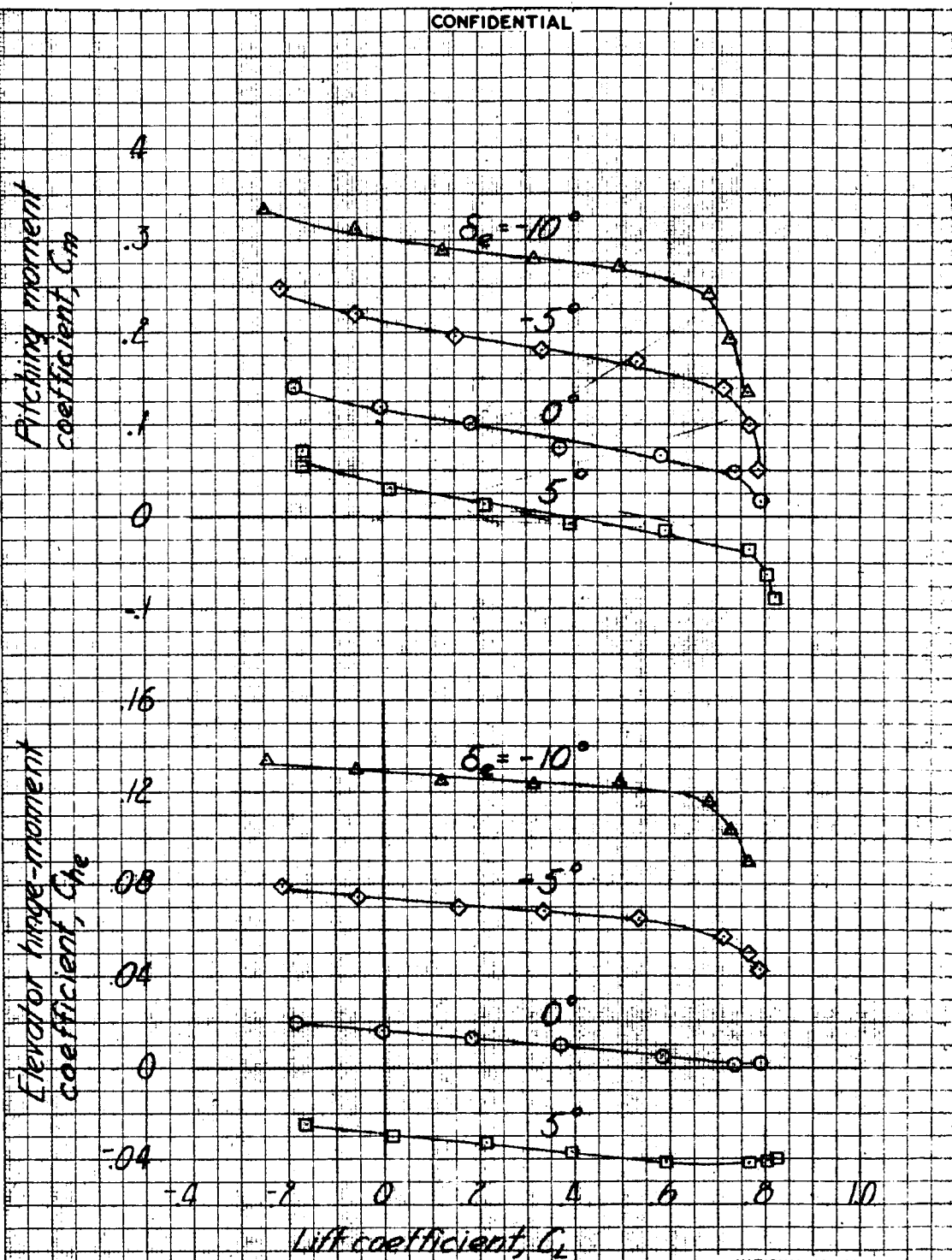
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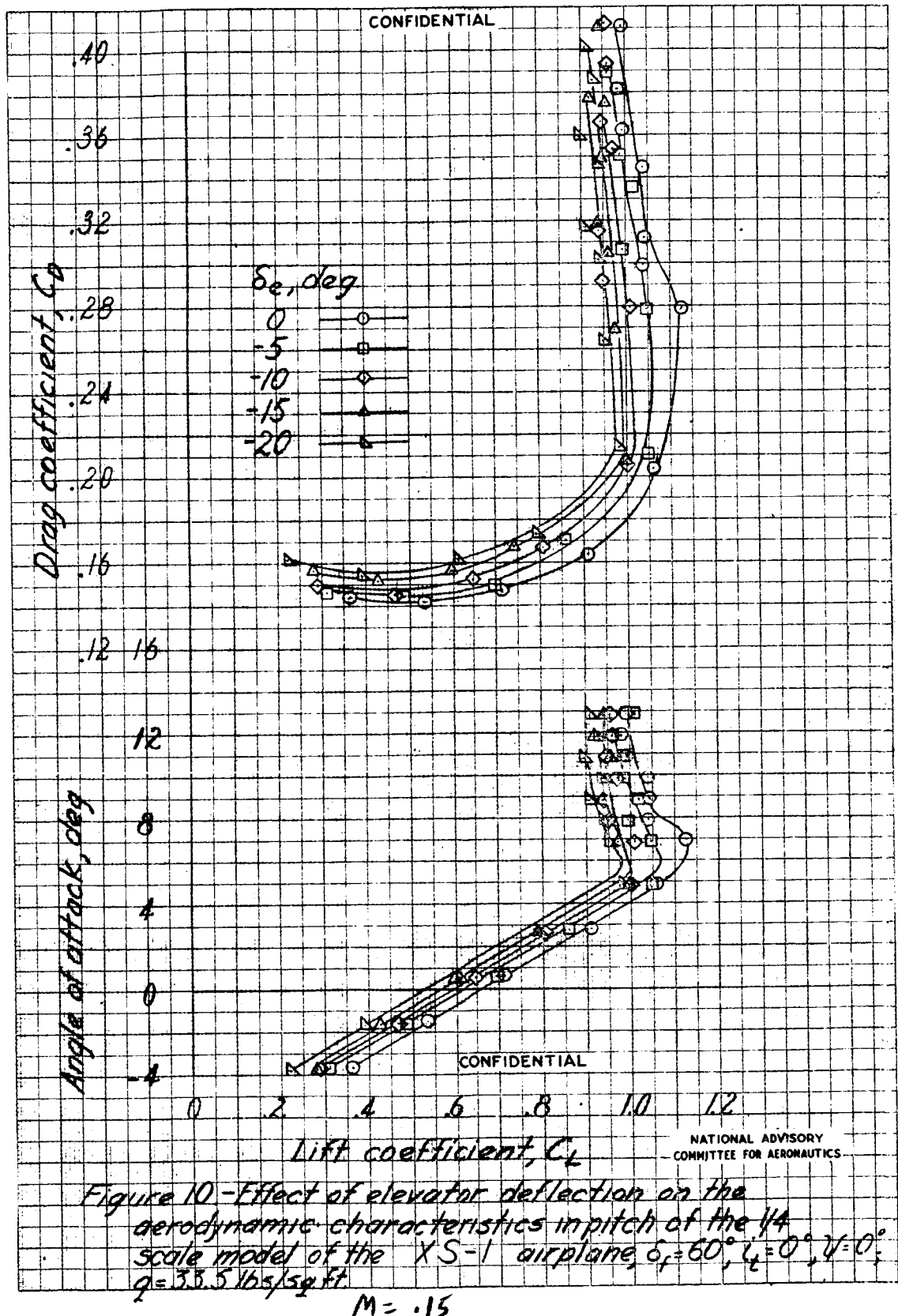
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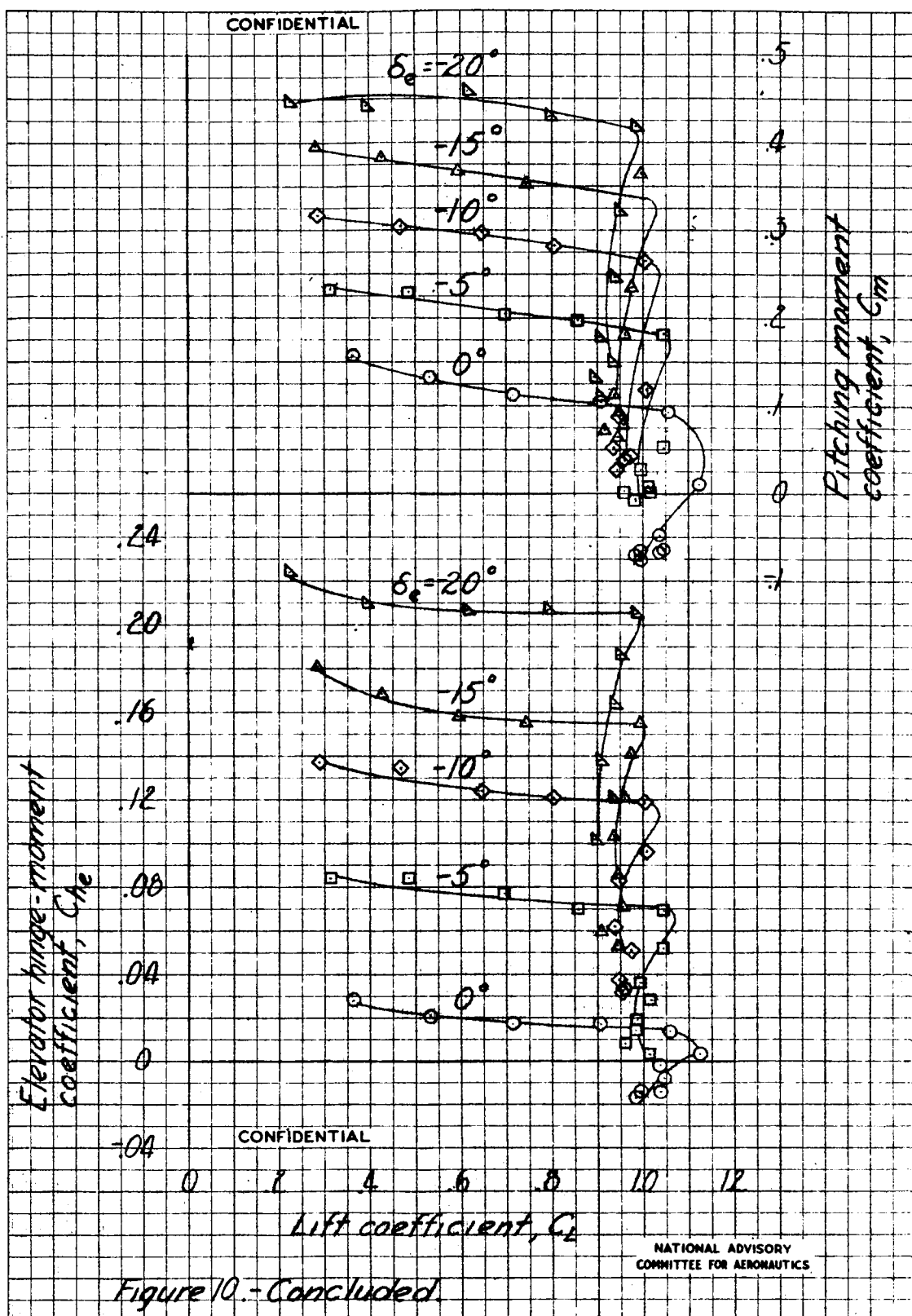
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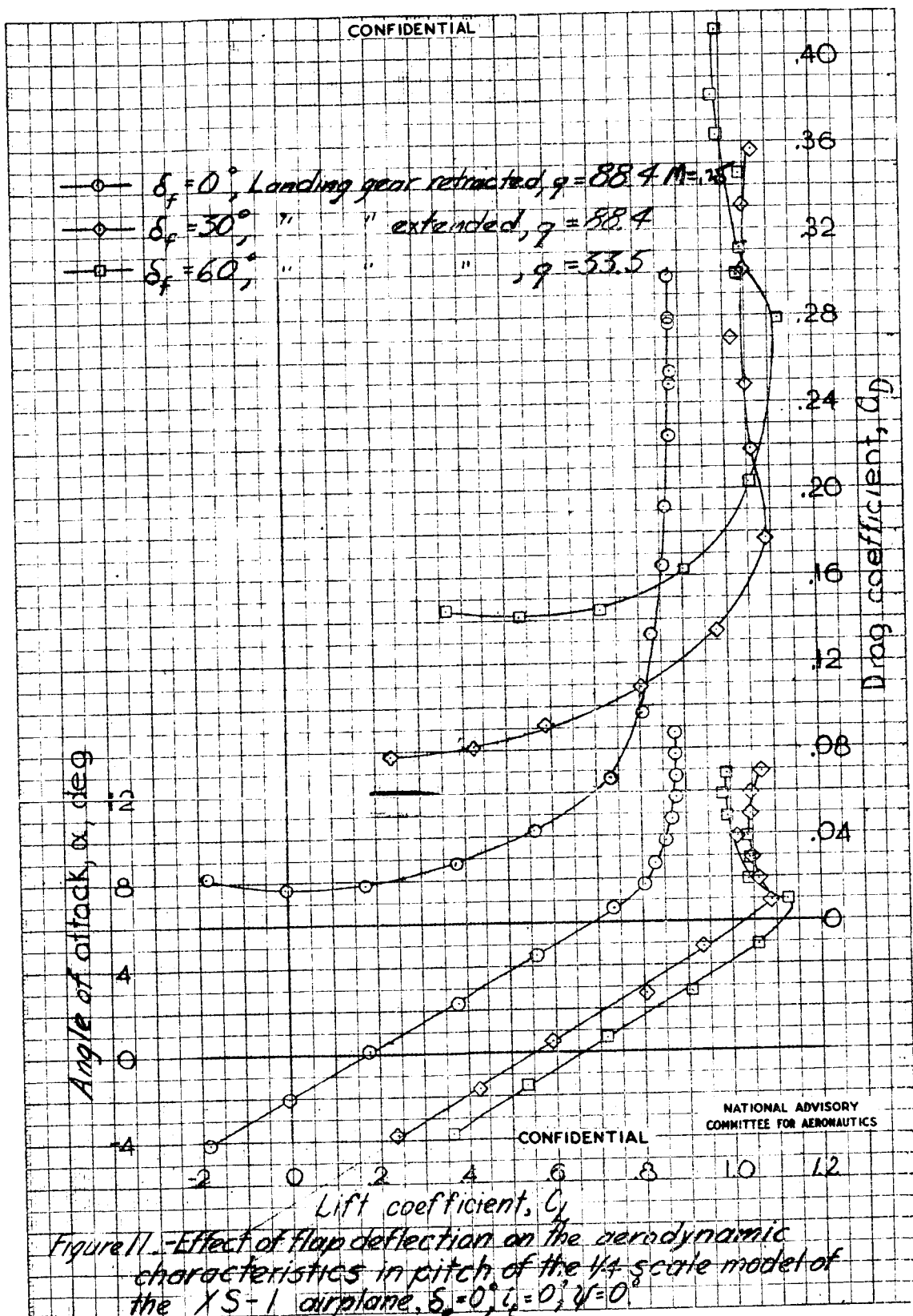
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Figure 9.- Concluded



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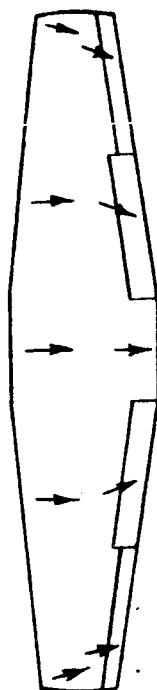
- $\circ$   $\delta_f = 0^\circ$ , Landing gear retracted,  $q = 88.4$  lbs/sq ft  
 $\diamond$   $\delta_f = 30^\circ$ , " " extended,  $q = 88.4$  lbs/sq ft  
 $\square$   $\delta_f = 60^\circ$ , " " " " ,  $q = 33.5$  lbs/sq ft

Pitching-moment coefficient,  $C_m$ 3  
2  
1  
0  
-1-2 0 2 4 6 8 10  
Lift coefficient,  $C_L$ 

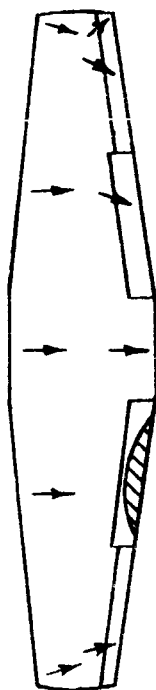
Figure 11.-Concluded

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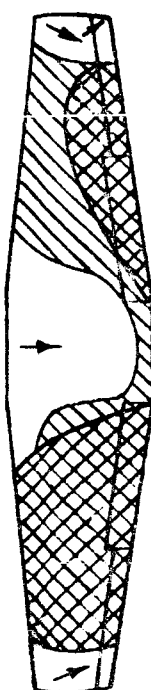
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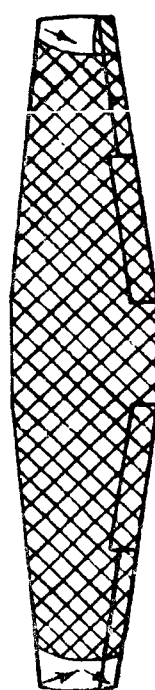
$\alpha = 2.4^\circ$



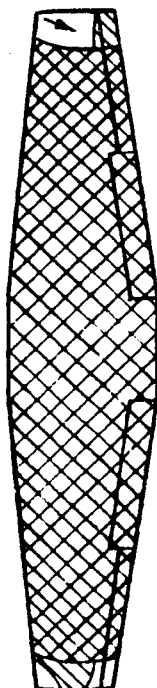
$\alpha = 6.7^\circ$



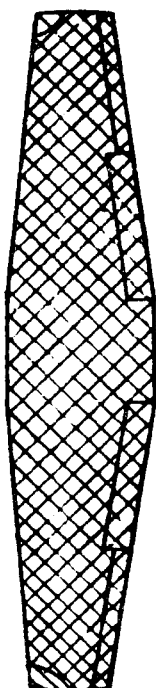
$\alpha = 8.8^\circ$



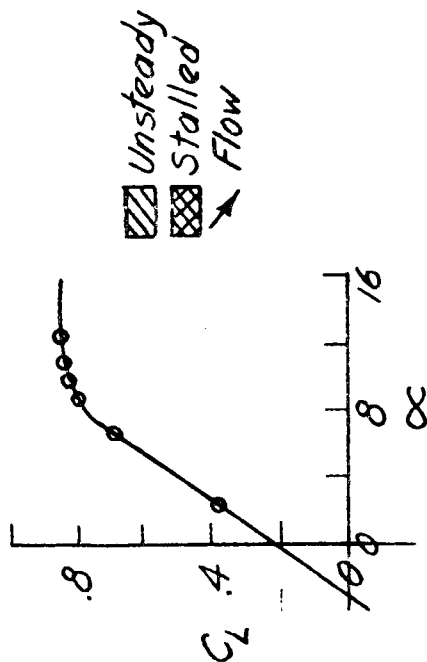
$\alpha = 9.8^\circ$



$\alpha = 10.8^\circ$



$\alpha = 12.8^\circ$



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Figure 12.-Tuft studies of the  $1/4$  scale model of the X5-1 isolated wing;  $\delta_f = 0^\circ$ ;  $\psi = 0^\circ$ ;  $q = 88.4 \text{ lbs/sq ft}$ .

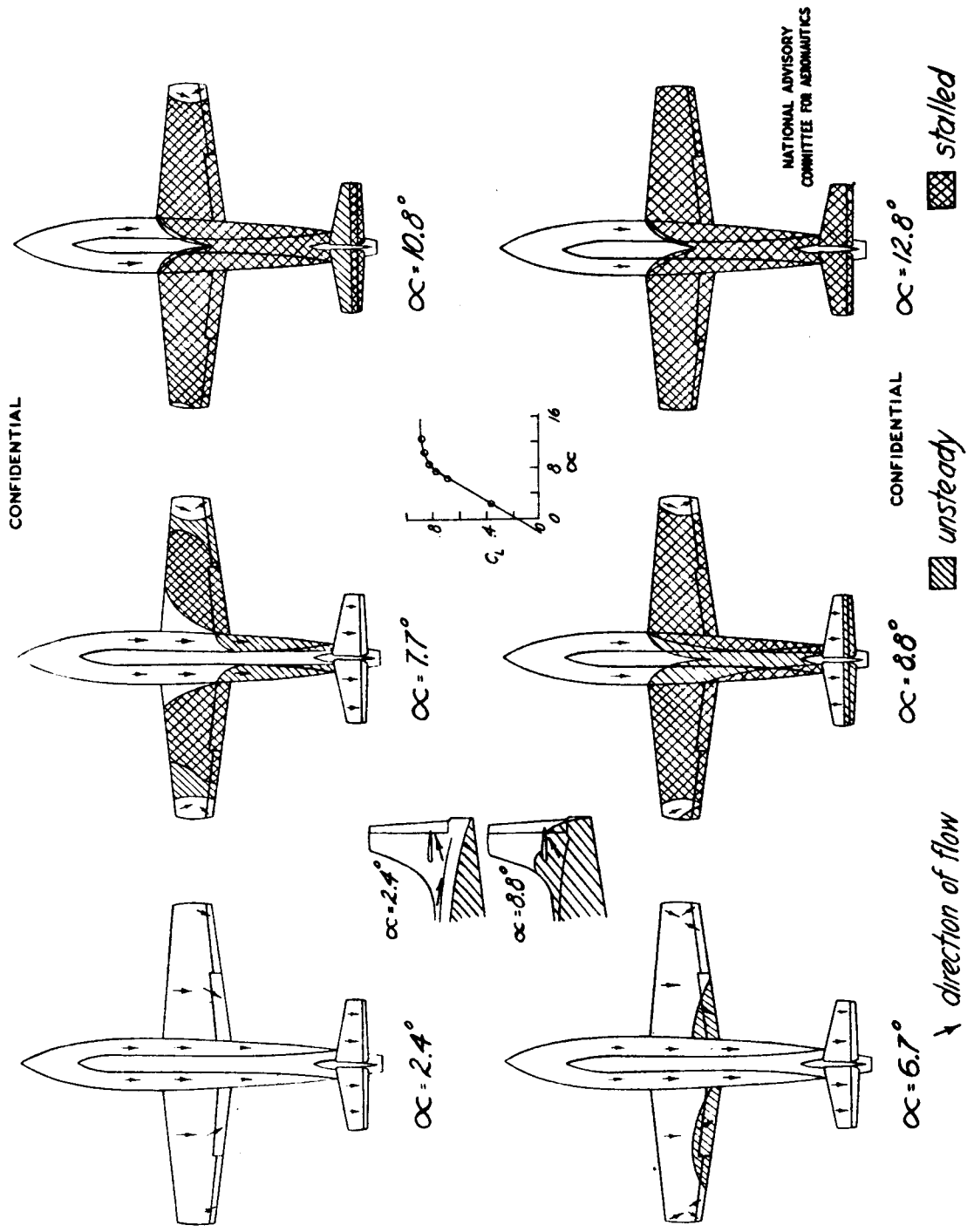


Figure 13-Tiff studies of the 1/4 scale model of the X-5-1 airplane  $\delta = 0^\circ$   $\alpha = 28.4$  kts/150 ft

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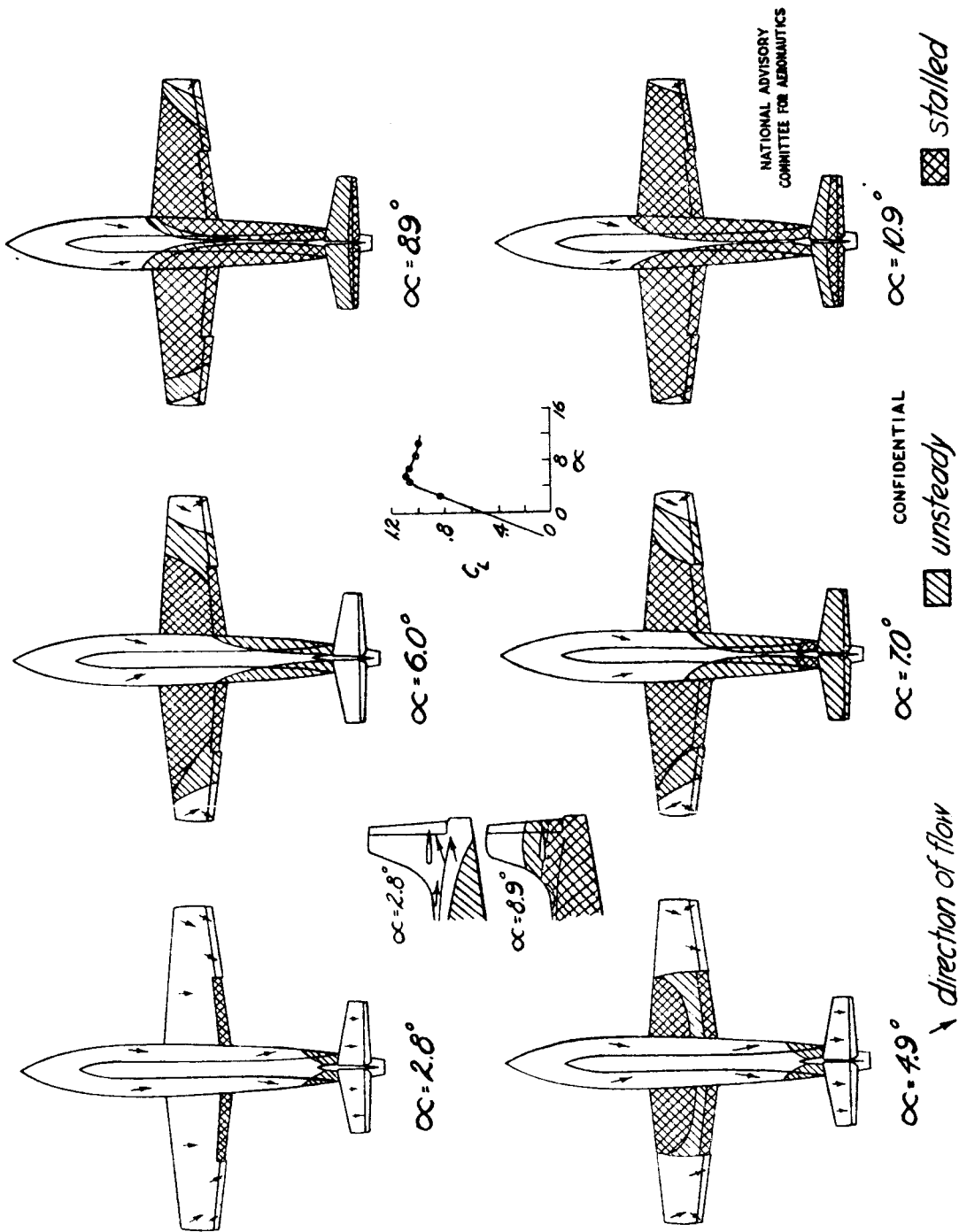
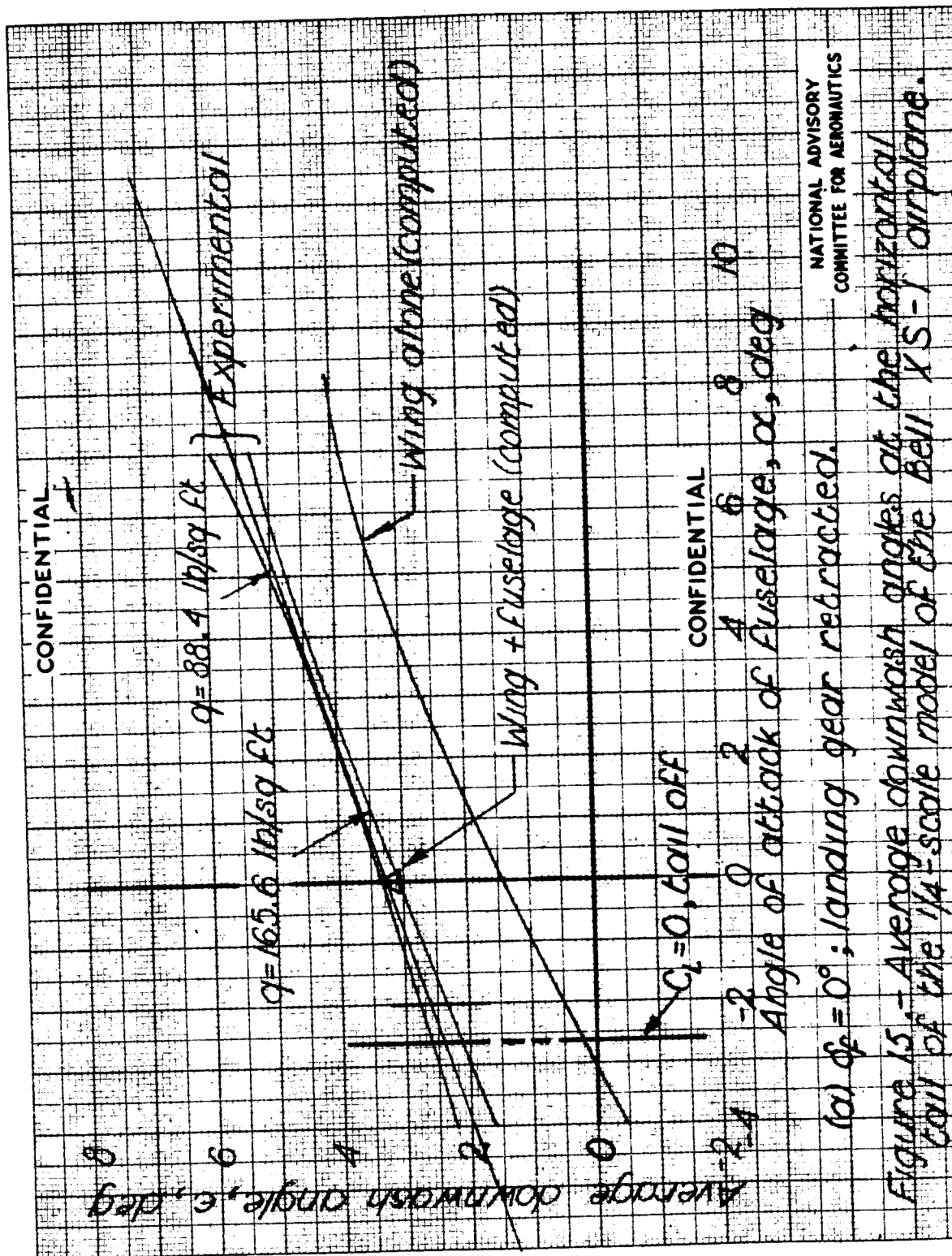
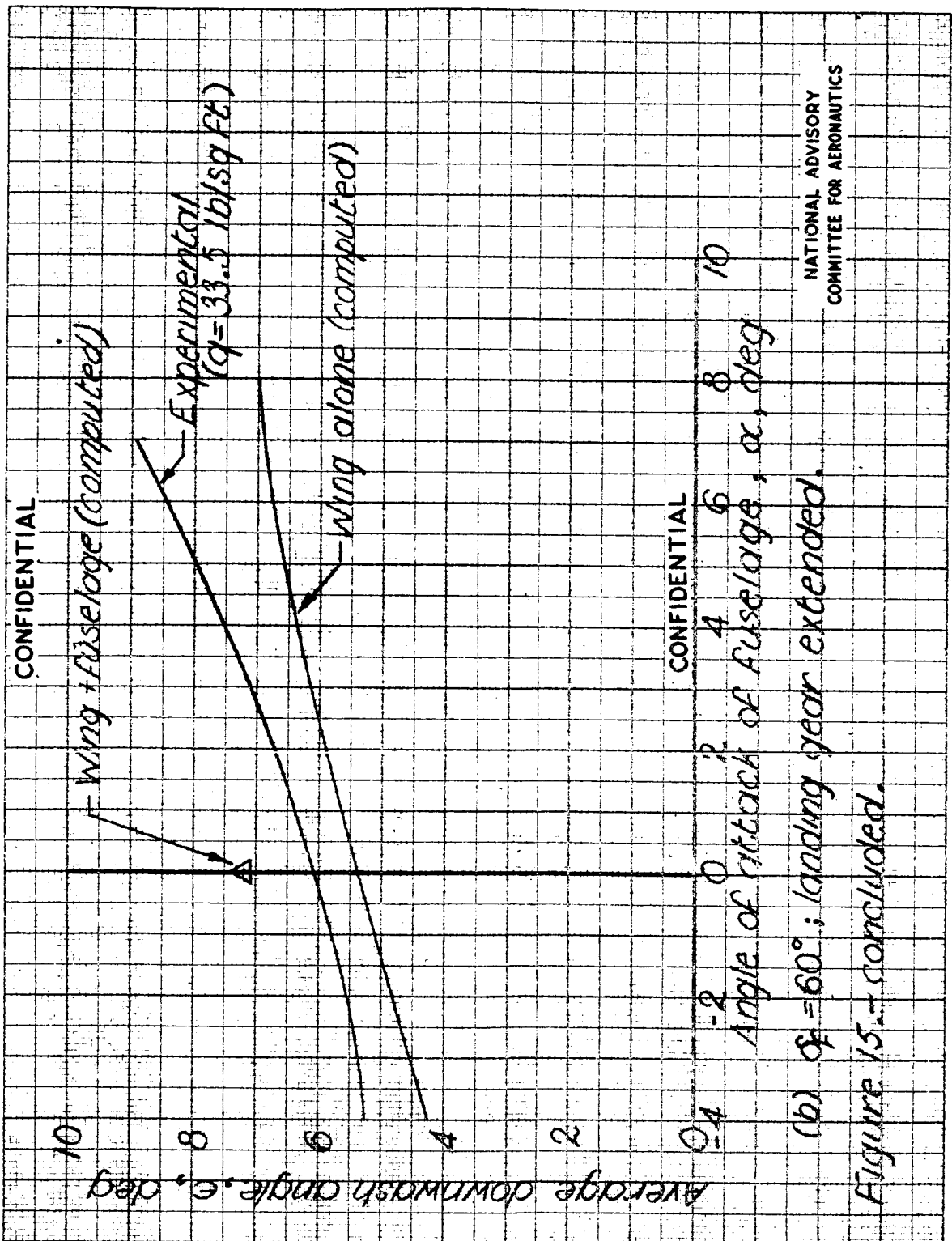


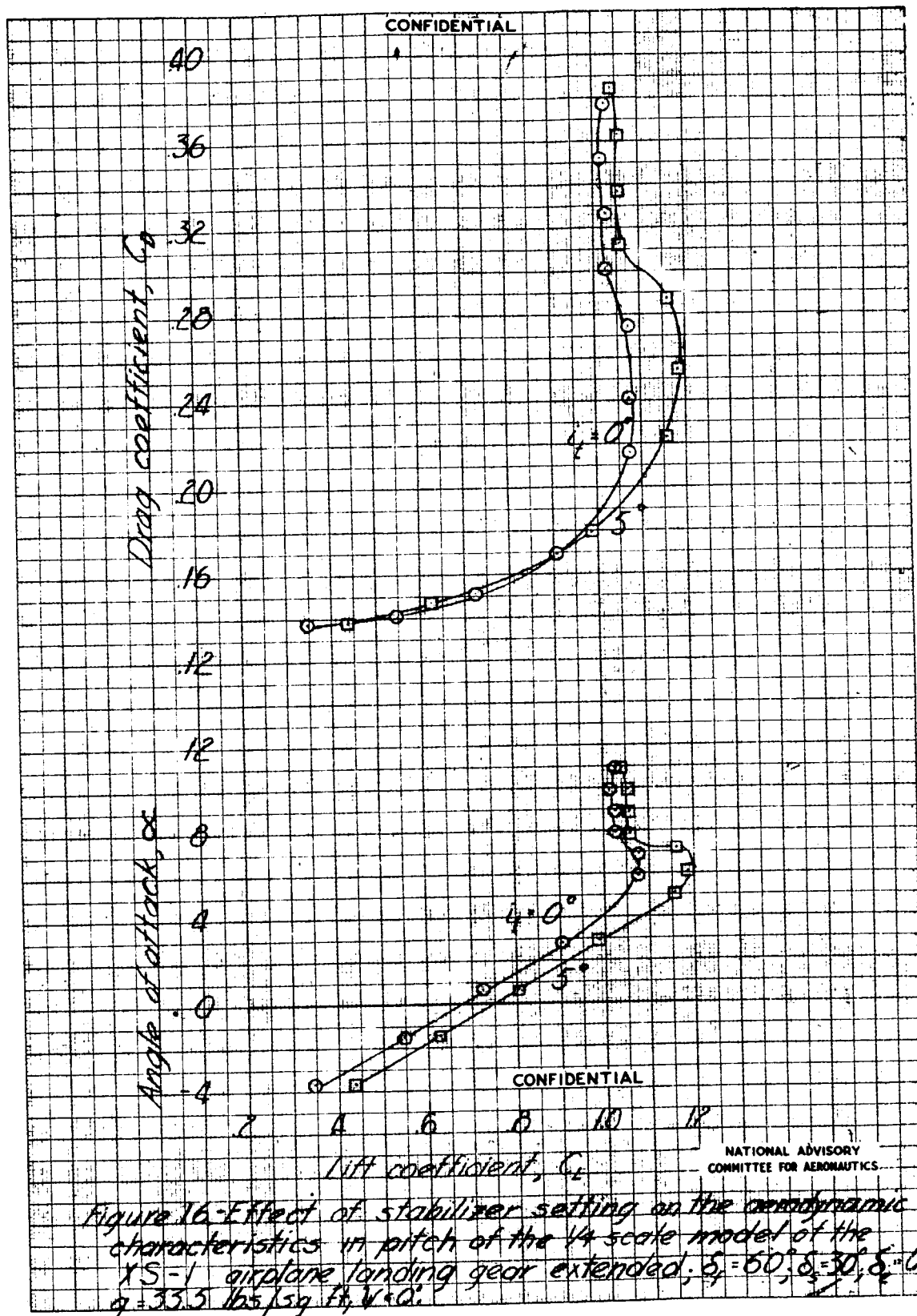
Figure 14: Tuft studies of the 1/4 scale model of the X-5-1 airplane,  $\delta_f = 60^\circ$ ,  $\psi = 0^\circ$ ,  $q = 33.5 \text{ lbs/sq ft}$ .





(b)  $\phi_f = 60^\circ$ ; landing gear extended.

Figure 15.-concluded.



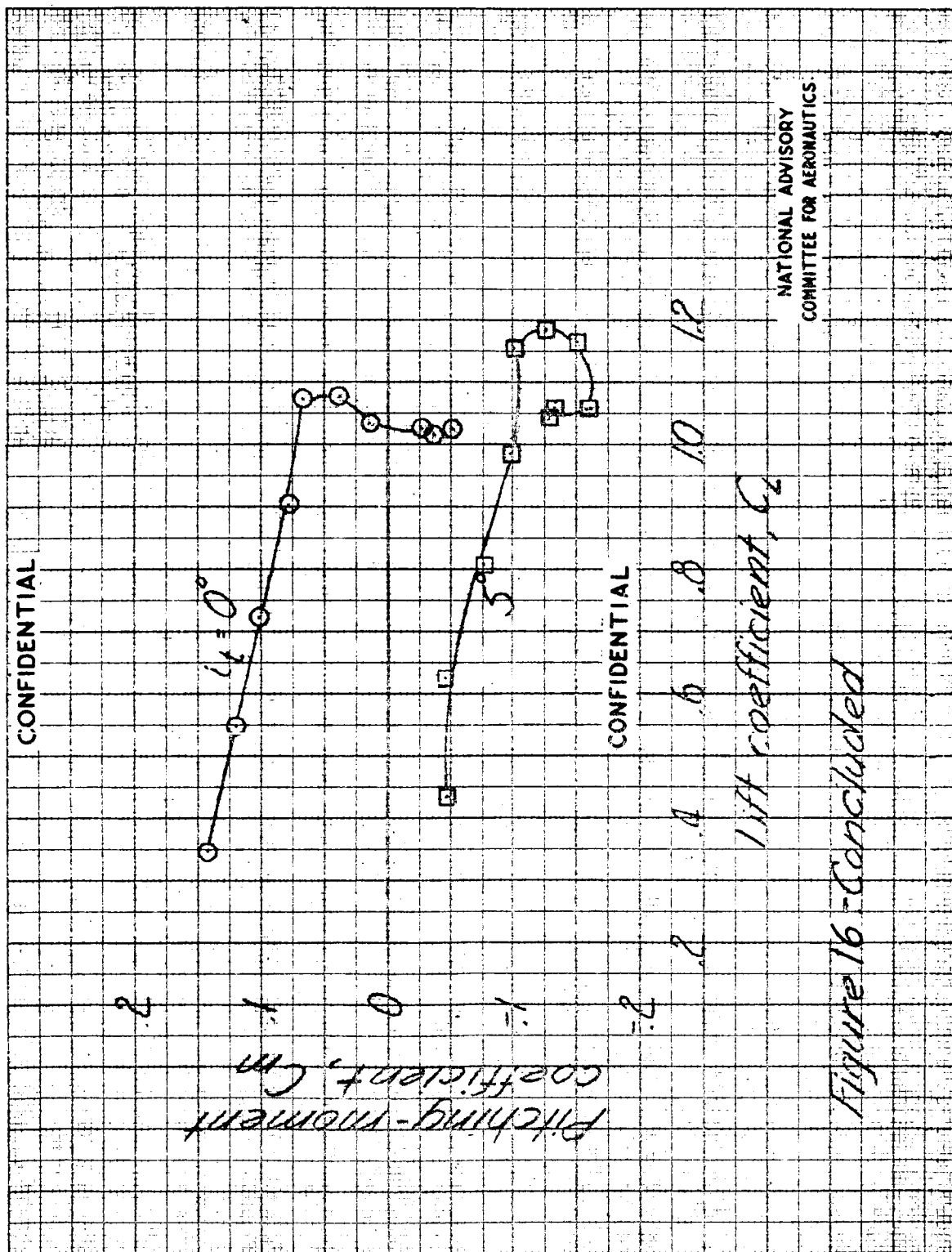
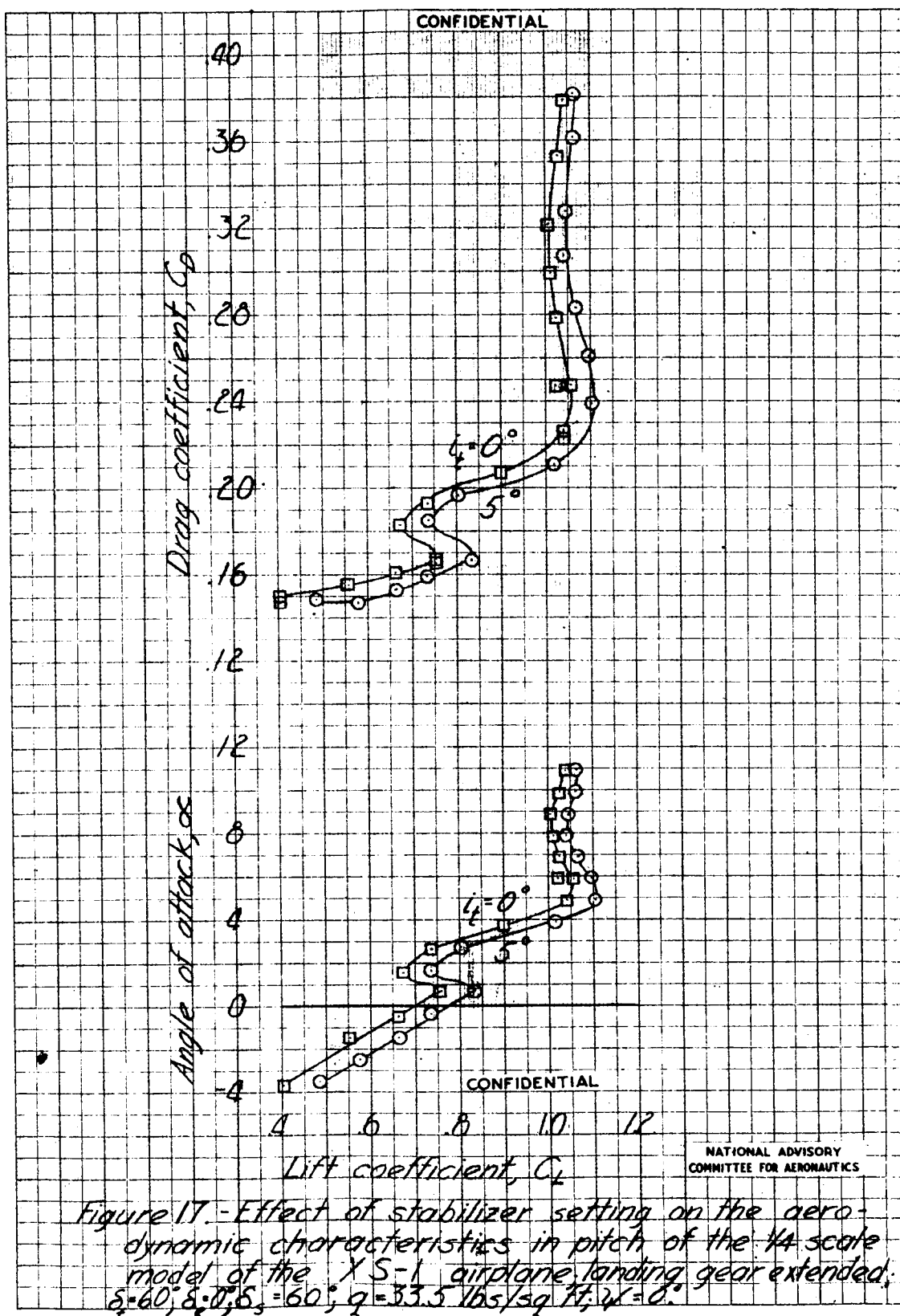
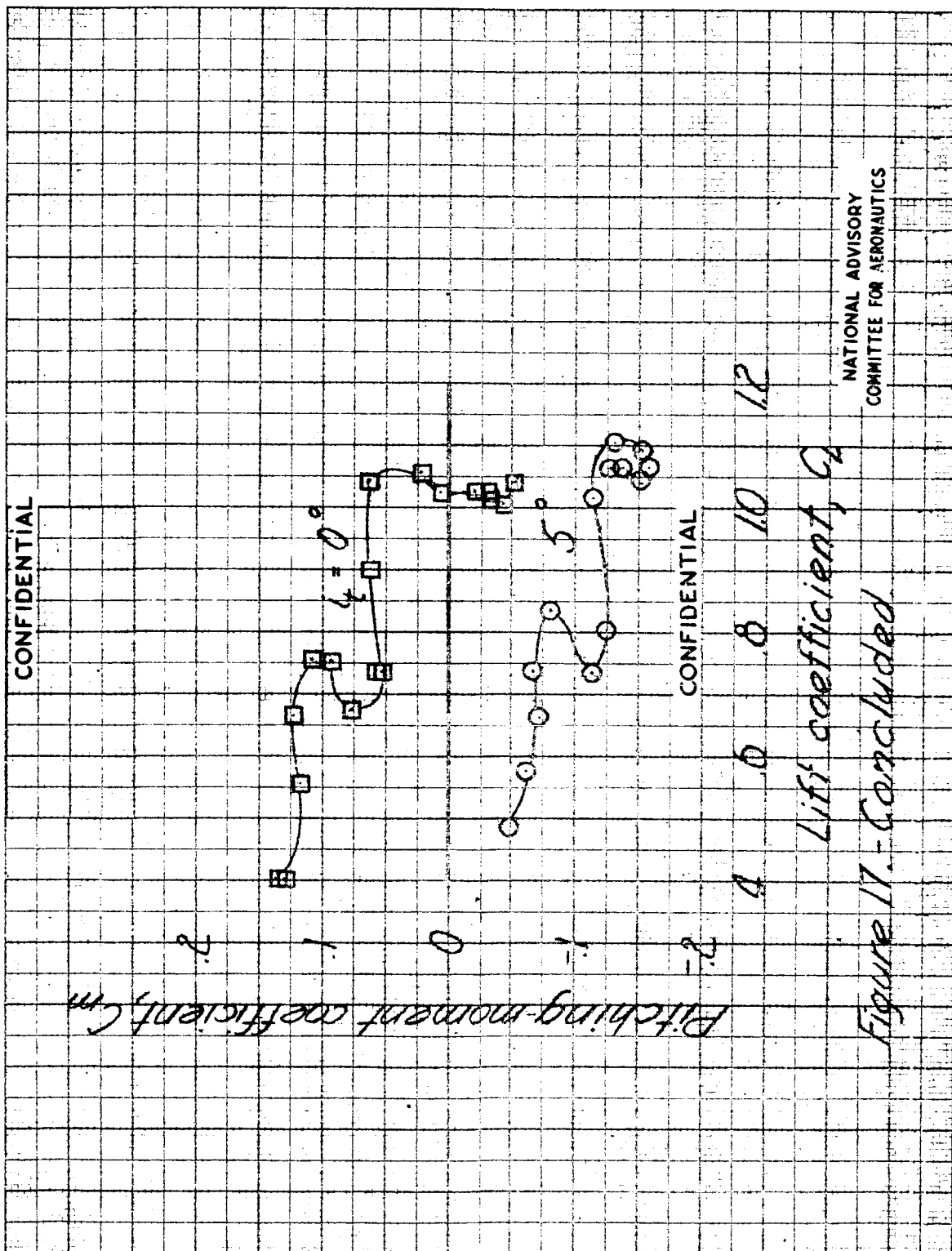


Figure 16 - Concluded

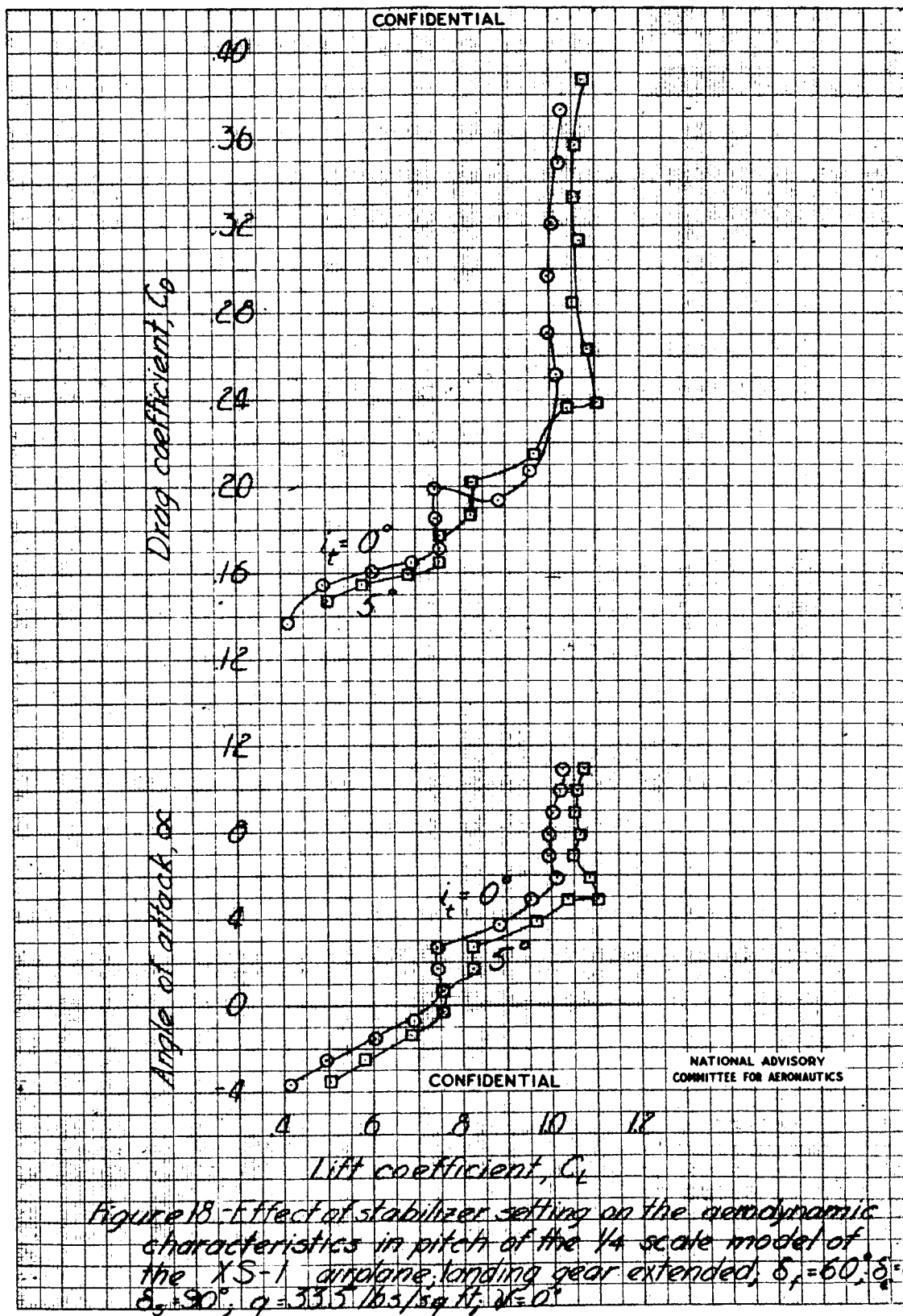


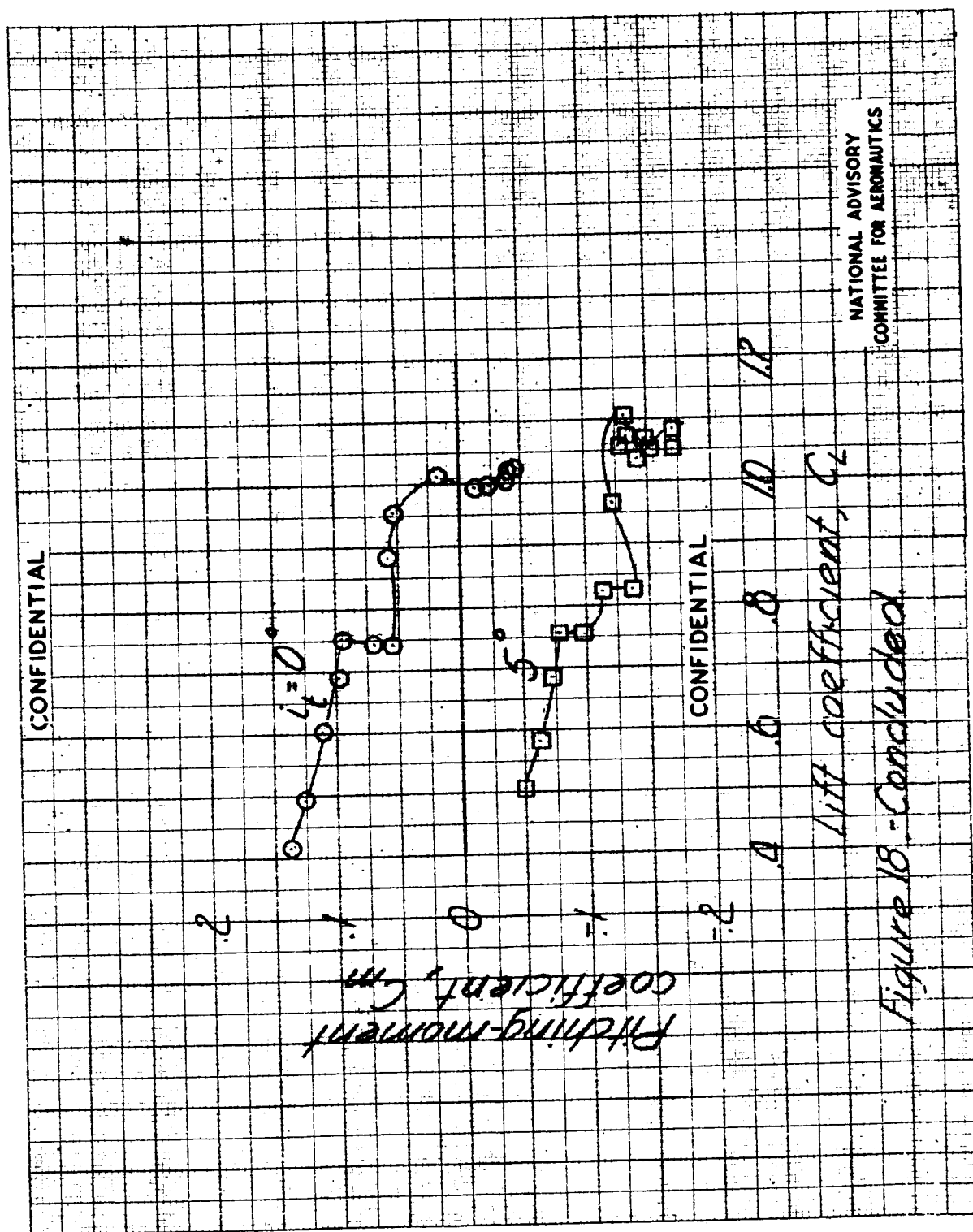


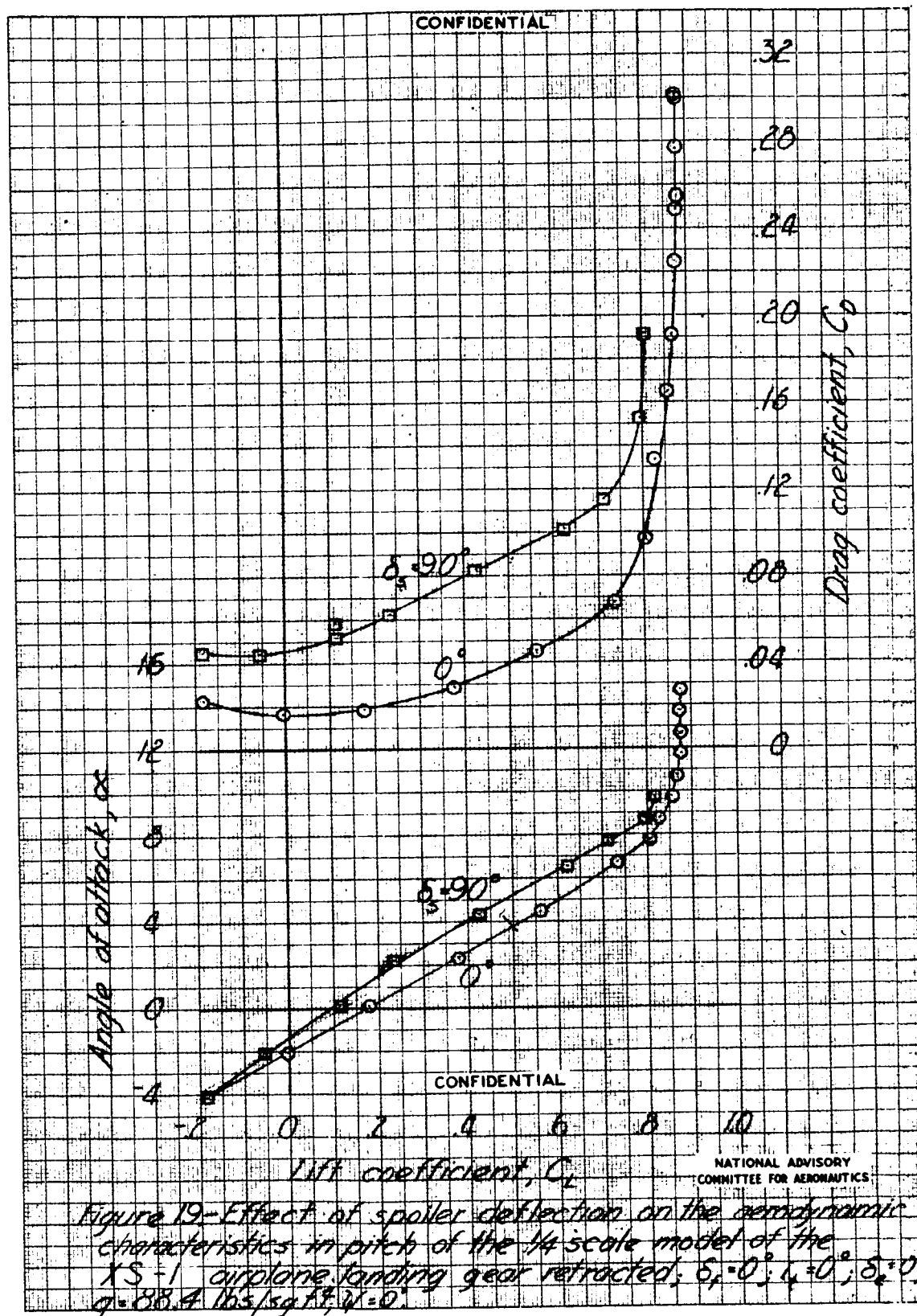


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Figure 17.-Concluded







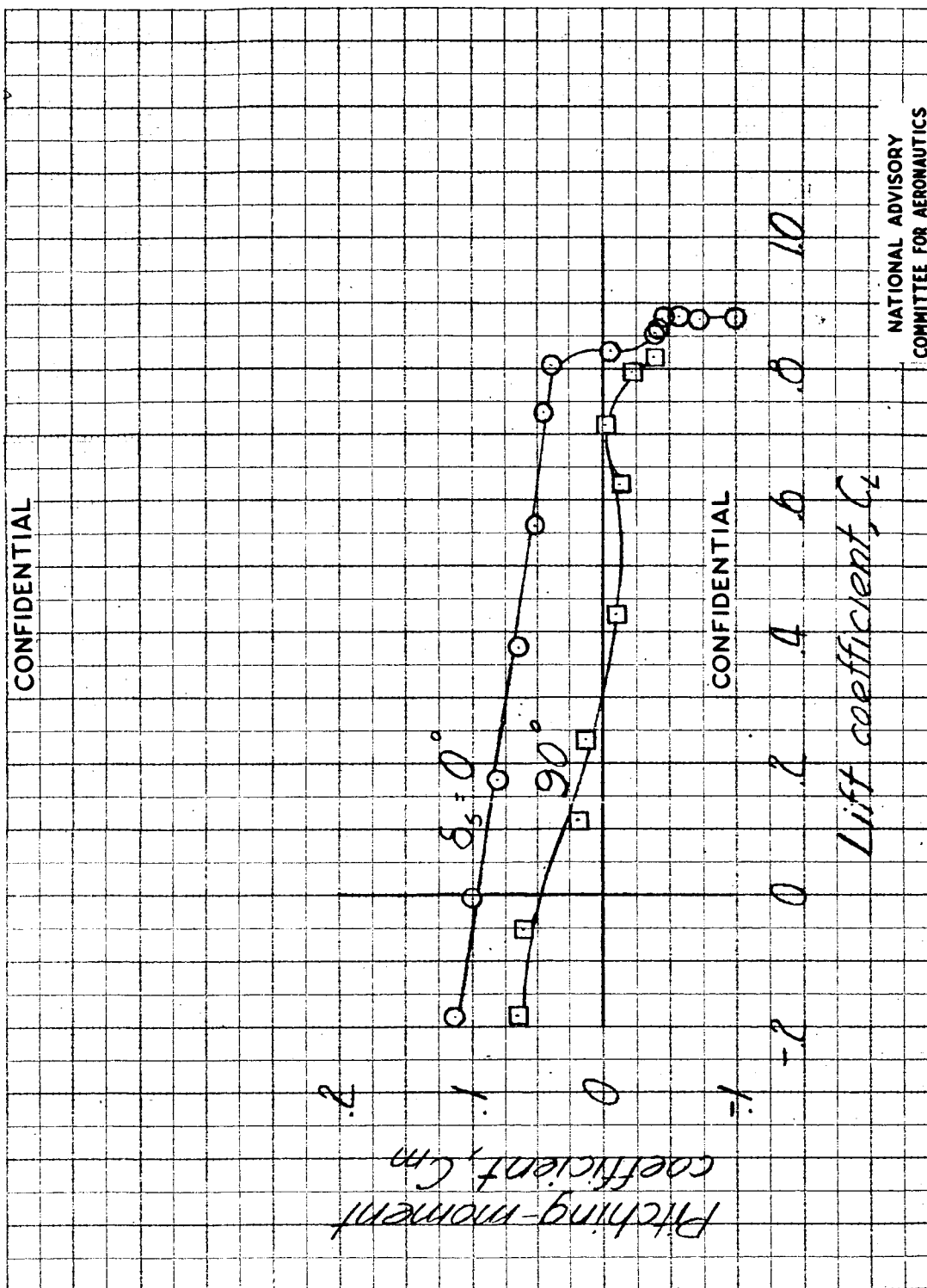


Figure 19- Concluded

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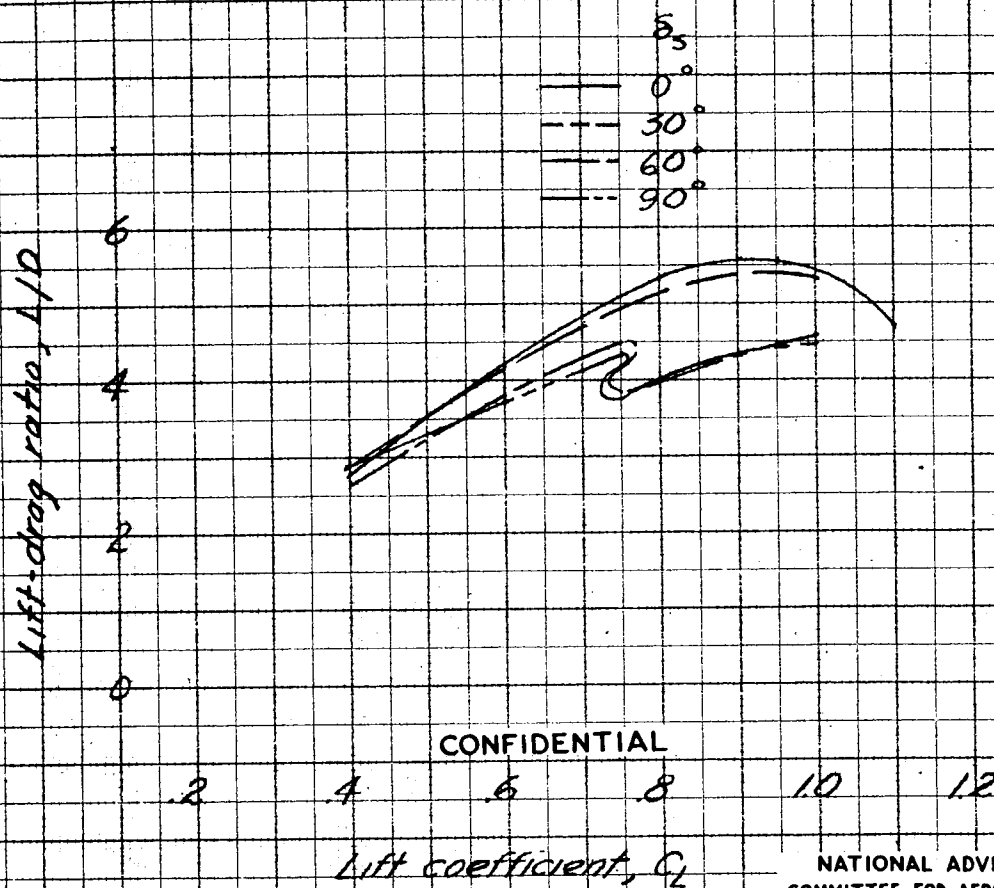
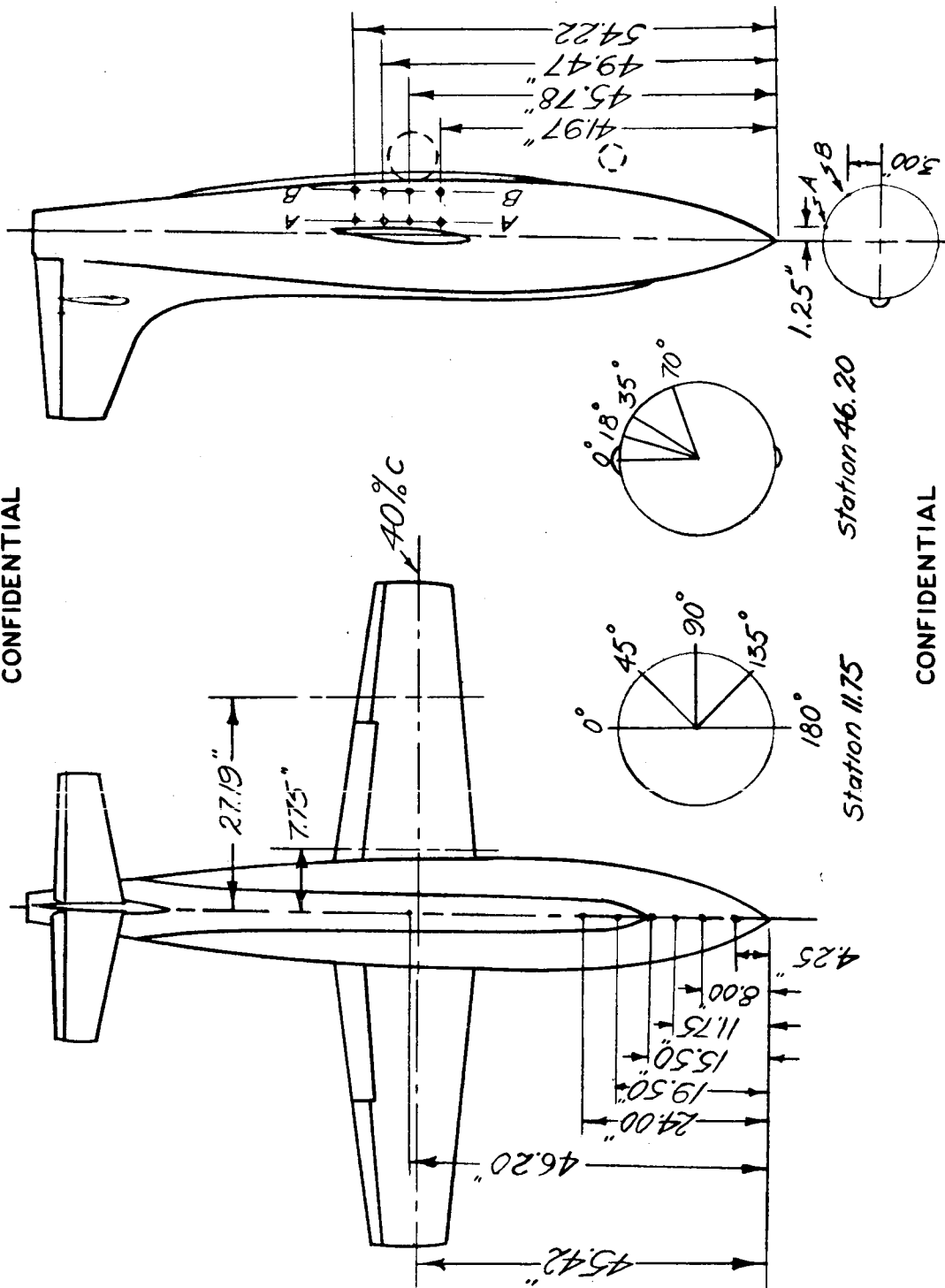


Figure 20 - Effect of spoiler deflection on the lift-drag ratio of the X-5-1 airplane;  $\delta_f = 60^\circ$ ;  $\delta_a = 0^\circ$ ;  $i_e = 0^\circ$ ;  $q = 33.5 \text{ lbs/sq ft}$

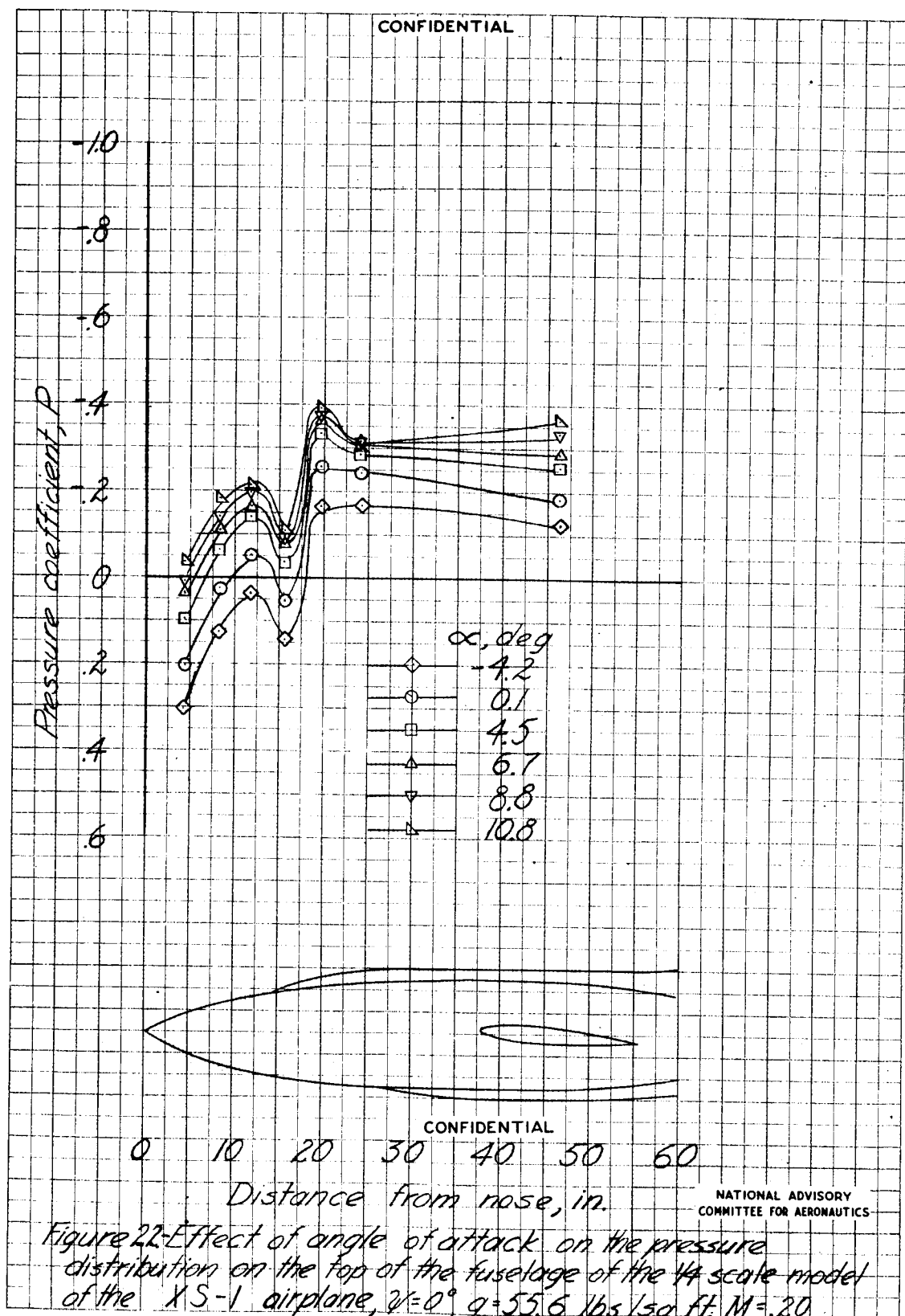
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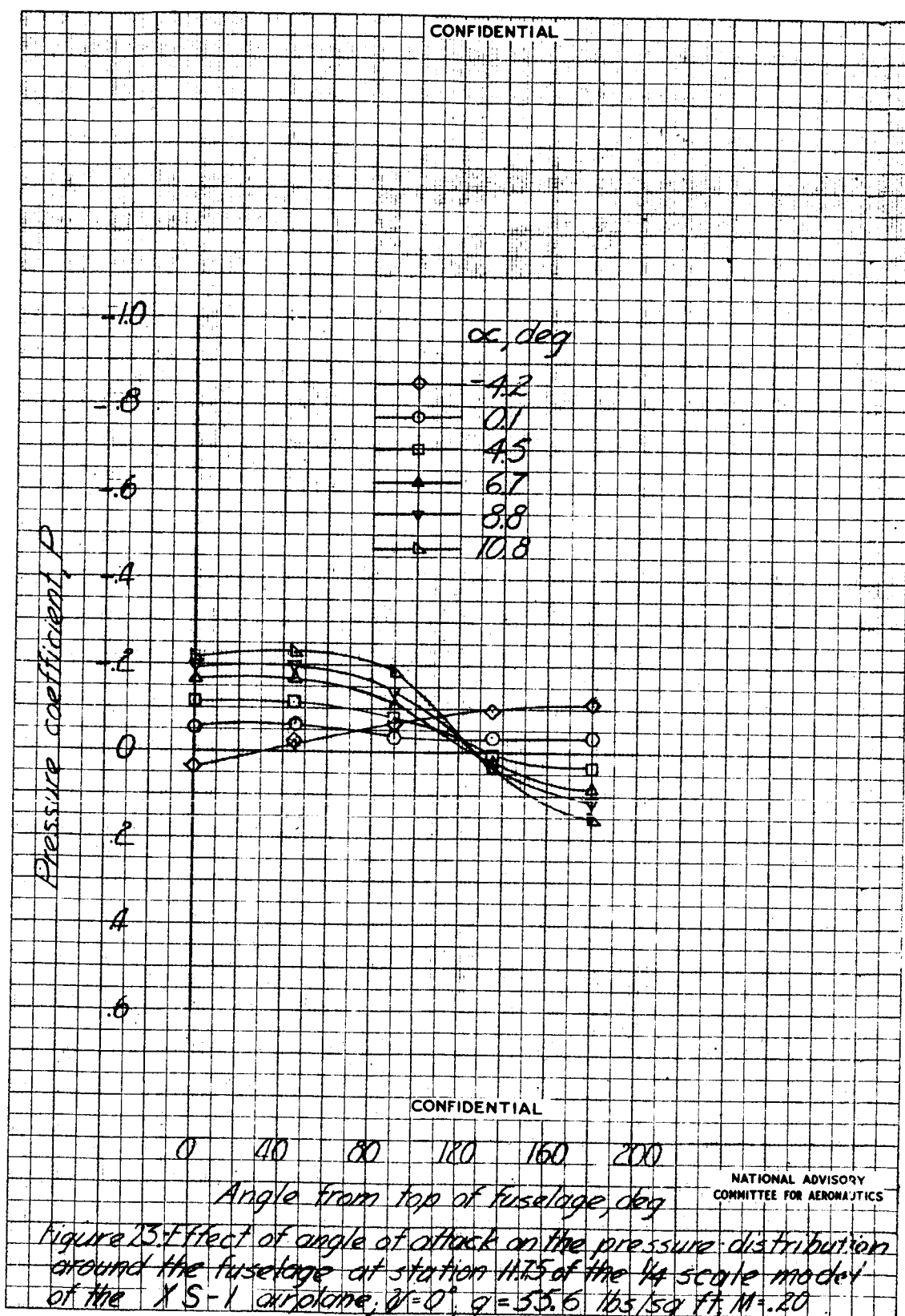


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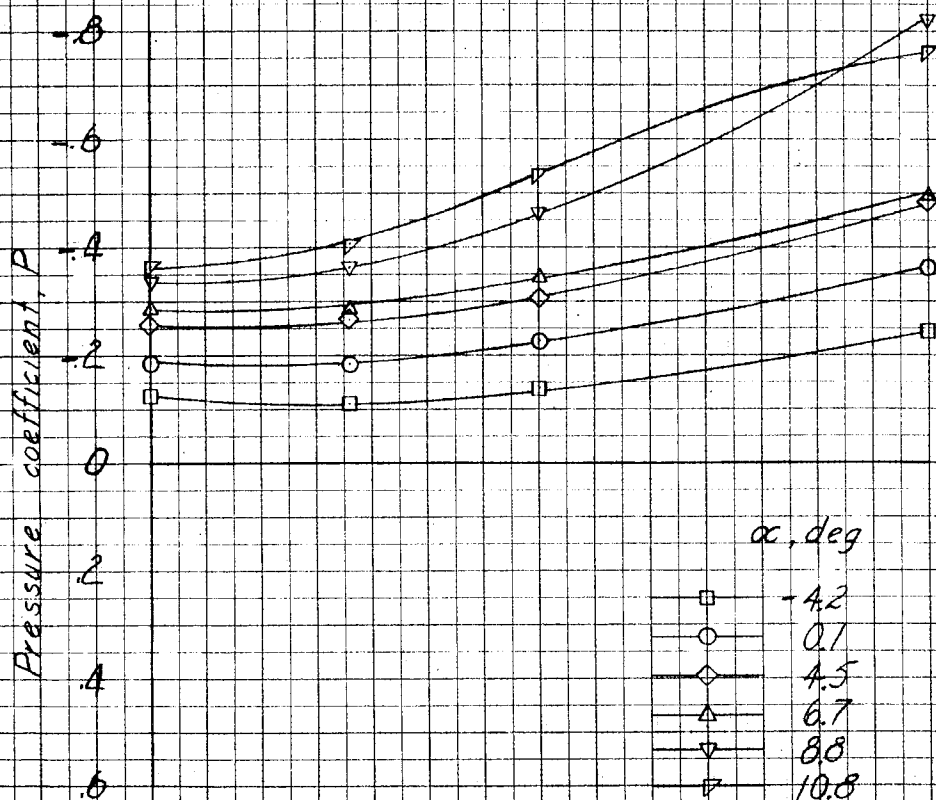
Figure 21.-Location of pressure orifices on the 1/4 scale model of the X-5-1 airplane.







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0 10 20 30 40 50 60 70

Angle from top of fuselage, deg

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Figure 24: Effect of angle of attack on the pressure distribution around the fuselage at station 46.2 of the  $1/4$  scale model of the X-51 airplane;  $\beta = 0^\circ$ ;  $q = 55.6$  lbs/sq ft;  $M = 2.0$

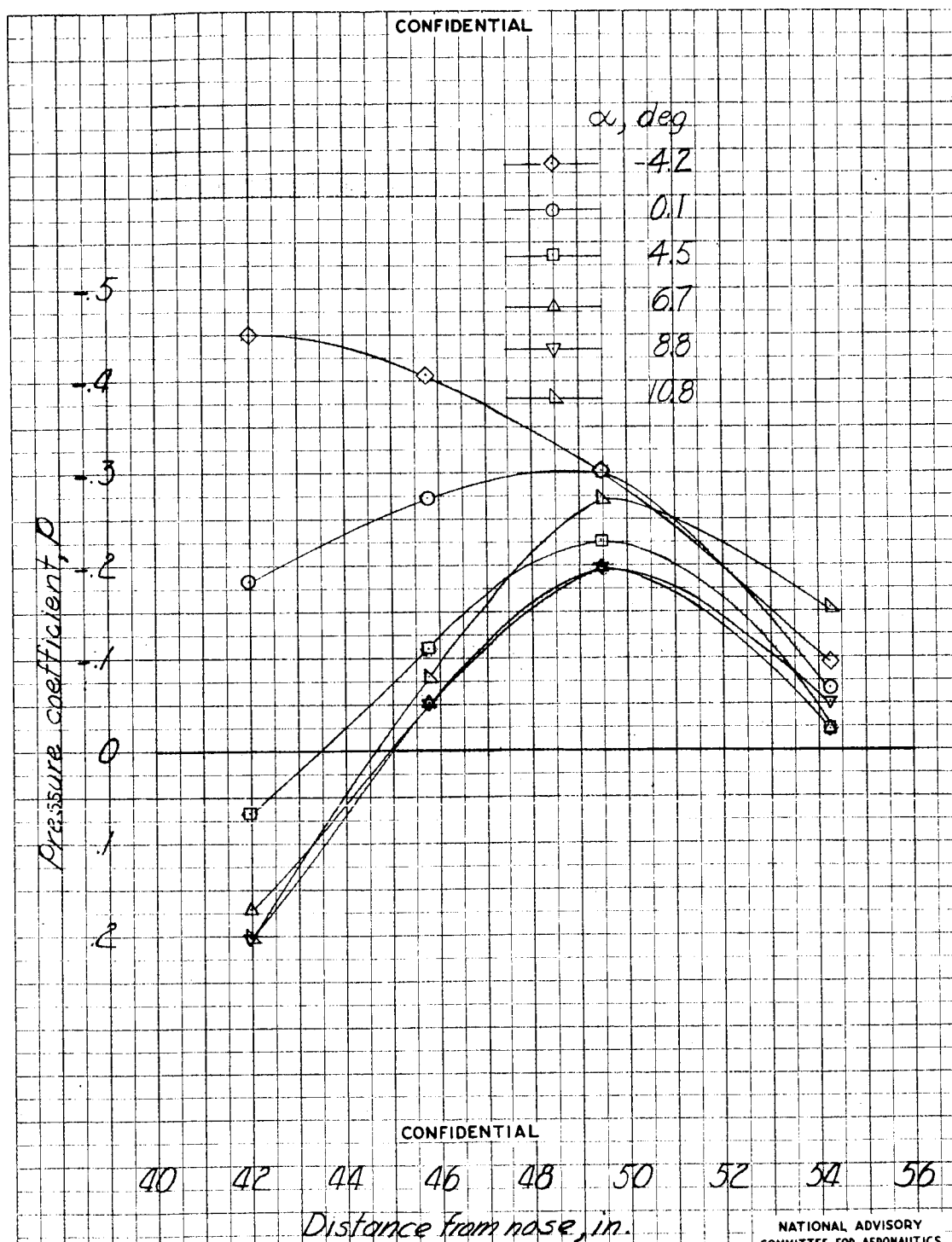
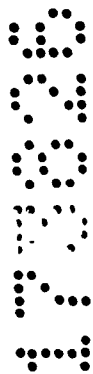


Figure 25: Effect of angle of attack on the pressure distribution along row A of the fuselage of the 1/4 scale model of the XS-1 airplane;  $\beta = 0^\circ$ ;  $\delta_f = 0^\circ$ ;  $q = 55.6$  lbs/sq ft;  $M = 2.0$



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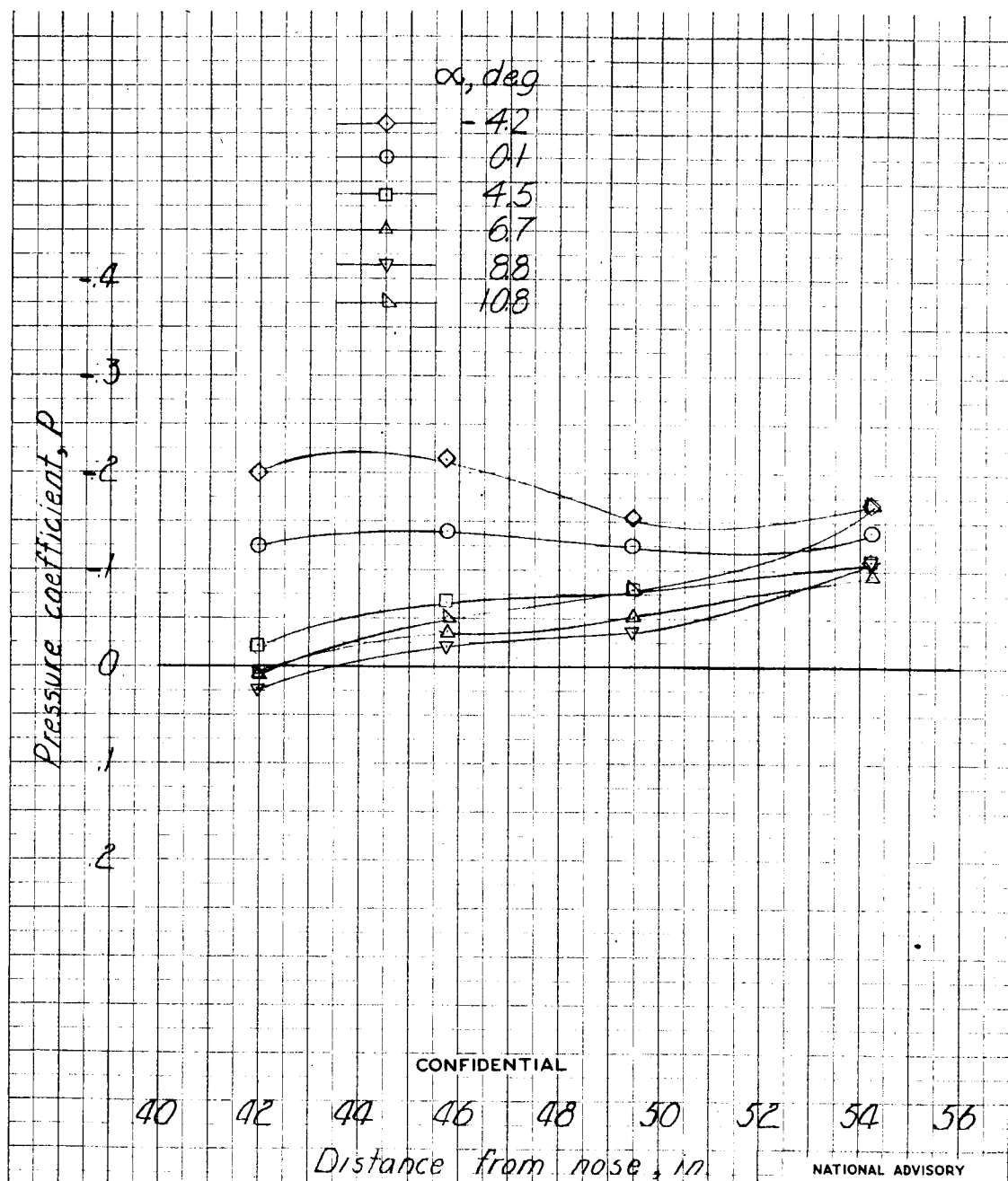
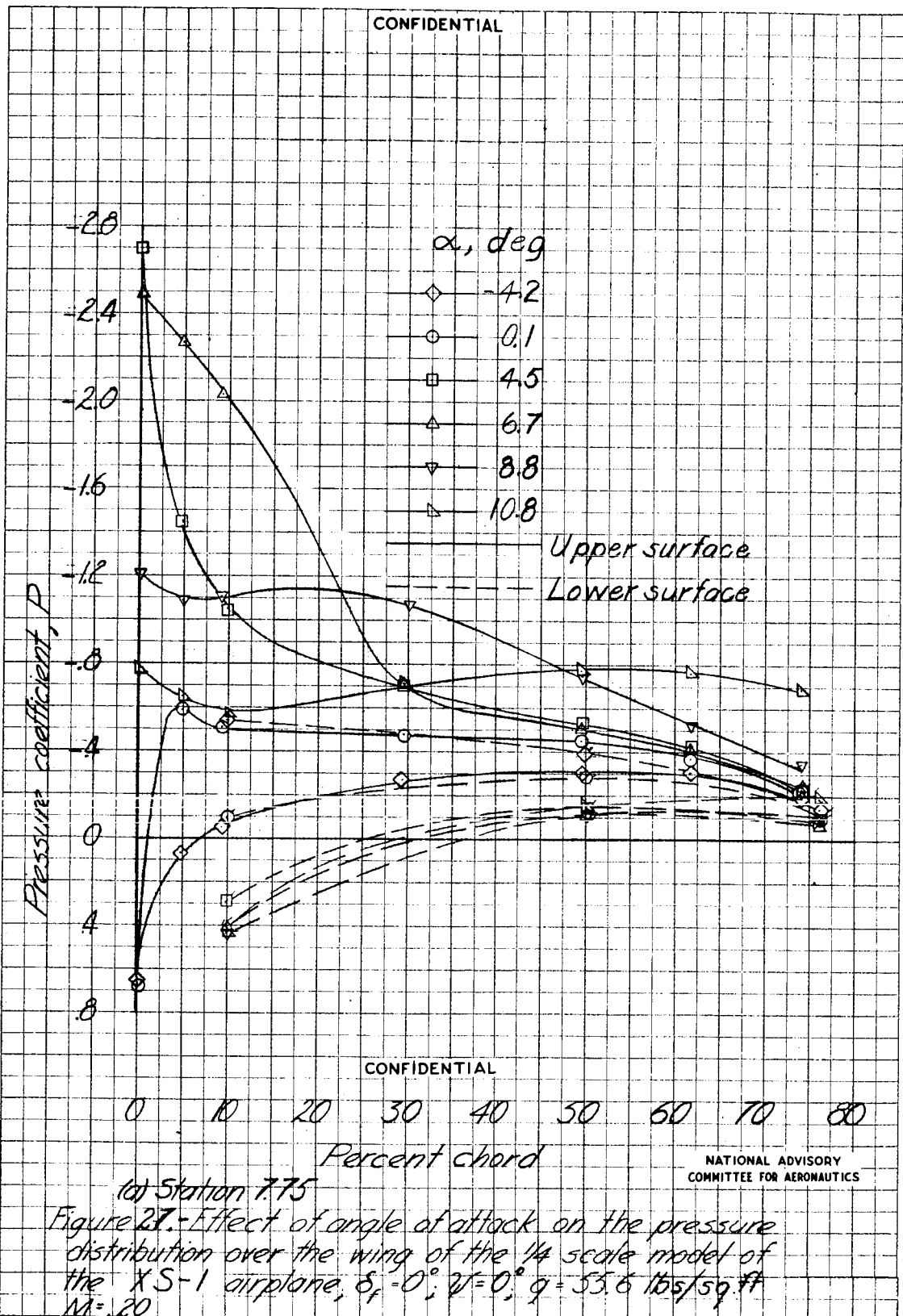
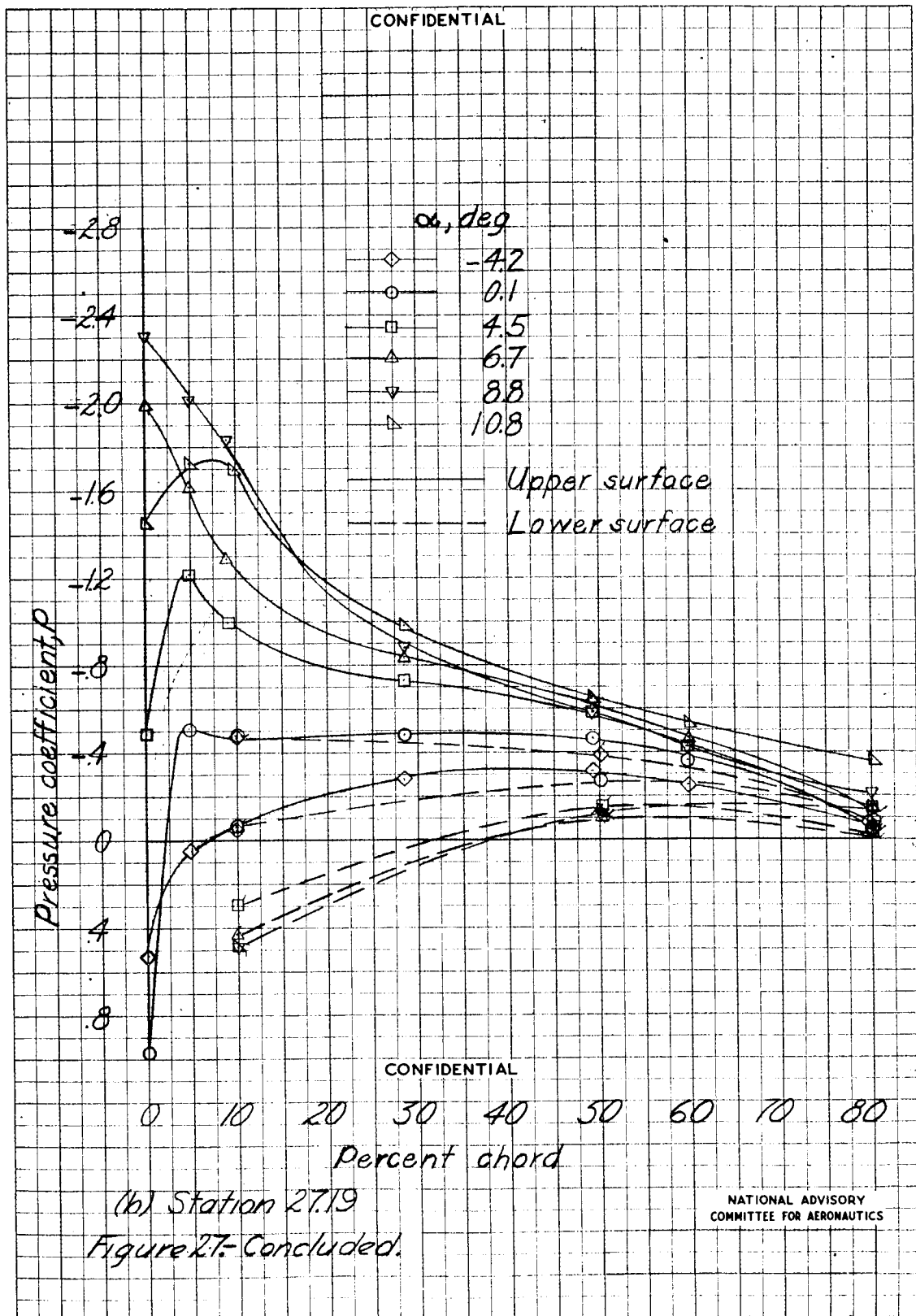
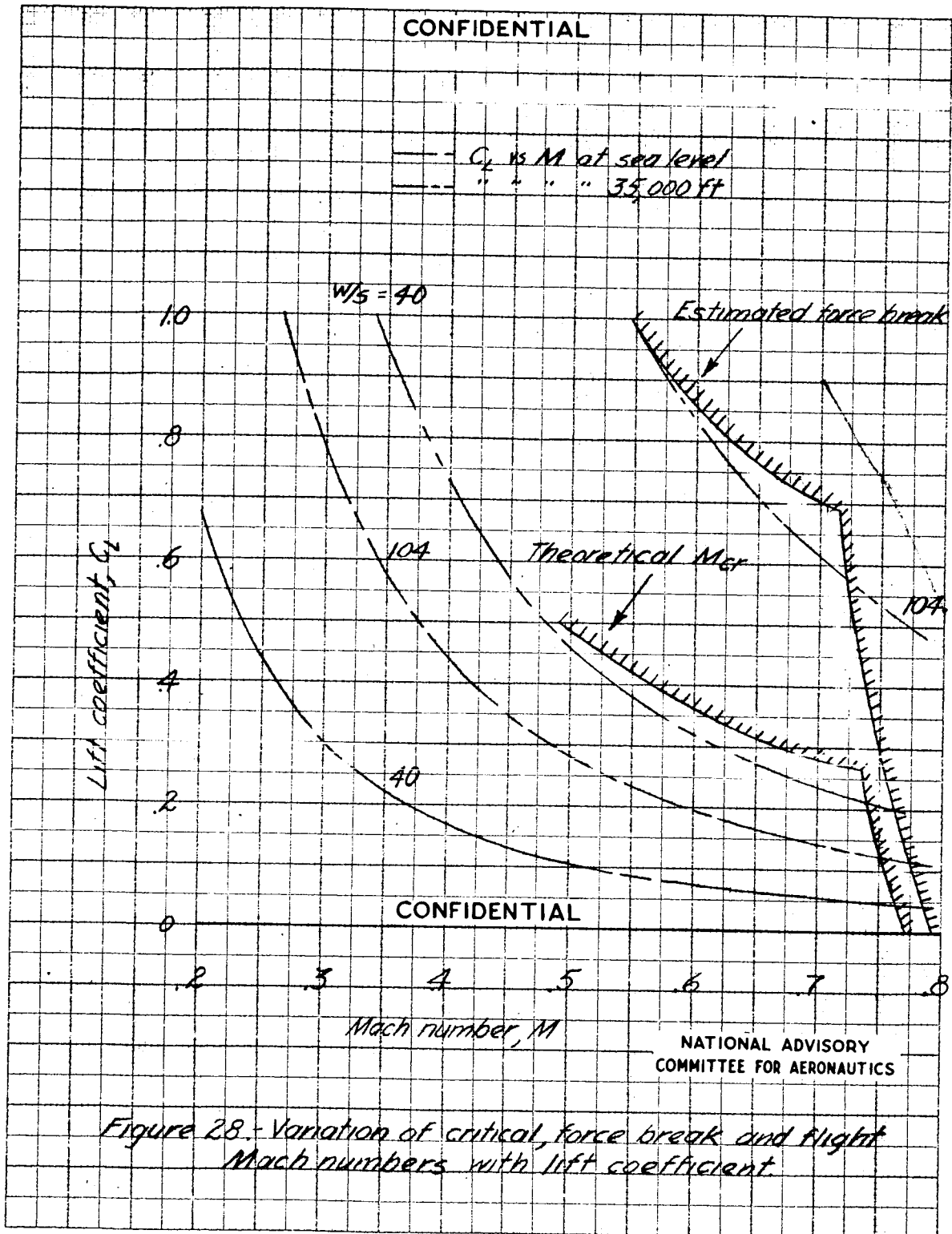


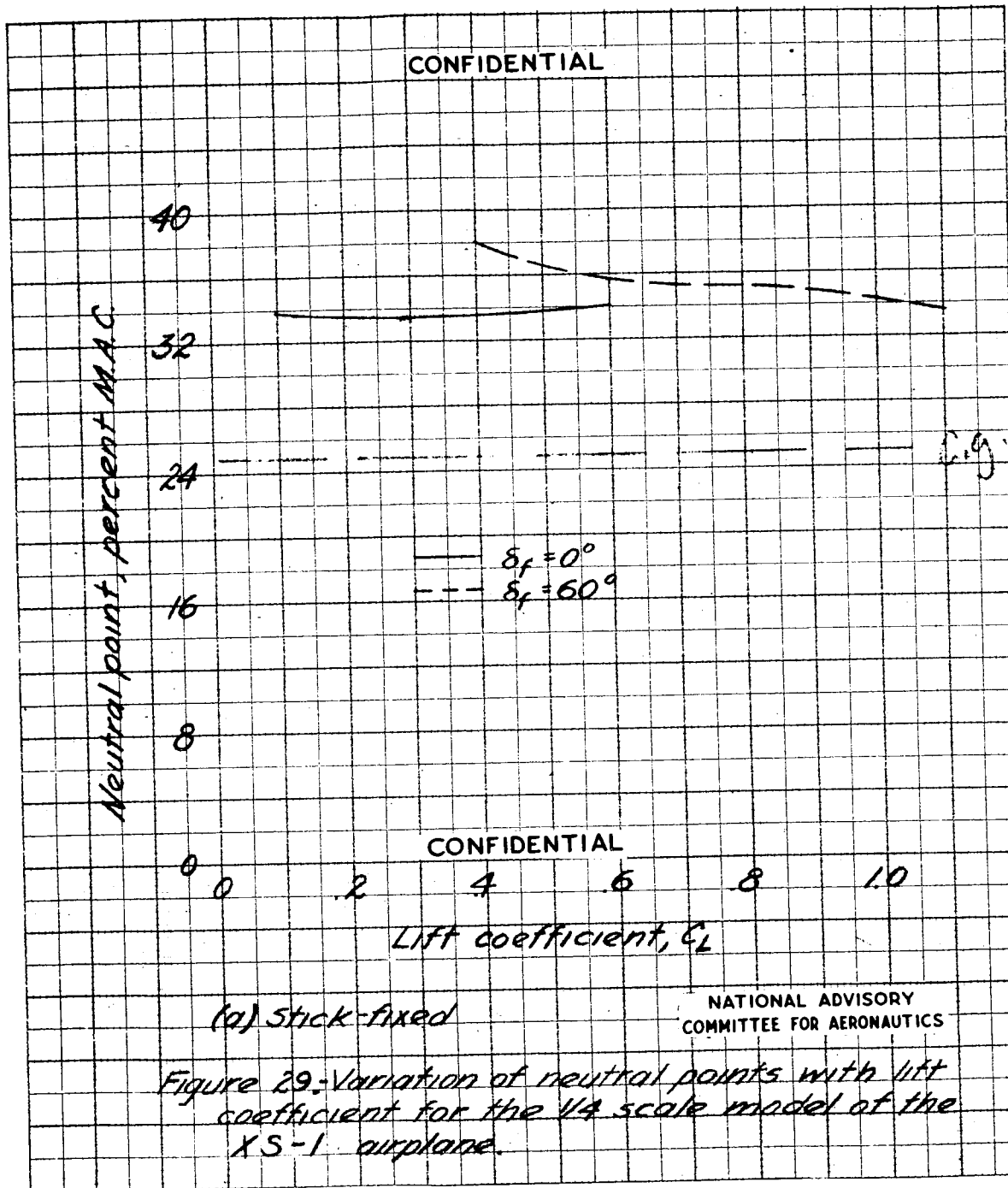
Figure 26. Effect of angle of attack on the pressure distribution along row B of the fuselage of the 1/4 scale model of the XS-1 airplane;  $\delta_f = 0^\circ$ ;  $\delta_r = 0^\circ$ ;  $q = 55.6 \text{ lbs/sq ft}$ ;  $M = 2.0$

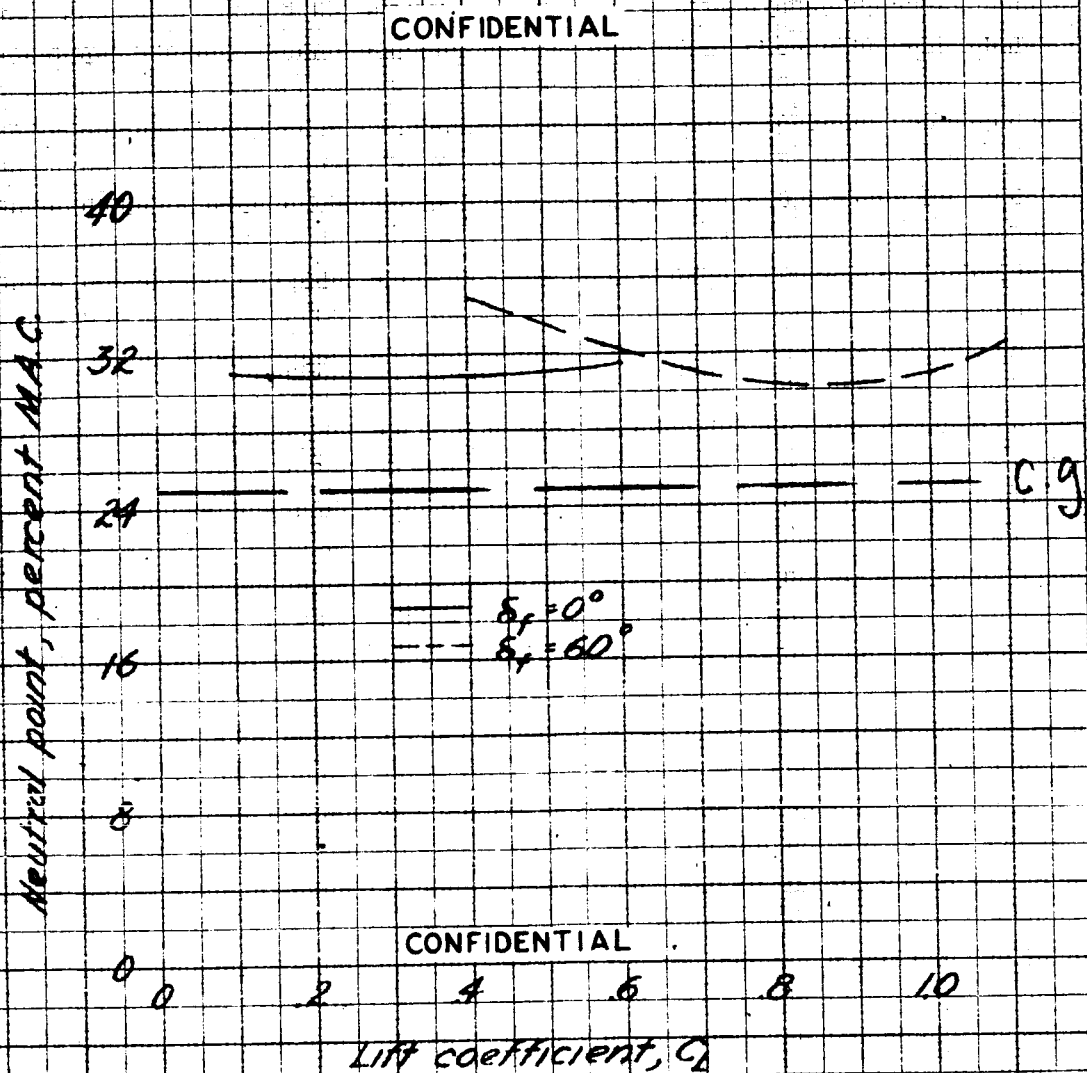












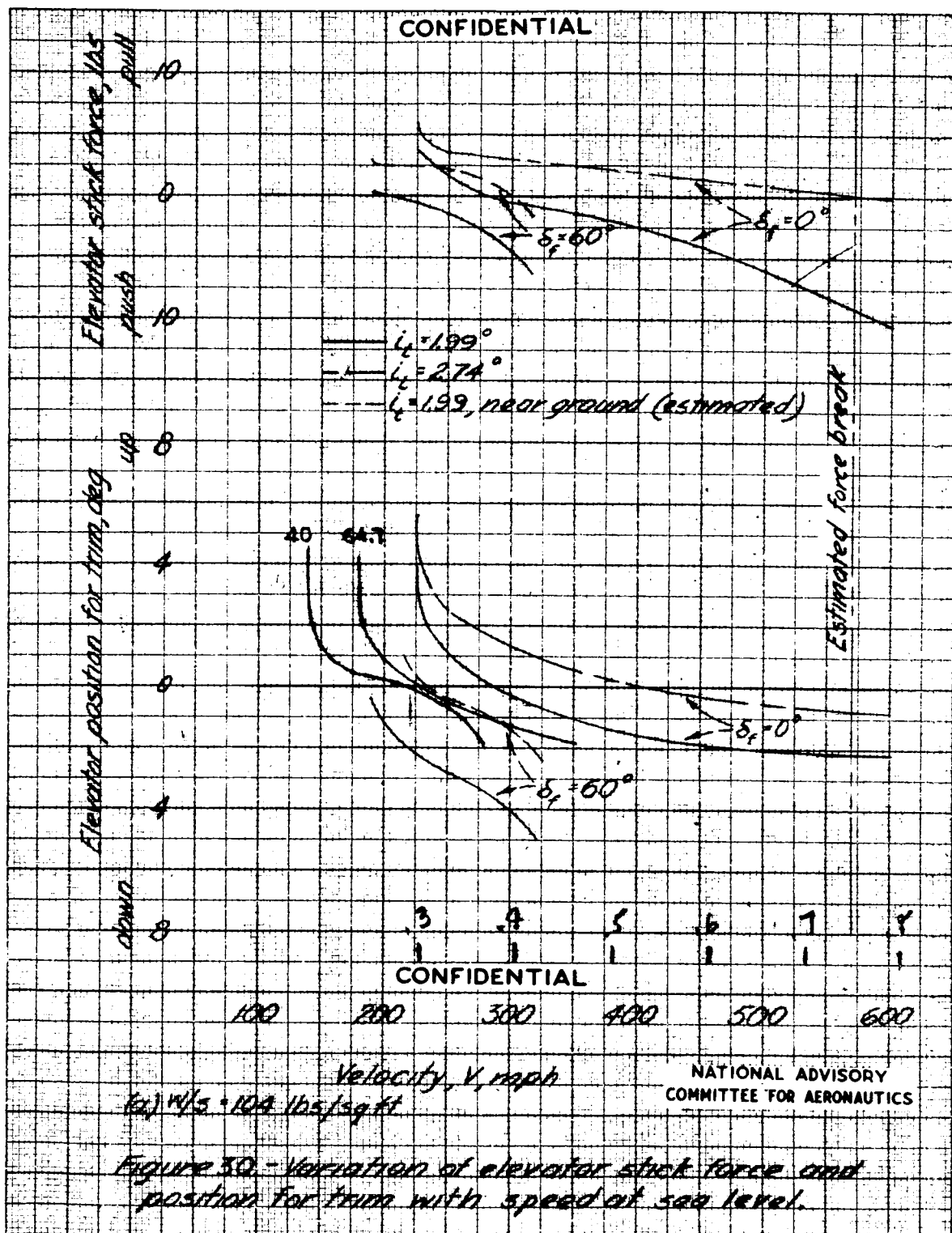
(b) Stick-free

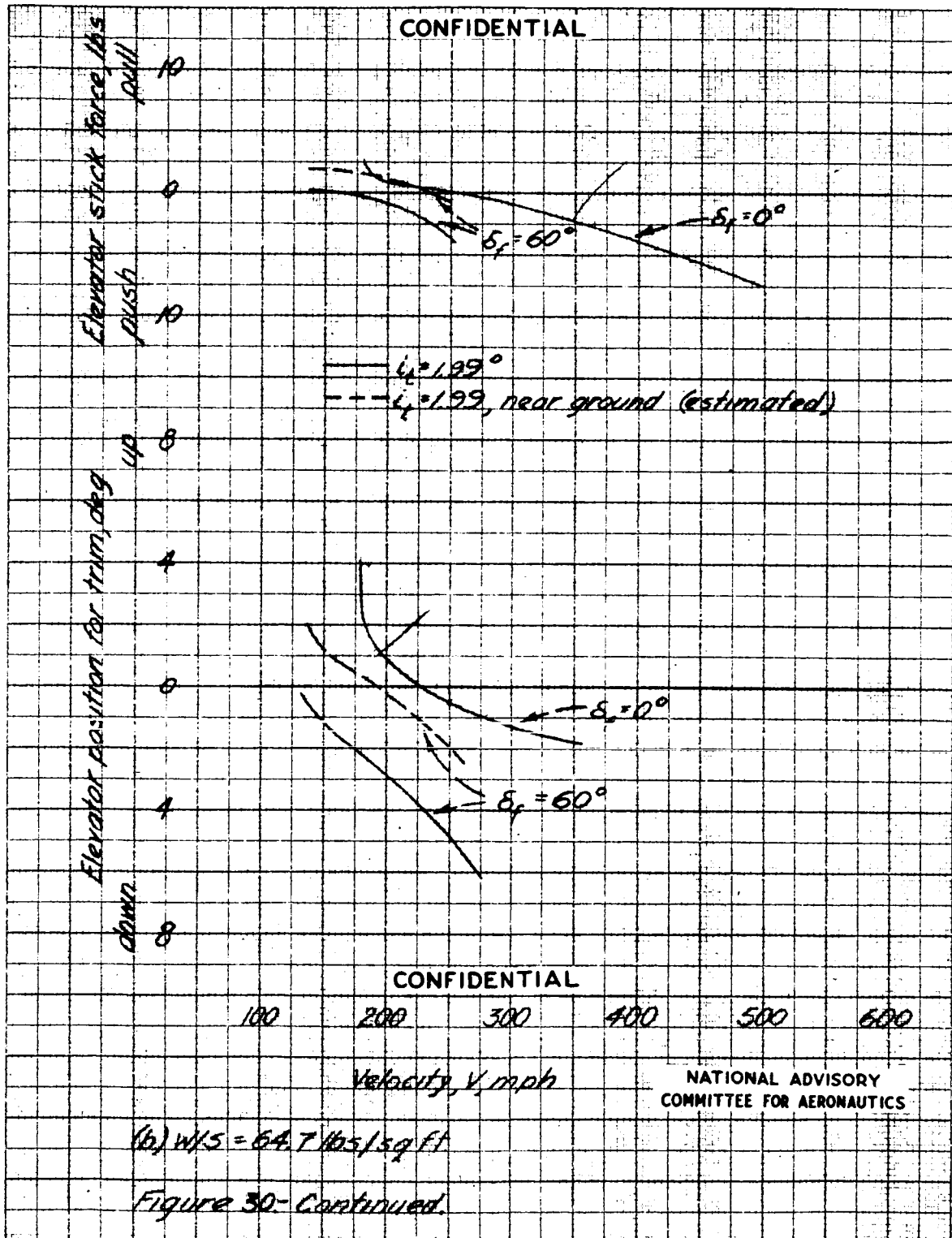
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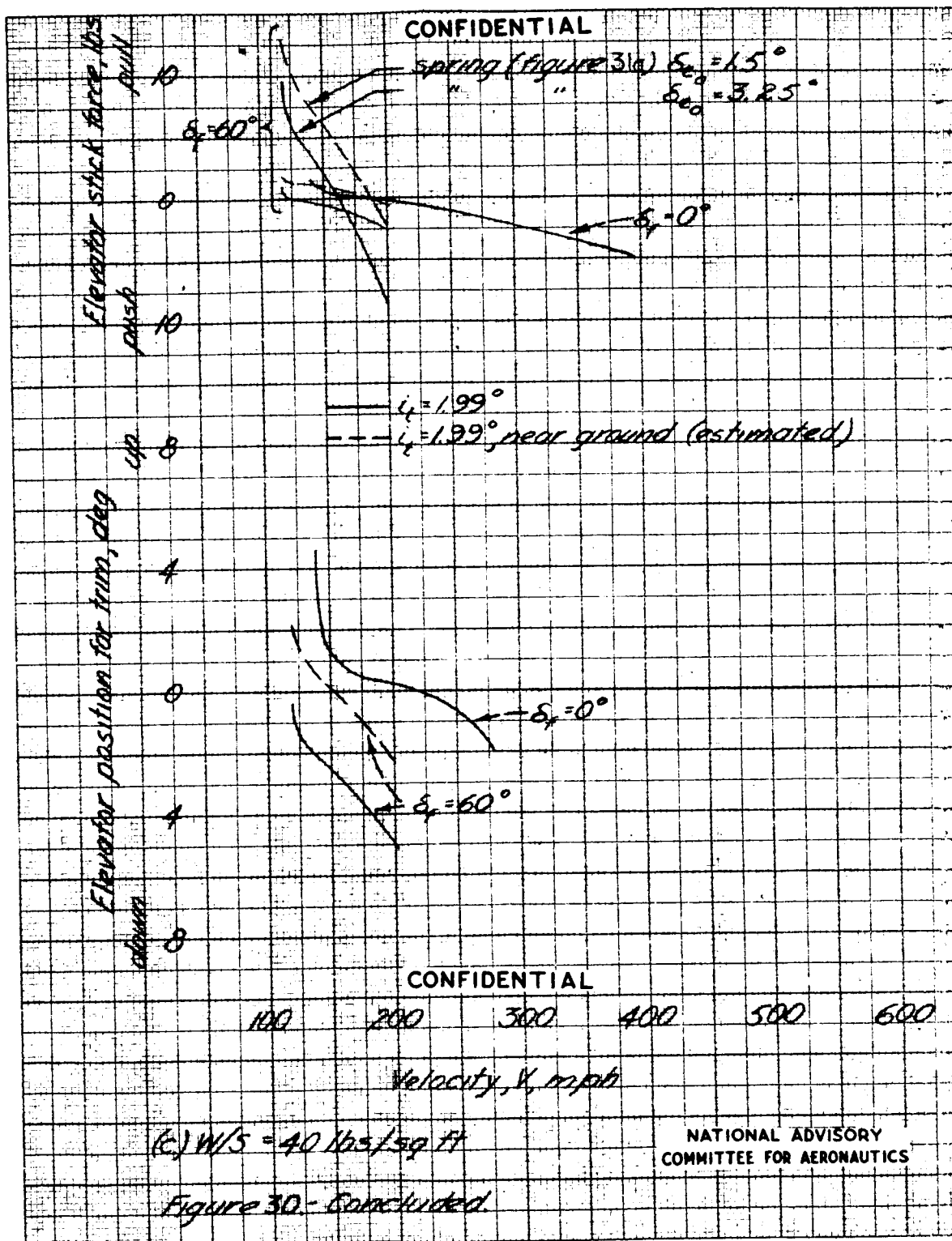
Figure 29: Concluded.

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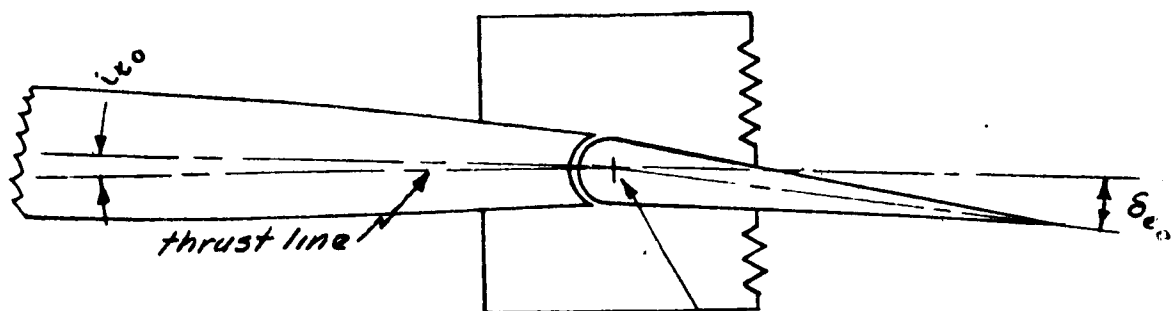
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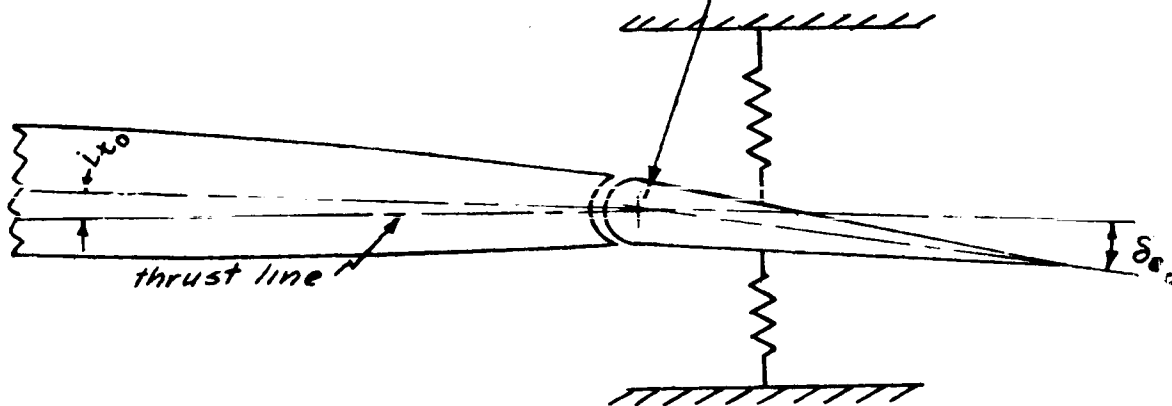
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$$\Delta H = K(\delta_e - \delta_{e_0})$$

$$K = -11.75 \text{ ft-lbs/deg}$$

(a) Spring attached to stabilizer.



$$\Delta H = K(\delta_e - \delta_{e_0} + i_t - i_{t_0})$$

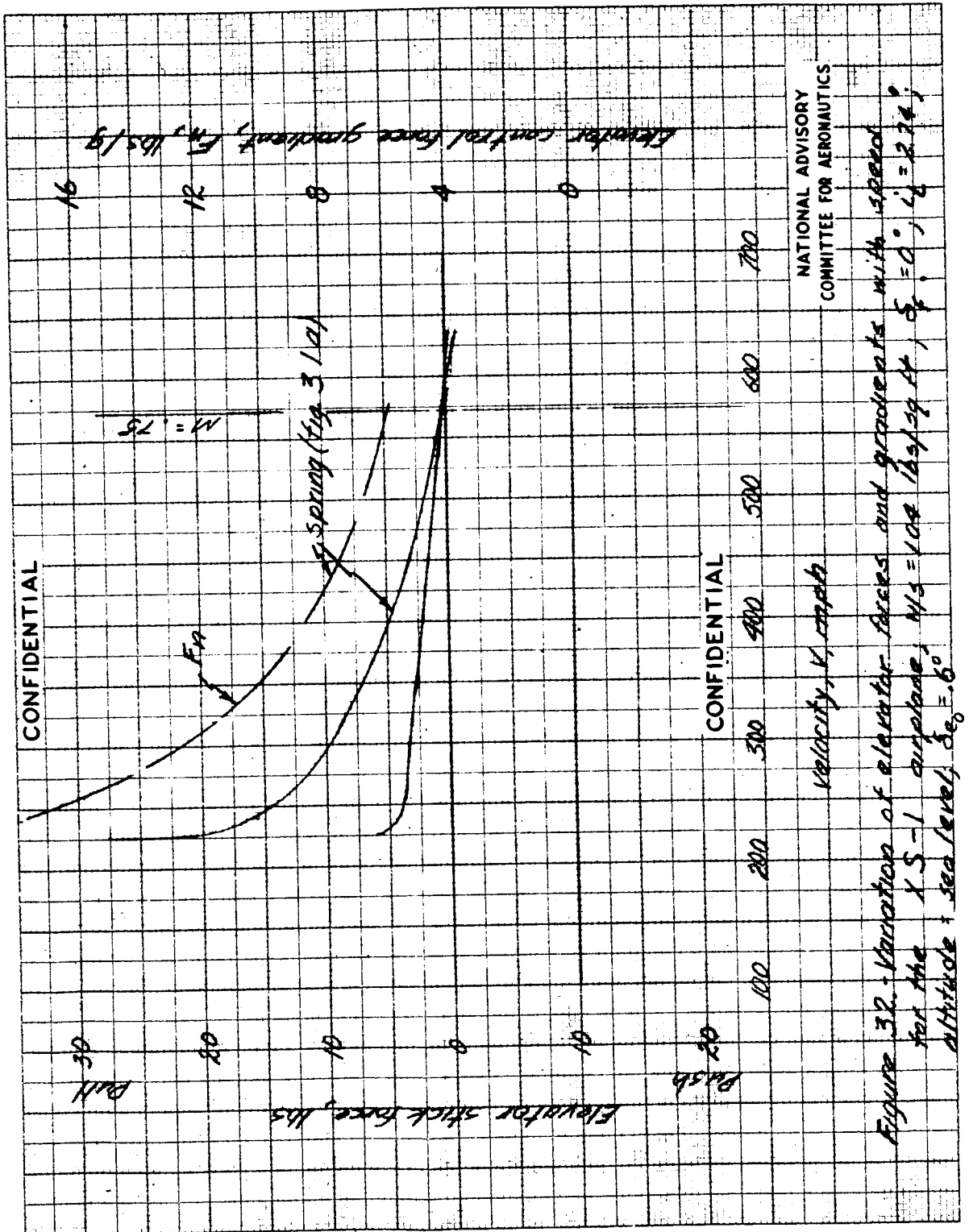
$$K = -11.75 \text{ ft-lbs/deg}$$

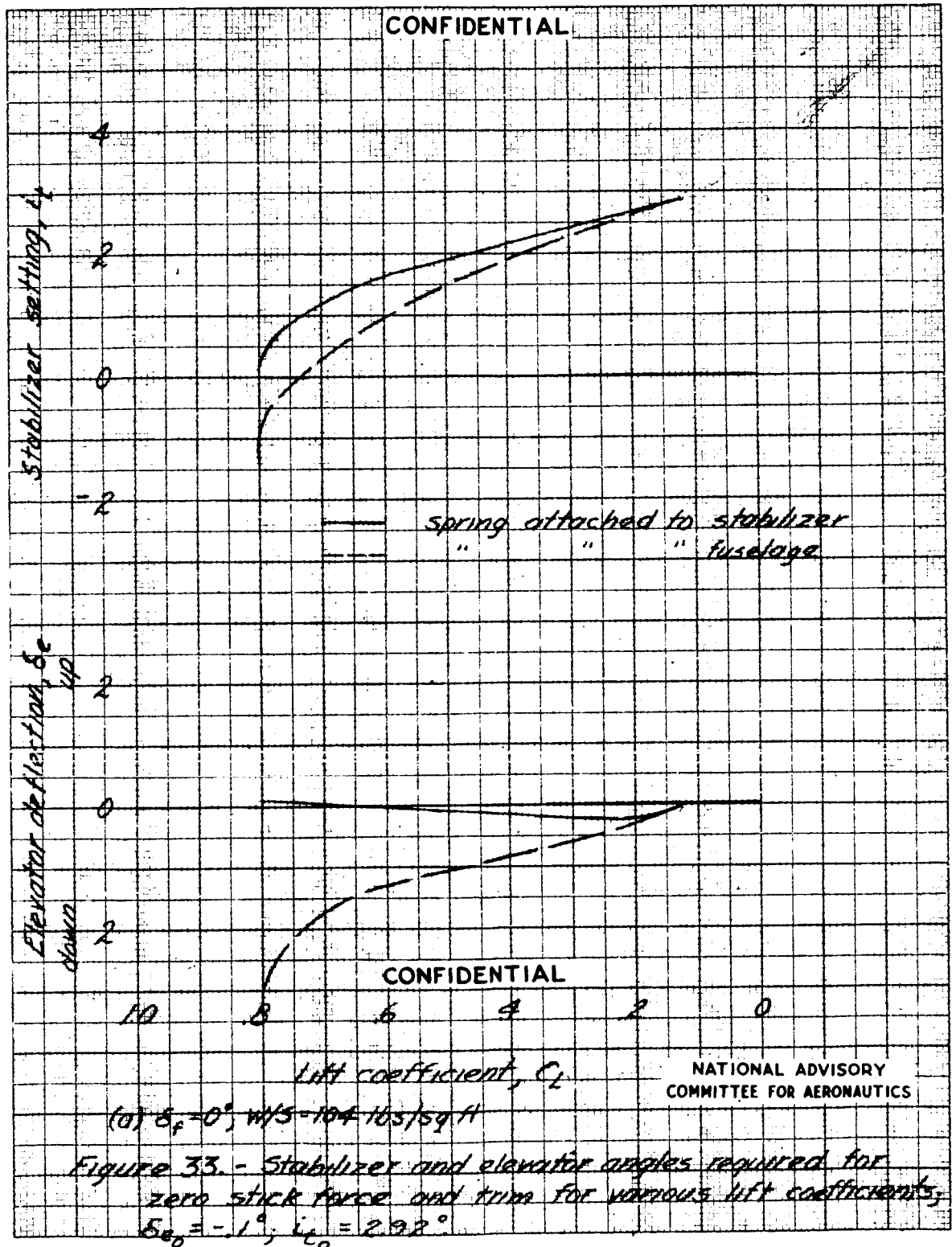
(b) Spring attached to fuselage.

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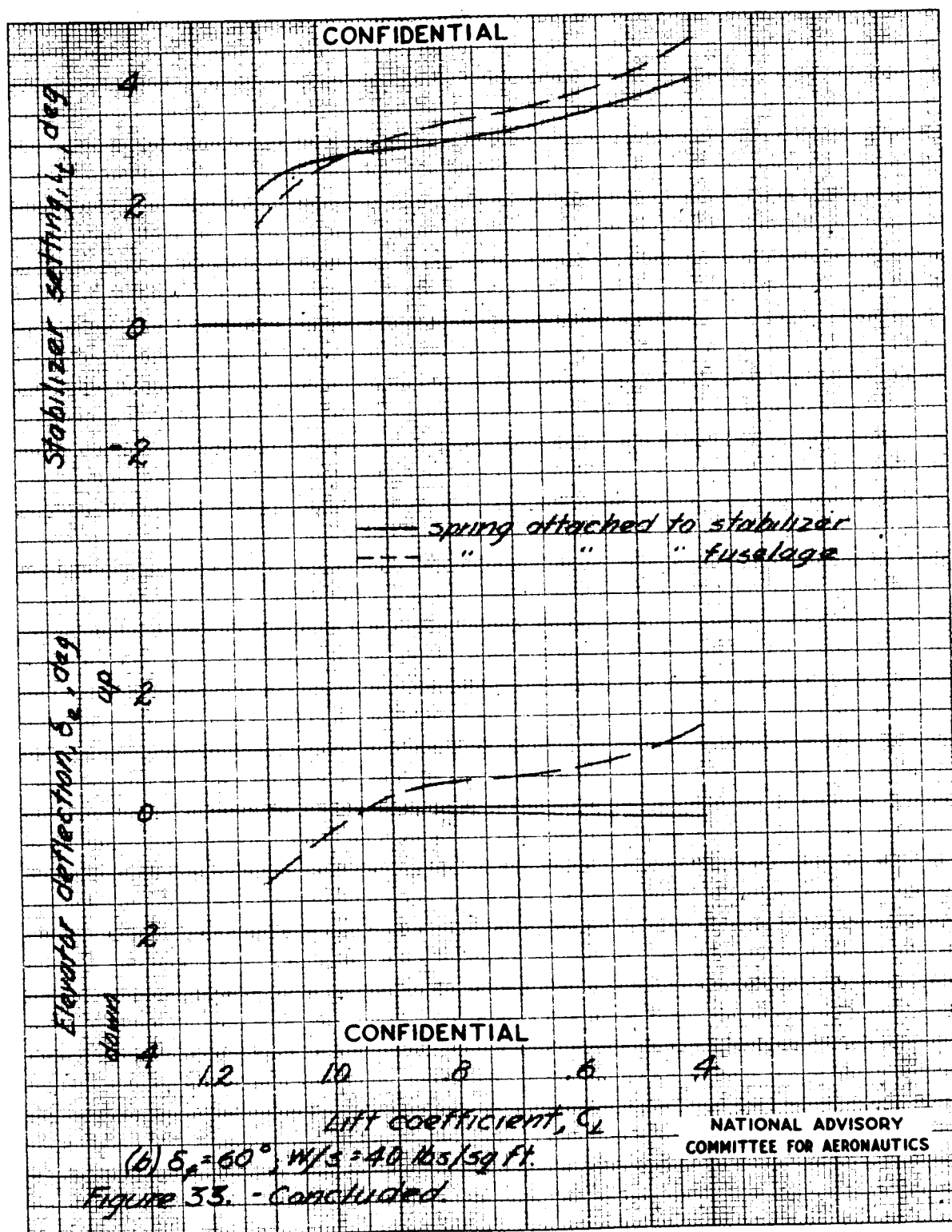
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Figure 31.- Schematic arrangements for spring loaded elevators.

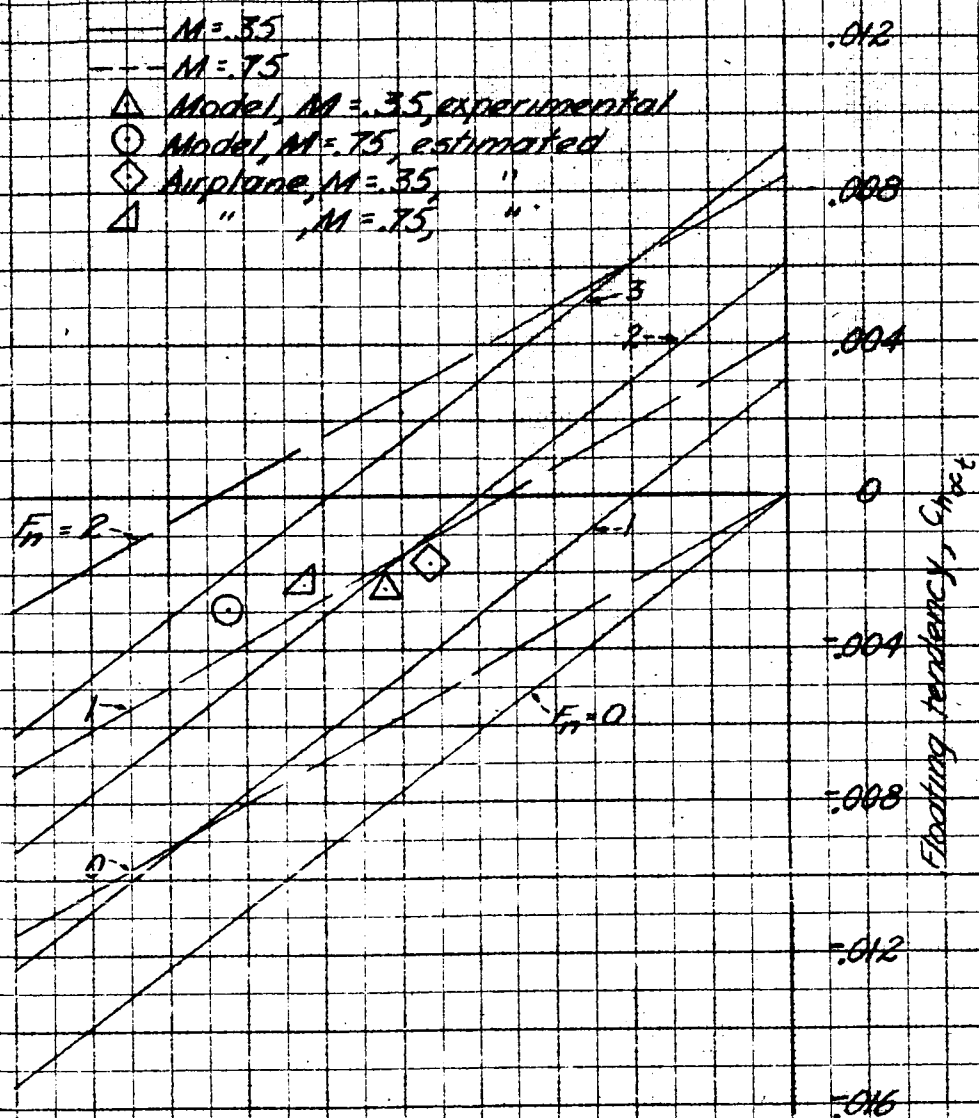








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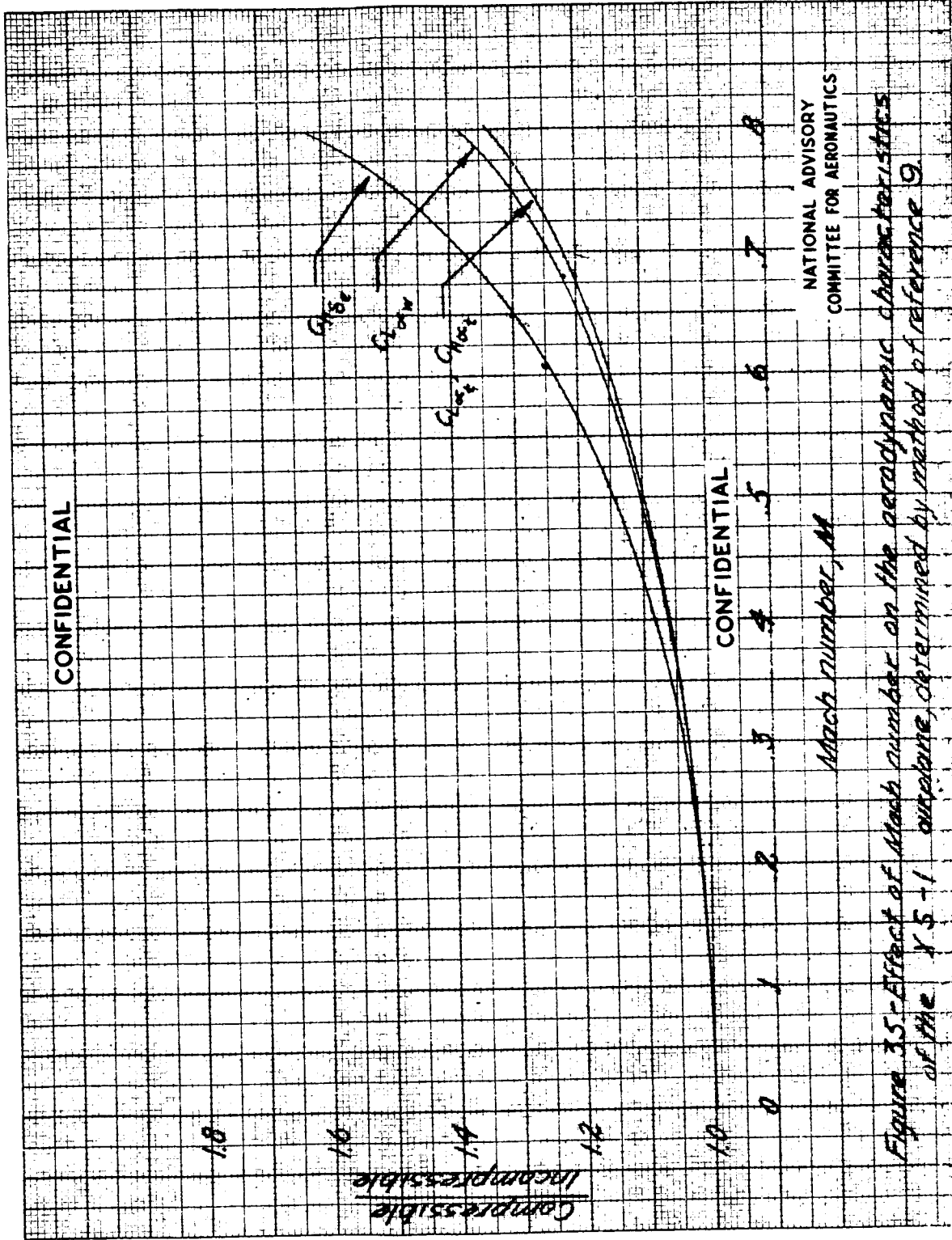
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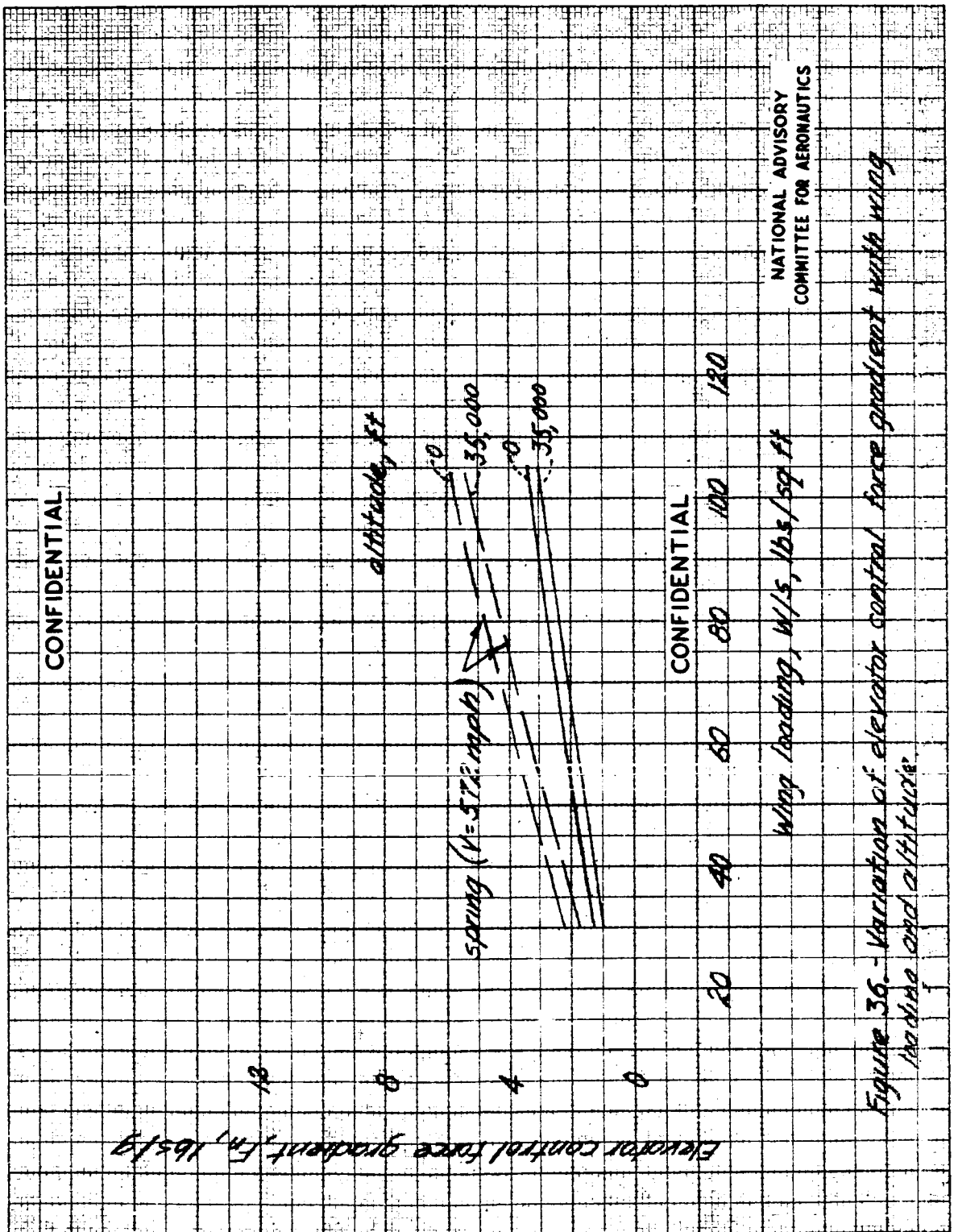
-0.20 -0.16 -0.12 -0.08 -0.04 0

Restoring tendency,  $Ch_{\delta e}$ NATIONAL ADVISORY  
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Figure 34.- Variation of elevator control force gradient with hinge moment parameters and speed;  $w/s = 64.7$  lbs/sq ft, altitude = 35,000 feet.

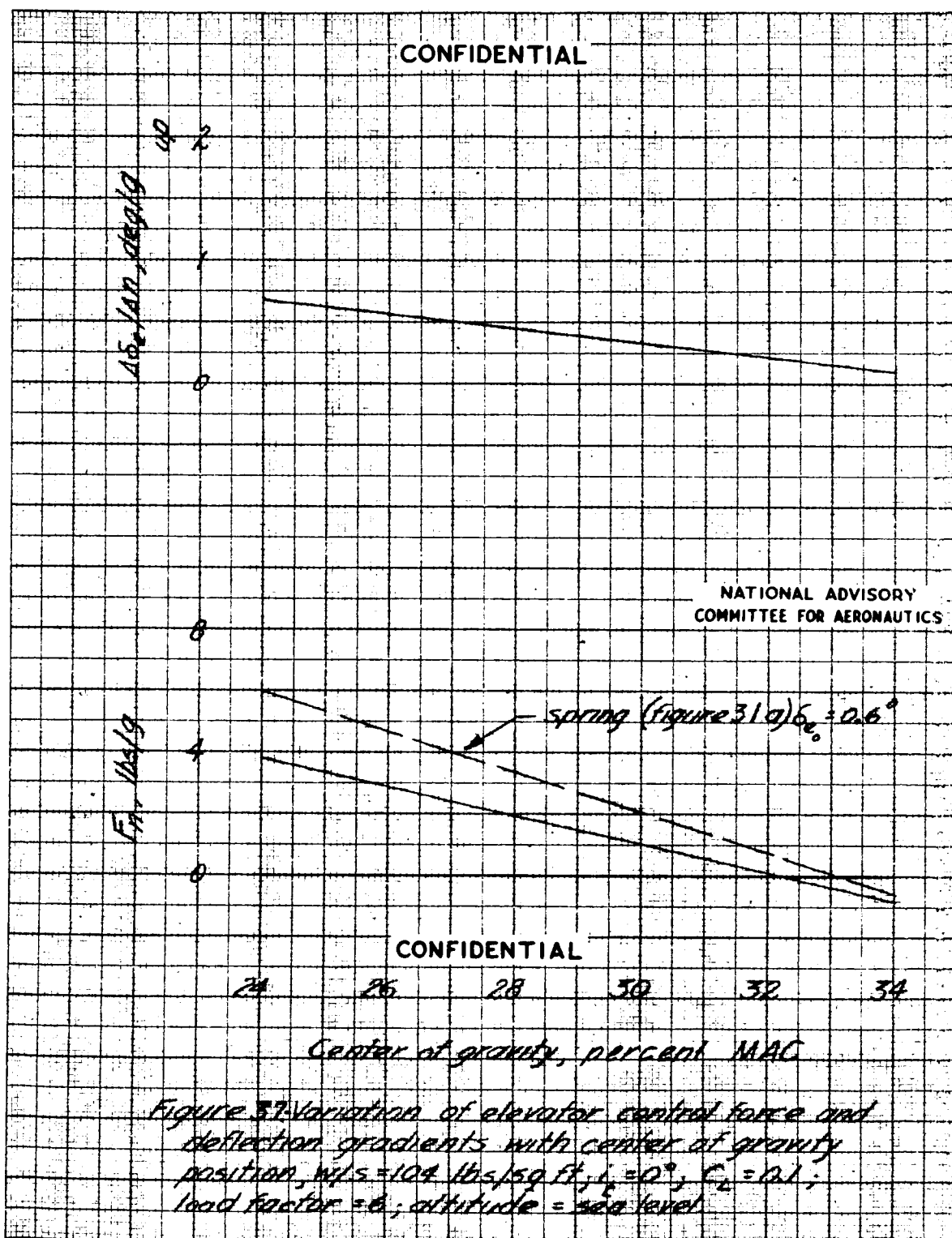
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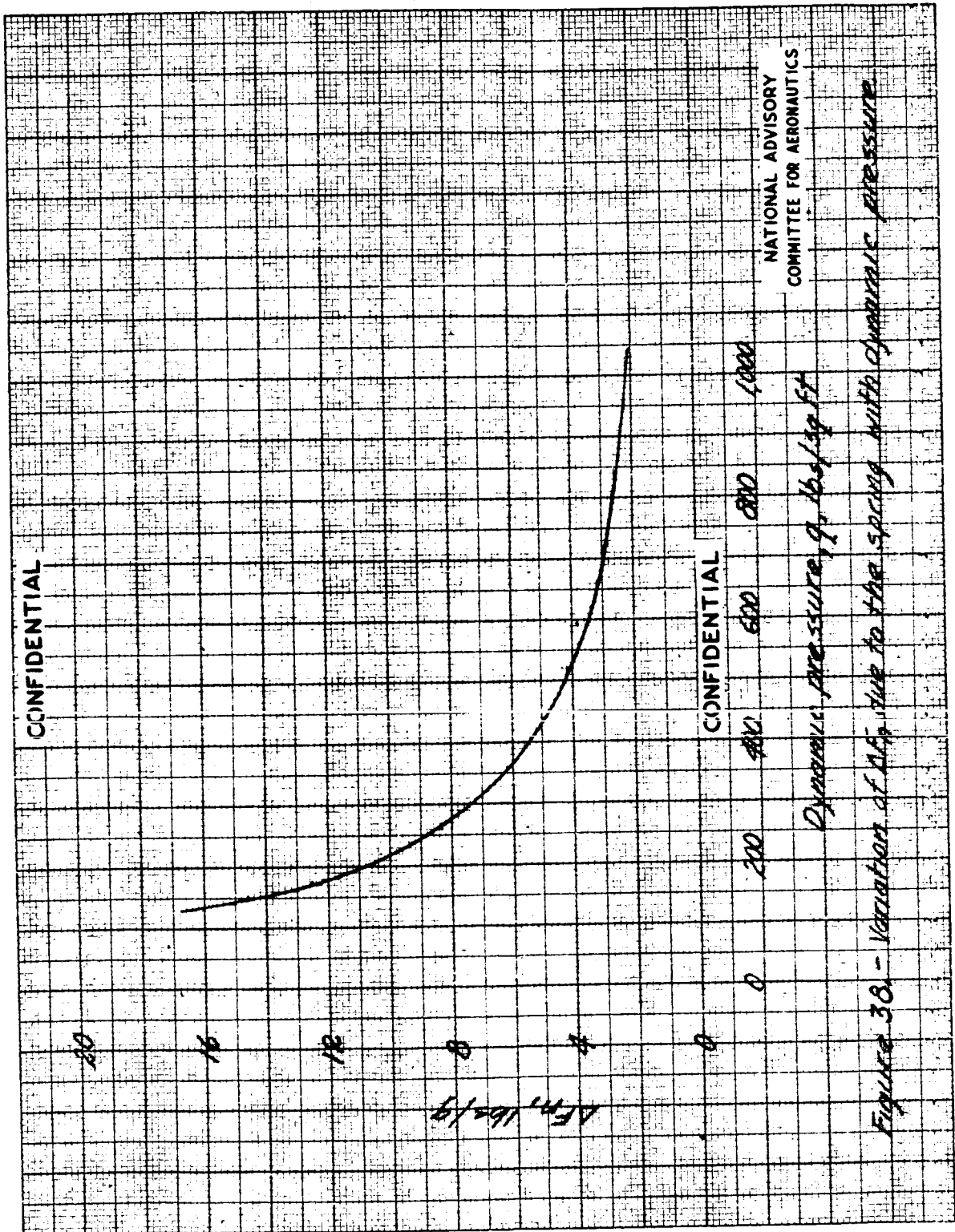


Figure 38 - Variation of  $\Delta P_p$  due to the spring with dynamic pressure

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# RESEARCH MEMORANDUM

CURRENT STATUS OF LONGITUDINAL STABILITY

By

Charles J. Donlan

Langley Memorial Aeronautical Laboratory  
Langley Field, Va.

CLASSIFICATION CANCELLED

CLASSIFIED DOCUMENT

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## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## RESEARCH MEMORANDUM

## CURRENT STATUS OF LONGITUDINAL STABILITY

By Charles J. Donlan

## SUMMARY

The problems of static and dynamic longitudinal stability both at high speeds and at low speeds are discussed and data are presented which indicate recent progress made in the solution of these problems.

It is shown that the incorporation of large amounts of sweepback on both the wing and the horizontal tail can significantly increase the Mach number at which critical trim changes and stability changes occur and can greatly reduce the trim changes and stability changes encountered at supercritical speeds. Data are also presented which demonstrate the possibility of obtaining satisfactory longitudinal stability in the landing configuration for wings with sweepback of the order of  $45^\circ$  utilizing various stall-control devices. Optimum arrangements for such devices, however, should be determined experimentally at the present time.

## INTRODUCTION

The purpose of this paper is to focus attention on some recent investigations that have been concerned with longitudinal-stability problems both at high speeds and at low speeds and to summarize briefly the current state of affairs in regard to these problems.

## SYMBOLS

|   |                                        |
|---|----------------------------------------|
| W | weight                                 |
| M | Mach number ( $V/a$ ); pitching moment |
| V | velocity                               |
| a | speed of sound                         |

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|            |                                                             |
|------------|-------------------------------------------------------------|
| AR         | aspect ratio ( $b^2/S$ )                                    |
| b          | span                                                        |
| S          | wing area                                                   |
| $C_L$      | lift coefficient ( $L/qS$ )                                 |
| q          | dynamic pressure ( $\frac{1}{2}\rho V^2$ )                  |
| $q_t$      | dynamic pressure at tail                                    |
| $\rho$     | density                                                     |
| c          | chord                                                       |
| $C_m$      | pitching-moment coefficient ( $M/qcS$ )                     |
| $\epsilon$ | downwash at tail                                            |
| P          | period of short-period longitudinal oscillation, seconds    |
| $T_{1/2}$  | time for oscillation to damp to one-half amplitude, seconds |
| $i_t$      | stabilizer incidence                                        |
| $\delta$   | control-surface deflection                                  |
| $\alpha$   | angle of attack                                             |

#### HIGH-SPEED PROBLEMS

##### Static Stability and Control

Recent investigations.— A number of longitudinal-stability investigations of various airplane configurations have been conducted at high subsonic Mach numbers in the high-speed wind tunnels of the NACA and at transonic Mach numbers up to 1.2 utilizing the NACA wing-flow method and the associated wind-tunnel transonic-bump technique. A number of these investigations are reported in references 1 to 7, and some of the configurations investigated, together with the Mach number range covered, are summarized in figure 1.

For the tailless configuration (a), data were obtained in the Langley high-speed 7- by 10-foot tunnel for a sting-supported model and

also for a semispan model up to a Mach number of 0.95. Wing-flow data were also obtained up to a Mach number of 1.2. The three sets of data are in general qualitative agreement, although the increase in the lift-curve slope with Mach number was somewhat more rapid for the sting-supported tunnel model than for the semispan tunnel model and semispan wing-flow model.

Configuration (b) was investigated as a semispan wing-flow model (reference 1) and was also tested on a transonic bump in the Langley high-speed 7- by 10-foot tunnel. This model is similar to the XS-1 model for which Langley 8-foot high-speed-tunnel data are available to a Mach number of 0.92 (references 2 and 3). The agreement between the data obtained by the wing-flow method and the transonic-bump method was satisfactory throughout most of the Mach number range.

Model (d) was similar to model (b) except for the swept tail. It also was tested as a wing-flow model (reference 4).

Model (c) was investigated on the transonic bump; model (e), as a semispan model in the Ames 16-foot tunnel (reference 5); and model (f), as a sting-supported model in the Langley 8-foot high-speed tunnel (references 6 and 7).

Despite the fact that most of the results available thus far are limited to relatively few configurations, it is interesting to observe in the data certain trends in regard to the manner in which stability and trim changes with Mach number are manifested.

Characteristic data.— Data representative of the variation of pitching-moment coefficient with lift coefficient for several Mach numbers for a straight-wing design are shown in figure 2. Although these data apply to the design indicated, similar trends in the data for other straight-wing designs have been observed. The data at  $M = 0.600$  are typical of the behavior before force break, and some comments regarding the predicability of the characteristics in this range are probably pertinent at this point.

The important changes in longitudinal stability for straight-wing designs at high Mach numbers are, of course, not indicated by formulas based on linear-perturbation theory. Such formulas, however, are useful in interpreting experimental trends at subcritical Mach numbers. In consideration of the Mach number effects on a wing and tail combination, the trends indicated by the theory may be divided into three categories: (1) direct changes in the position of the wing aerodynamic center, (2) changes in the downwash at the tail, and (3) disproportionate changes in the lift-curve slopes of the wing and tail resulting from the differences

in aspect ratio. For a flat elliptic wing of aspect ratio 4, theory indicates a forward shift of the aerodynamic center of only about 1.4 percent at a Mach number of 0.8 (reference 8). However, forward shifts of the aerodynamic center of 5 percent or more have been obtained experimentally on straight wings at high Mach numbers, particularly for those employing sections having large trailing-edge angles. At the present time, therefore, it appears that the changes in wing aerodynamic-center position with Mach number must be determined experimentally even at subcritical speeds. A limited amount of German data has indicated that this effect is minimized for small trailing-edge angles.

The theories regarding the change in downwash characteristics at the tail and the change in the lift-curve slopes of the wing and tail with Mach number, however, appear to agree fairly well with experiment at subcritical Mach numbers (references 9 and 10). These two effects have indicated forward shifts in the neutral point of the order of 5 percent in some cases. At Mach numbers approaching that of force break and at supercritical Mach numbers, recourse must be made to experiment.

Marked changes in the variation of the basic wing-fuselage pitching moment with lift coefficient are apparent at a Mach number of 0.905 and 0.933, and the appearance of flat spots in the resultant pitching-moment curve in the lower lift range is somewhat characteristic for this type of design at supercritical speeds. In many instances local reversals in slope have been encountered, particularly for different stabilizer and elevator settings. The nonparallelism of the pitching-moment curves in this range for the different stabilizer settings is significant and evidences the nonlinear contribution of the tail to stability. Consequently, in evaluating the stability characteristics of a design possessing nonlinearities of this kind, it is essential, of course, to consider conditions at tail settings in the vicinity of trim at the particular lift coefficient in question and also the lift-coefficient range over which the nonlinearities extend.

Similar data for a sweptback tailless configuration are shown in figure 3. The data for  $M = 0.70$  and  $0.95$  were obtained from Langley high-speed 7- by 10-foot tunnel tests of a semispan model. The data for  $M = 1.00$  were obtained from wing-flow tests of a smaller model. The increased slope of the pitching-moment curves at the higher Mach numbers is again evident. At  $M = 0.95$  the control effectiveness has been considerably reduced and appreciable trim changes occur, but the vicious changes in stability that are frequently manifested by straight-wing designs at supercritical speeds are absent.

The effect that sweepback can have on delaying the Mach number at which significant trim changes and stability changes are manifested is further illustrated in figure 4. The straight-wing design and the tailless design are the configurations for which typical data have been

presented. (See figs. 2 and 3.) The model with a  $45^\circ$  swept wing and tail was an arbitrary configuration investigated on the transonic bump. In evaluating the control settings required for trim at the various Mach numbers, appropriate flight plans at altitude were assumed for each configuration. It is interesting to note the manner in which the initial trim changes have been postponed to higher Mach numbers for the swept configurations and, in particular, the extremely small trim changes associated with the  $45^\circ$  configuration. Above their respective critical speeds, both the straight-wing design and the tailless configuration manifested irregular trim changes. It is desirable to keep trim changes as small as possible, although the amount of trim change that can safely be tolerated depends to a considerable extent on the type of stability

associated with  $\left(\frac{\partial C_m}{\partial C_L}\right)_M$ . For the straight-wing configuration two boundaries are presented for the parameter  $\left(\frac{\partial C_m}{\partial C_L}\right)_M$  at supercritical speeds. The

lower boundary is associated with the local flat spots in the pitching-moment data previously discussed. (See fig. 2.) These flat spots extended over a lift-coefficient range of less than 0.1 and are relatively unimportant for the particular flight plan employed for this example, inasmuch as the minimum lift coefficient attained is about 0.2. The response of the airplane to disturbances necessary to effect accelerations of the order of 2g or 3g is probably more nearly associated with some value between the two boundaries.

For the  $35^\circ$  swept design, this parameter is more precisely determinable and does not change appreciably up to a Mach number of 0.88, although it also increases rather rapidly at the higher supercritical Mach numbers.

For the  $45^\circ$  swept configuration, changes in the parameter have been delayed until a Mach number of about 0.95 has been reached and then

$-\left(\frac{\partial C_m}{\partial C_L}\right)_M$  increases rather gradually. This comparison illustrates the need for employing a large degree of sweepback if trim and stability changes in the transonic region are to be minimized.

Two factors greatly affecting the value of  $\left(\frac{\partial C_m}{\partial C_L}\right)_M$  are the wing-fuselage-aerodynamic-center position and the downwash at the tail. The manner in which these factors changed with Mach number for the straight-wing design and the  $45^\circ$  swept design are shown in figure 5.

The large variation in the local position of the wing-fuselage-aerodynamic-center position (denoted by  $-\left(\frac{\partial C_m}{\partial C_L}\right)_M$  tail off) for the

straight-wing design is immediately apparent, and this variation is reflected in the behavior of the tail-on results, although the magnitude of the fluctuations has been decreased because of the increased tail effectiveness effected by the reduction in  $\frac{dc}{dC_L}$  at the tail at the supercritical Mach numbers.

For the 45° swept configuration, the wing-fuselage-aerodynamic-center position varied only a small amount, and the increase in  $\left(\frac{\partial c_m}{\partial C_L}\right)_M$  (tail on) at the higher Mach number was largely due to the increased tail effectiveness caused by the reduction in downwash slope at the tail.

### Dynamic Stability

The parameter  $\left(\frac{\partial c_m}{\partial C_L}\right)_M$  also influences to some extent the frequency of the short-period longitudinal oscillation. Some computations for a few characteristic designs were made in order to observe the manner in which this quantity affected the dynamic stability characteristics, and the results of the computations for a tailless design investigated are presented in figure 6. It is immediately apparent that altitude has a pronounced effect on the period of the oscillation and that the period becomes shorter as the speed is increased. The period varies in a somewhat hyperbolic manner with  $\left(\frac{\partial c_m}{\partial C_L}\right)_M$  so that for the values  $-\left(\frac{\partial c_m}{\partial C_L}\right)_M$  less than 0.05 the period will increase very rapidly, whereas for values of  $-\left(\frac{\partial c_m}{\partial C_L}\right)_M$  greater than 0.15 the period will change only slightly. The importance of the frequency of the short-period oscillation will probably have to await flight experience, inasmuch as it will depend to some extent on the damping characteristics. It will be noted that while the damping, as evaluated by the number of seconds to damp to 1/2 amplitude, depends to a considerable extent on altitude and speed, it is independent of the parameter  $\left(\frac{\partial c_m}{\partial C_L}\right)_M$ . It is influenced significantly, however, by the damping in pitch; and for airplanes with a tail, the damping will be more rapid than that indicated here. For a particular design the characteristics of the short-period oscillation can be rapidly evaluated inasmuch as one needs only to determine the roots of the second-degree equation usually associated with this mode of the longitudinal motion.

## LOW-SPEED PROBLEMS

One of the factors that has limited the amount of sweepback that can be beneficially employed on transonic designs has been the difficulty of providing satisfactory stability and control characteristics in the landing condition.

Basic wing characteristics.— At lift coefficients prior to that at which separated flow ensues on the wing, the position of the aerodynamic center of the wing can be estimated fairly reliably. The shift in the aerodynamic-center position that occurs at high lift coefficients is less amenable to theoretical computations, and numerous experimental investigations have been concerned with this effect. From the data examined thus far it appears that aspect ratio and sweep angle are still the two most important factors that influence the type of pitching-moment variation to be expected at the stall. The familiar manner in which sweep angle and aspect ratio affect the character of the pitching-moment variation at the stall is illustrated in figure 7, which is taken from reference 11. Combinations of sweep and aspect ratio that fall above the line on the figure have been found to yield the characteristically unstable pitching-moment variation indicated. Other factors such as airfoil section, wing taper, Reynolds number, and surface roughness have been found to influence the lift coefficient at which instability is first manifested, but the ultimate variation at that stall has still been found to be consistent with that indicated in the figure.

While figure 7 reflects the behavior of plain wings, it has been found that the addition of trailing-edge flaps has resulted in an unstable pitching-moment variation even for wings falling in the stable region in figure 7. A considerable number of investigations have therefore been concerned with the development of devices designed to alleviate the tip stalling that is responsible for this behavior (references 12 to 15).

Stall-control devices.— At the present time stall-control devices have been successfully applied to wings with leading-edge sweep angles up to  $42^\circ$ . Some of the results of an investigation (references 12 and 13) covering the effect of stall-control devices on the pitching-moment characteristics of a  $42^\circ$  sweptback wing equipped with a split flap are shown in figure 8. This wing has an NACA 64<sub>1</sub>-112 section and an aspect ratio of 4. This investigation was conducted in the Langley 19-foot pressure tunnel at a Reynolds number of about 6,840,000. The basic wing-fuselage combination exhibited an unstable pitching-moment variation at the stall. The addition of leading-edge flaps of the type indicated, covering about 60 percent of the span, resulted in a stable break of the

pitching-moment curve at the stall, and this type of leading-edge device was the most satisfactory tested. Similar effects were also obtained with a leading-edge slot arrangement which covered 60 percent of the span except for a small region of instability just before  $C_{L_{max}}$ . This unstable region was removed by the addition of a fence located at the inboard end of the slot. This effect is somewhat typical of fence behavior. If located properly, fences, in general, have been found helpful in minimizing local unstable variations in the pitching-moment curve up to the maximum lift coefficient but do not appreciably affect the ultimate character of the pitching-moment variation at the stall.

Effect of fuselage.— The percent span of leading-edge flap or slot required to effect satisfactory pitching-moment behavior at the stall depends somewhat on the size of the fuselage to which the wing is attached and, to a lesser extent, on the position of the wing on the fuselage. The effect is illustrated in figure 9 (reference 13). The configuration represented by 0.575 leading-edge slots is the same wing configuration discussed in figure 8 and the fuselage is seen to have little effect on the character of pitching-moment variation at the stall. When the leading-edge flap span was increased to  $0.725\frac{b}{2}$ , however, the wing-fuselage combination was unstable at the stall; whereas the wing alone still exhibited favorable characteristics. Similar results were obtained for a high- and low-wing arrangement. It appears from tuft studies of these configurations that the flow over the fuselage delays the stalling of the center section to such an extent that initial separation again began over the flapped portion of the wing.

Effect of tail location.— The addition of a tail adds further complications but, in general, it has been found that stable behavior of the resultant pitching moment at the stall is most likely to be achieved when the basic wing-fuselage pitching moment exhibits a stable variation. The location of the tail, however, is an important consideration and the effect of adding a tail to the wing-fuselage configuration with  $0.575\frac{b}{2}$  leading-edge flaps and  $0.50\frac{b}{2}$  trailing-edge flaps is shown in figure 10.

A study of these data indicates that the most satisfactory pitching-moment behavior at the stall was actually achieved with the low tail position by virtue of the decreased rate of change of downwash associated with this tail location. This low position was close to the edge of the wing wake, however, and may be objectionable from other considerations. The more desirable midtail location possessed a local region of instability just before  $C_{L_{max}}$  which was removed by the addition of a fence.

## CONCLUSIONS

In recapitulation, the following generalizations can be made:

1. The incorporation of large amounts of sweepback on both the wing and the horizontal tail has been found to increase the Mach number at which trim changes and stability changes are first manifested and to reduce greatly the trim changes and stability changes encountered at supercritical speeds.

2. Longitudinal stability in the landing condition has been attained for configurations with sweep angles of the order of  $45^\circ$  utilizing various stall-control devices, but at the present time optimum arrangements for these devices must be determined experimentally.

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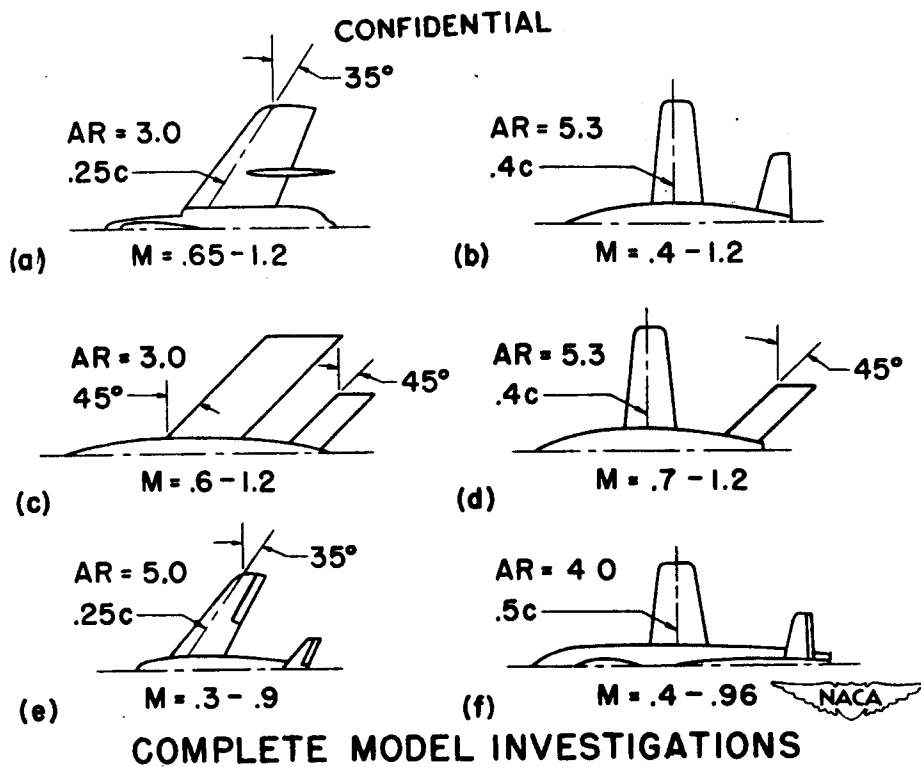


Figure 1.

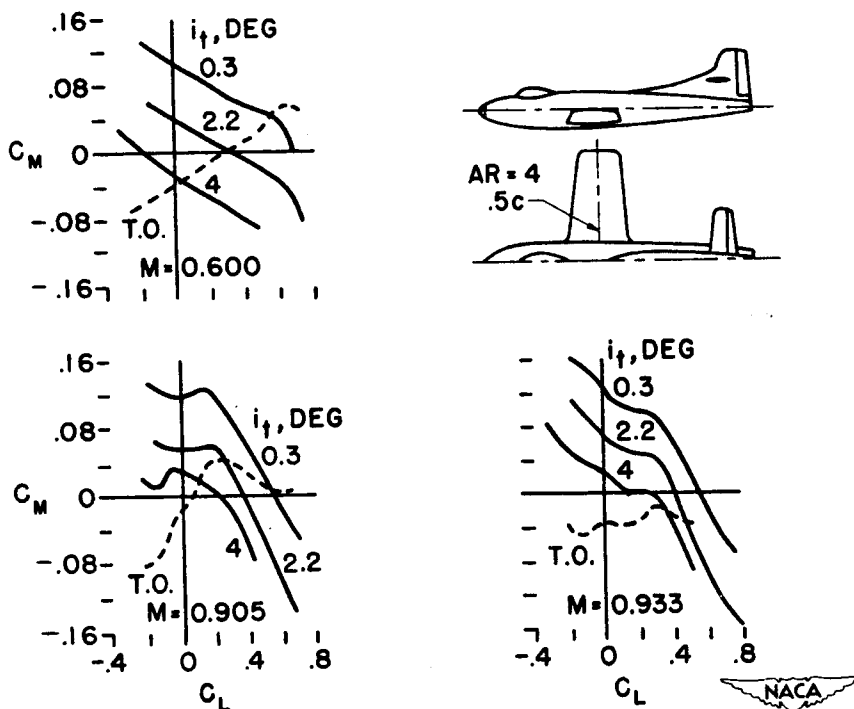
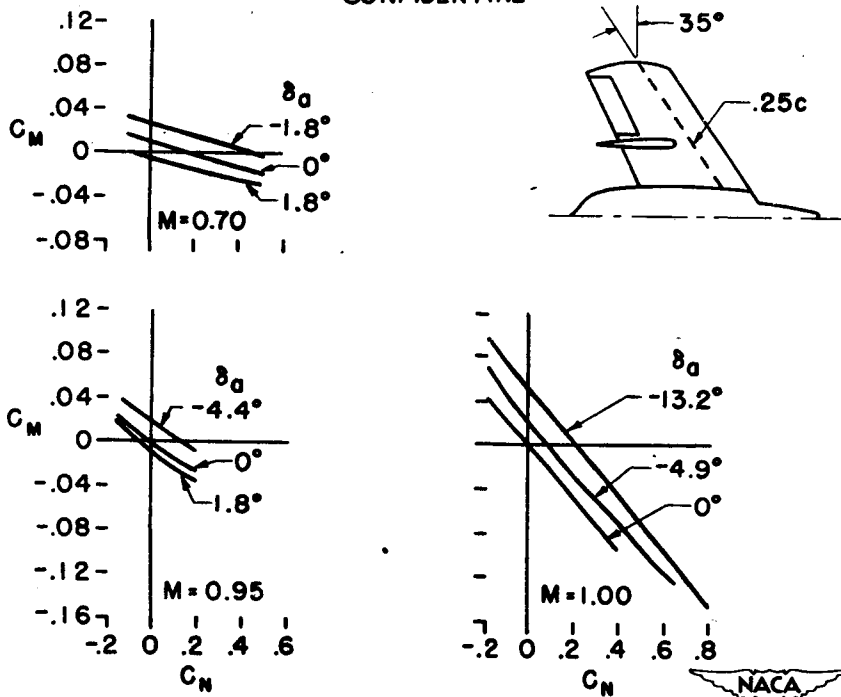


Figure 2.

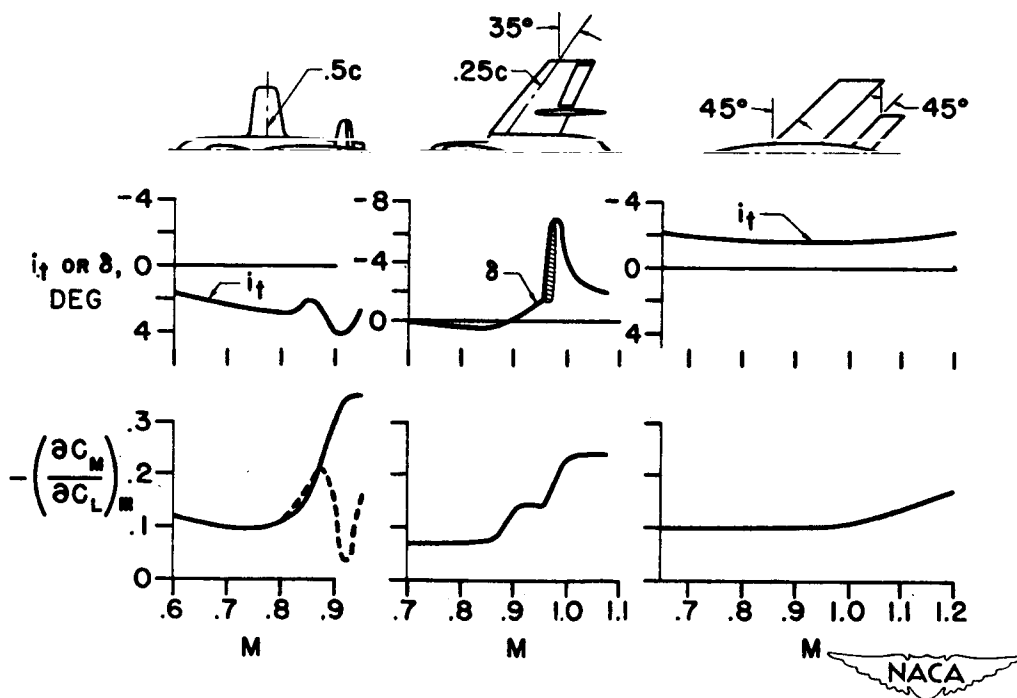
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## PITCHING CHARACTERISTICS OF A TAILLESS DESIGN

Figure 3.



## LONGITUDINAL STABILITY OF THREE TRANSONIC DESIGNS

Figure 4.

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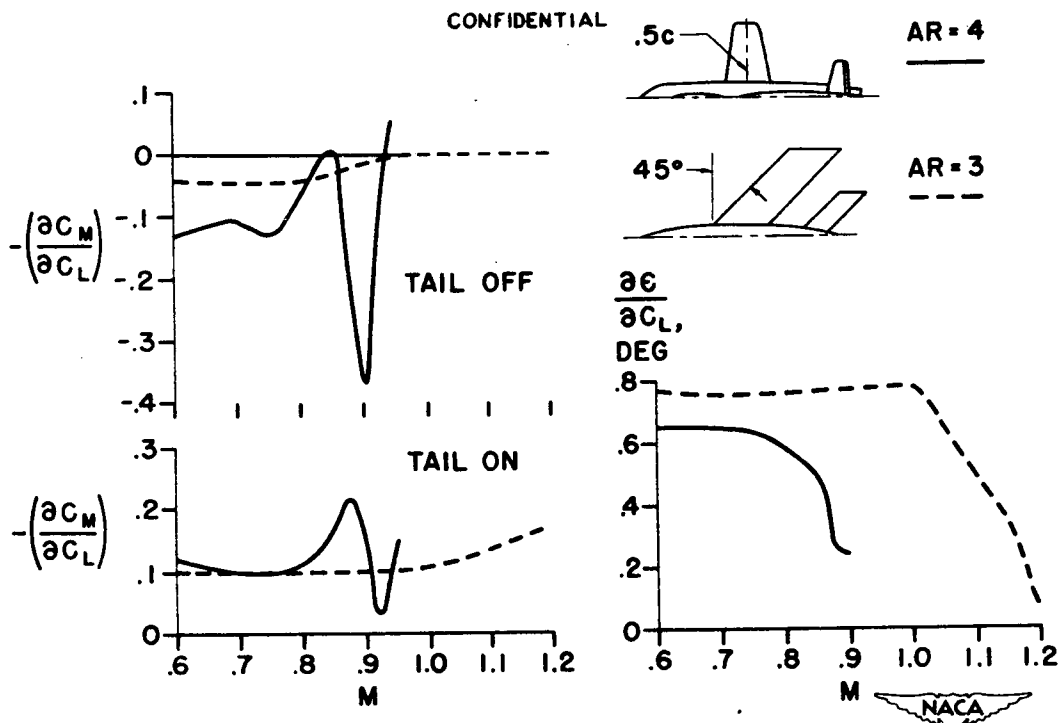


Figure 5.

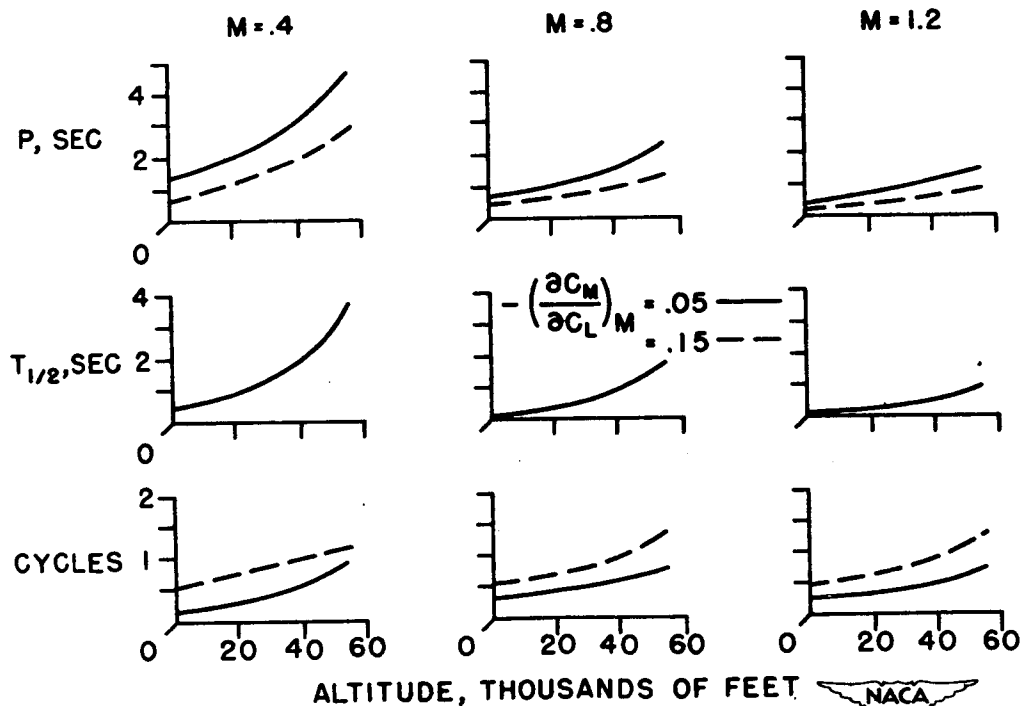


Figure 6.

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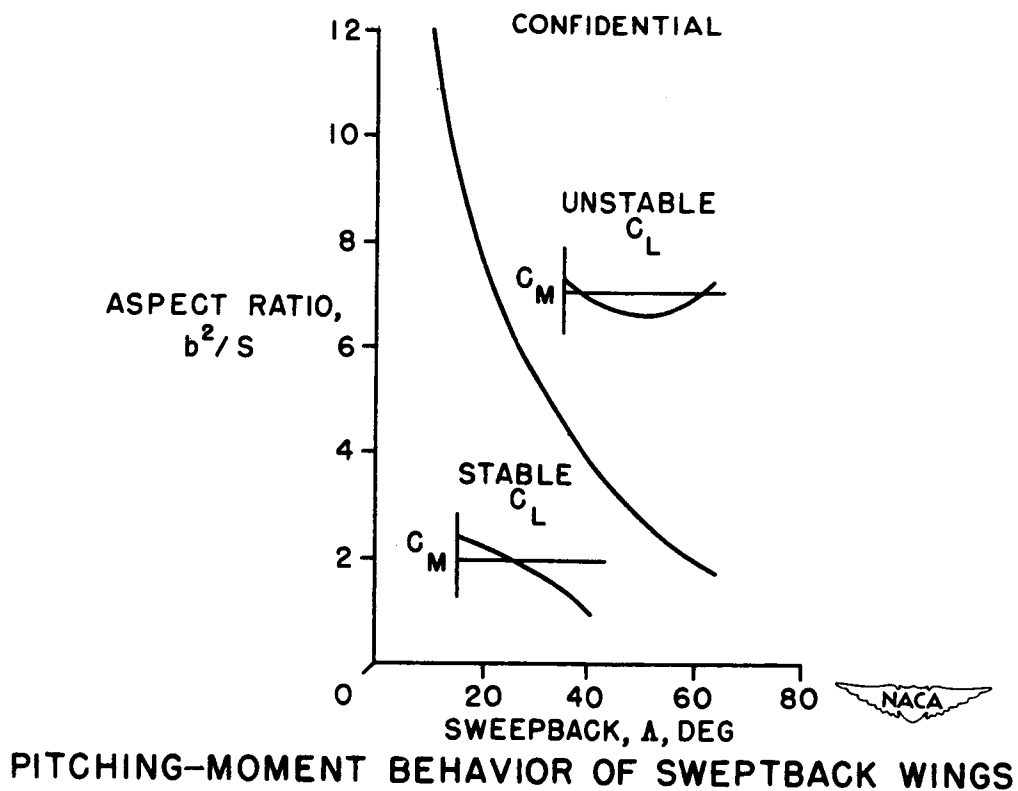
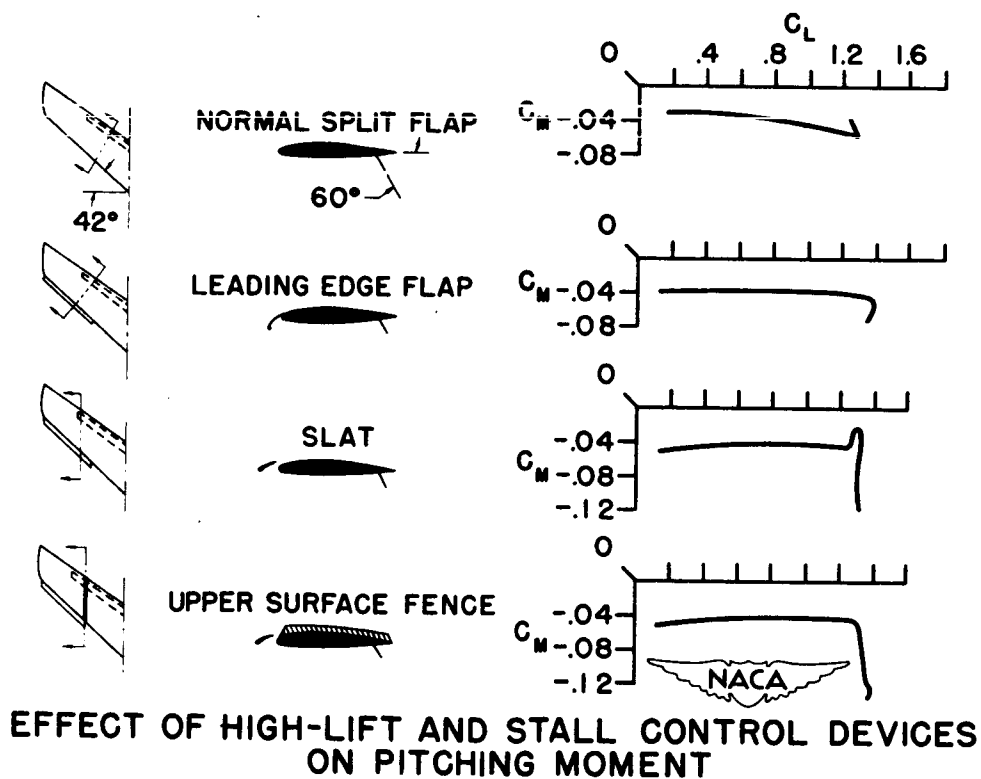
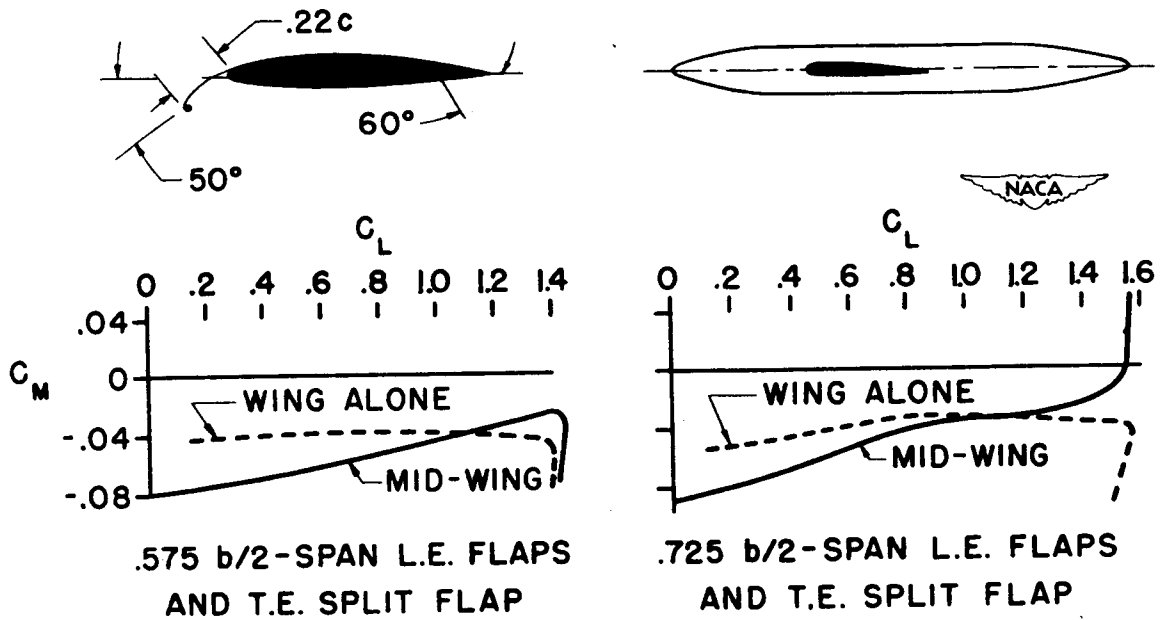


Figure 7.

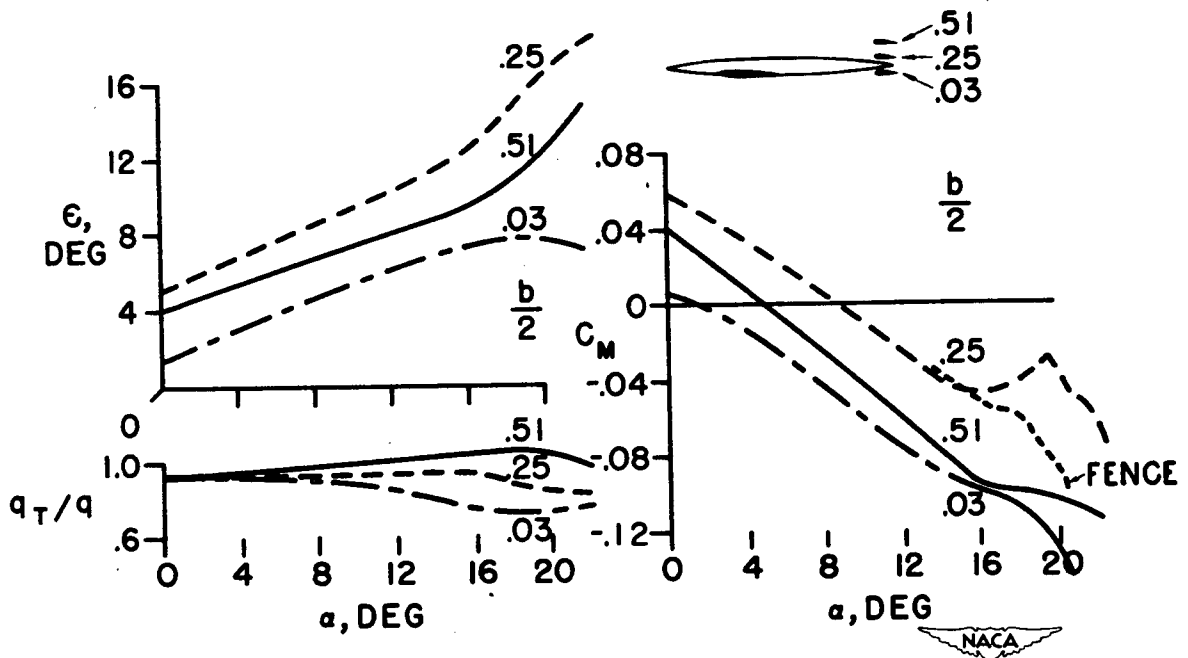
Figure 8.  
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## EFFECT OF FUSELAGE ON PITCHING MOMENT

Figure 9.



## EFFECT OF TAIL LOCATION

Figure 10.

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# FACTORS AFFECTING STATIC LONGITUDINAL STABILITY AND CONTROL

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## INTRODUCTION

The purpose of this paper is to review the various factors that constitute static longitudinal stability and control and to indicate how these factors may be influenced by power effects and Mach number effects.

## SYMBOLS

|                  |                                                                                                                |
|------------------|----------------------------------------------------------------------------------------------------------------|
| $C_m$            | pitching-moment coefficient                                                                                    |
| $C_L$            | lift coefficient                                                                                               |
| $b$              | wing span                                                                                                      |
| $S$              | wing area                                                                                                      |
| $i_t$            | stabilizer incidence, degrees                                                                                  |
| $\epsilon'$      | increment of power-off downwash at horizontal tail (at given angle of attack) from zero-lift downwash, degrees |
| $\Delta\epsilon$ | increment in downwash, at a given angle of attack, due to power, degrees                                       |
| $T_C$            | thrust coefficient $\left( \frac{\text{Thrust}}{\rho V^2 D^2} \right)$                                         |
| $\Delta T_C$     | increment in thrust coefficient from power-off condition to a specified power condition                        |
| $F$              | plan-form factor                                                                                               |
| $M$              | Mach number                                                                                                    |

## BASIC CONCEPTS

Static stability relates to the behavior of an airplane in a series of steady states of motion. It is of interest, therefore, to align the



practical conditions for stability as desired by pilots with the conditions for stability that result from a mathematical treatment of the subject. From the pilot's point of view an airplane possesses stick-position stability if the stick must be moved rearward to retrim the airplane at a speed lower than the initial trimmed speed or moved forward to retrim the airplane at a speed higher than the initial trimmed speed. If the rearward movement of the stick requires a pull force or if the forward movement of the stick requires a push force, the airplane also possesses stick-force stability.

The basic mathematical condition for static stability is that the constant term  $E$  of the quartic stability equation be positive:

$$\lambda^4 + B\lambda^3 + C\lambda^2 + D\lambda + E = 0 \quad (1)$$

The development of equation (1) for the stick-fixed condition may be found in references 1 and 2 and for the stick-free condition in reference 3.

Physically, a positive value of  $E$  indicates that one of the longitudinal modes of motion of the airplane will consist of a long-period oscillation, classically termed a phugoid oscillation. The question of the characteristics of this oscillation, and whether it is stable or unstable, is one of dynamic stability and therefore is not discussed herein. A discussion of its importance from the pilot's point of view may be found in reference 4. It will suffice to say that if  $E$  is positive, the phugoid oscillation will be present in some form but, more important, the previously mentioned relationships concerning stick-position and stick-force stability will be satisfied.

On the other hand, if  $E$  is negative, the long-period phugoid oscillation is replaced essentially by a slow divergence and the pilot will find it necessary to reverse his customary procedure for retrimming the airplane. This reversal of customary flight procedure, while not particularly desirable, is generally not catastrophic because the divergence that develops as a consequence of this type of instability depends on speed changes that take considerable time to develop.

The expression "static stability" has also been used to describe the weathercock tendencies of an airplane while flying at a constant speed. This type of stability is essentially angle-of-attack stability and is extremely important in that it prevents, for example, the airplane from developing excessive load factors when encountering a gust or other disturbances. Static stability is frequently referred to as "maneuvering stability," inasmuch as it also controls the inherent maneuverability of the airplane. The mathematical condition

for this type of stability is that the coefficient  $C$  of equation (1) be positive. Physically, if  $C$  is positive, a short-period oscillation is present; if  $C$  is negative, the oscillation is replaced by a very rapid and dangerous divergence. It should be emphasized that from the point of view of safety it is the type of stability associated with  $C$  that is most important to the pilot.

In simplified treatments of stability where power effects and compressibility effects are ignored, little misunderstanding results from the different interpretations attached to the term "static stability" because the same factor, the slope of the curve of pitching-moment coefficient against lift coefficient, usually determines the sign of both  $E$  and  $C$ . When the effects of power and compressibility are taken into account, however, the terms  $E$  and  $C$  are no longer dependent on the same parameters and a more precise interpretation of their significance is essential.

The four concepts and definitions commonly employed in current discussions of longitudinal stability are summarized in table I. The type of stability associated with  $E$  is manifested as stick-position stability or as stick-force stability. The degree of stability is

measured by the static margin, defined as  $-\left(\frac{dC_m}{dC_L}\right)_{C_m=0}$ . The

parameter  $-\left(\frac{dC_m}{dC_L}\right)_{C_m=0}$  can be evaluated from wind-tunnel tests with

the controls either fixed or free if the tests are conducted to simulate the appropriate power condition and flight plan. The center-of-gravity position for which the static margin vanishes for either the stick-position or stick-force condition defines the neutral point.

The type of stability associated with the term  $C$  is interpreted and measured by the pilot in terms of the control movement or control force required to effect a given acceleration at a constant speed. The degree of stability is proportional to the so-called maneuver margin. The maneuver margin can be evaluated from wind-tunnel tests as the sum of the slope of the pitching-moment curve obtained at a constant Mach number and for a fixed power condition and a term representing the damping-in-pitch characteristics of the airplane. For heavily loaded airplanes flying at high altitudes  $K$  is negligible

and the maneuver margin is given essentially as  $-\left(\frac{\partial C_m}{\partial C_L}\right)_M$  which can

easily be obtained from wind-tunnel tests. The maneuver point coincides with the center-of-gravity position corresponding to zero maneuver margin.

If the manner in which the pitching-moment coefficient varies with the lift coefficient is known, all the essential stability parameters can be evaluated.

### STABILITY AT SUBCRITICAL SPEEDS

The stability of a conventional-type airplane is determined by the relative contributions of the wing-fuselage combination and the horizontal tail. At subcritical speeds the contribution of the basic wing-fuselage combination can be estimated fairly reliably, and numerous papers and charts are available for simplifying such calculations. (See references 5 to 10.) The contribution of the horizontal tail in the absence of slipstream or jet effects can also be estimated fairly reliably for both the flaps-retracted and flaps-deflected conditions (reference 11) with the aid of downwash charts such as those prepared by Silverstein and Katzoff (reference 12). Reliable methods are also available for estimating the hinge-moment characteristics of the elevator; thus, rational estimates of the stick-free stability characteristics can be made. (See reference 13.) The addition of a propeller or a jet may, however, cause important changes in the contributions supplied by the various components, and a knowledge of the manner in which these effects are manifested is extremely helpful in design.

### POWER EFFECTS

Propeller effects.— Successful methods have been developed for estimating the effects of the slipstream on the wing-fuselage characteristics (references 14 to 18), but attempts to predict the complex changes in flow at the tail plane have been less successful.

During the war years a large amount of experimental data pertaining to propeller effects were obtained particularly for single-engine airplanes. Typical investigations are reported in references 19 to 28. These data have been analyzed and a method has been developed for estimating the effects of power on the contribution of the tail to stability. The essence of the method is presented in figures 1 and 2, which were taken from an unpublished analysis.

The data of figure 1 constitute a correlation of the change in downwash angle resulting from an increment in thrust coefficient above the windmilling condition. A correlation study of 26 specific model configurations has indicated that the most powerful factors influencing the incremental downwash angle are the initial downwash angle  $\epsilon'$  and a factor  $F$

dependent on the wing plan form. It has been observed that taper ratio is of particular importance, and the manner in which the plan-form correlation factor  $F$  varies with wing taper ratio is also shown in figure 1. The dashed lines parallel to the design curve represent the order of accuracy which might be expected in applying this chart to a new design.

A correlation chart for estimating the power-on tail effectiveness is shown in figure 2. The dependency of the tail-effectiveness ratio on the relative position of the slipstream and tail position should be noted. The lines parallel to the design curve again indicate the order of accuracy of the correlation. These correlation charts at present are applicable only to single-engine tractor monoplanes with flaps retracted, but it is hoped eventually to obtain similar correlation charts for the flap-down condition.

From correlation charts such as those shown in figures 1 and 2 it is possible to construct curves of the variation of pitching-moment coefficient with lift coefficient for any power condition, and from these curves all of the essential stability parameters can be evaluated. References 29 and 30 contain graphical methods for determining the location of the neutral point.

The scarcity of systematic data on multiengine installations has thus far prevented the development of similar correlations for these configurations.

Jet effects.— The influence of jets on the longitudinal stability is, in general, not as pronounced as propellers. (See reference 31.) Direct jet effects are easily computable and charts are available for estimating the inflow field about a jet; thus, the calculation of downwash changes in the vicinity of the horizontal tail is possible (reference 32).

#### COMPRESSIBILITY EFFECTS

Up to the speed at which the critical Mach number of the wing is exceeded, the effects of compressibility on the stability characteristics of an airplane are relatively small, and rational estimates of these effects can be made utilizing formulas based on linear perturbation theory. The more significant changes in stability occur when the critical speed of the wing is exceeded and shock waves are found which result in large pressure changes over the wing. As a consequence, the lift and the lift-curve slope decrease rapidly, and for cambered sections the angle of attack for zero lift shifts in a positive direction. These changes are generally more pronounced for wings having greater camber.

The aerodynamic center of the wing may shift either in a forward or rearward direction depending upon the thickness and shape of the airfoil section and the wing plan form. The wing-aerodynamic-center shift associated with a particular airplane will also be affected by the fuselage or nacelles.

An example of the manner in which compressibility effects were manifested on a World War II fighter is shown in figure 3. (See reference 33.) The characteristics exhibited at  $M = 0.5$  are typical of the behavior below the force break. As the critical speed of the wing was exceeded the aerodynamic center of the wing-fuselage combination moved forward as evidenced by the increased slope of the tail-off pitching-moment curve at  $M = 0.76$ . Despite the forward movement of the wing aerodynamic center, however, the slope  $(\partial C_m / \partial C_L)_M$  for the tail-on configuration was actually increased. A noticeable change in trim is also evident. Thus, while the maneuver margin is considerably increased the static margin, as a consequence of the trim change, becomes unstable. The cause of this behavior usually is that the airplane will experience a nose-down tendency that is so great that either the elevator is not powerful enough to pull the nose of the airplane up or the control forces become too great for the pilot to handle. This behavior is referred to as the "tucking under" tendency.

If an airplane has an adjustable stabilizer, severe trim changes of this type can be compensated for without much difficulty. If the airplane has a fixed stabilizer, however, another solution to this problem is required. The solution adopted for the airplane having the characteristics shown in figure 3 involved the use of dive-recovery flaps. The essential characteristic of a dive-recovery flap is illustrated in figure 4 (reference 34). The dive flap is located on the under surface of the wing and, when deflected, causes an increase in the local downwash at the tail and a change in the angle of zero lift. The effect is the same as though the stabilizer had been moved nose downward, and a powerful positive pitching moment is created. The optimum flap deflection for a particular configuration, however, must be determined experimentally.

Severe compressibility effects may be delayed to higher Mach numbers by utilizing thinner wing sections and by employing plan forms having low aspect ratios or plan forms incorporating sweepback. (See references 35 and 36.) The incorporation of sweepback is particularly beneficial and can significantly increase the Mach number at which serious longitudinal-stability problems are encountered and might be expected also to reduce the trim changes and stability changes encountered at supercritical speeds.

## LOW-SPEED PROBLEMS OF SWEEPBACK WINGS

One of the factors that limits the amount of sweepback that can be employed, however, is the difficulty of providing satisfactory stability and control in the landing condition.

Basic-wing characteristics.—At lift coefficients prior to that at which separated flow ensues on the wing, the position of the aerodynamic center of the wing can be estimated fairly reliably. The shift in the aerodynamic-center position that occurs at high lift coefficients is less amenable to theoretical computations, and numerous experimental investigations have been concerned with this effect. From the data examined thus far it appears that aspect ratio and sweep angle are still the two most important factors that influence the type of pitching-moment variation to be expected at the stall. The familiar manner in which sweep angle and aspect ratio affect the character of the pitching-moment variation at the stall is illustrated in figure 5, which is taken from reference 37. Combinations of sweep and aspect ratio that fall above the line in the figure have been found to yield the characteristically unstable pitching-moment variation indicated. Other factors such as airfoil section, wing taper, Reynolds number, and surface roughness have been found to influence the lift coefficient at which instability is first manifested, but the ultimate variation at the stall has still been found to be consistent with that indicated in figure 5.

While figure 5 reflects the behavior of plain wings, it has been found that the addition of trailing-edge flaps has resulted in an unstable pitching-moment variation even for wings falling in the stable region. A considerable number of investigations have, therefore, been concerned with the development of devices designed to alleviate the tip stalling that is responsible for this behavior.

Stall control devices.—Methods that have been tried in attempts to alleviate the tip stalling of sweptback wings have included wing twist, changes in wing plan form at the tip, flat-plate separators — which attempt to control the lateral flow of the boundary layer — and leading-edge flaps and slats. Combinations of these methods have also been tried on specific configurations. Perhaps the most successful stall control device thus far investigated has been the leading-edge slat. Figure 6 illustrates the behavior of this device on a moderately swept wing. (See reference 38.) It will be noticed that in this example the slat had to be extended over approximately 50 percent of the wing semispan before the desired stable pitching moment at the stall was attained. Inasmuch as the leading-edge slat modifies the span loading along the wing it might be expected that the optimum extent of flap for a particular configuration would depend on the wing plan form and the location of the wing on the fuselage.

Effect of tail location.— The attainment of satisfactory pitching-moment characteristics for the wing-fuselage combination does not guarantee that the configuration with a horizontal tail added will also be satisfactory. The optimum location of the tail must usually be found experimentally. Figure 7 which is taken from reference 39 illustrates a case where the basic wing-fuselage pitching-moment behavior was satisfactory but the resultant pitching-moment behavior with the tail in position was unsatisfactory. It is generally easier, however, to find a tail location that will result in satisfactory stability for the complete configuration if the basic wing-fuselage combination also possesses a stable pitching-moment variation at the stall.

#### CONCLUDING REMARKS

It must constantly be borne in mind that even if ample rigid-model wind-tunnel data are available on which to base predictions of stability, the effects of aeroelastic distortion may result in the airplane having completely different characteristics from those estimated. Some of the effects of elevator-fabric distortion are indicated, for example, in reference 40. At the same time the basic concepts of stability discussed still apply and if wind-tunnel data on an aeroelastically similar model were available reliable stability estimates could be made. There is, however, a great deal of research necessary before satisfactory methods of predicting aeroelastic effects can be developed.

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| TABLE I                                        |                                                                                                  |
|------------------------------------------------|--------------------------------------------------------------------------------------------------|
| TYPE                                           | CRITERION                                                                                        |
| STICK-<br>POSITION<br>STABILITY                | STATIC MARGIN $= - \left( \frac{dC_m}{dC_L} \right)_{C_m=0}$ WITH ELEV. FIXED                    |
| STICK-<br>FORCE<br>STABILITY                   | STATIC MARGIN $= - \left( \frac{dC_m}{dC_L} \right)_{C_m=0}$ WITH ELEV. FREE                     |
| STICK-<br>POSITION<br>MANEUVERING<br>STABILITY | MANEUVER MARGIN $= - \left( \frac{\partial C_m}{\partial C_L} \right)_M + K$<br>WITH ELEV. FIXED |
| STICK-<br>FORCE<br>MANEUVERING<br>STABILITY    | MANEUVER MARGIN $= - \left( \frac{\partial C_m}{\partial C_L} \right)_M + K$<br>WITH ELEV. FREE  |

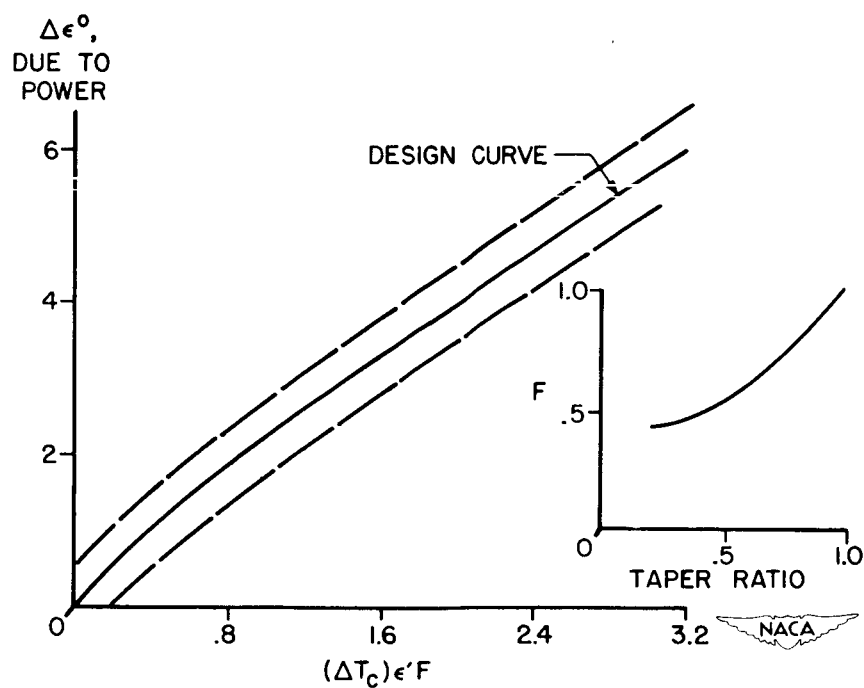


Figure 1.- Downwash correlation for single-engine tractor airplanes.

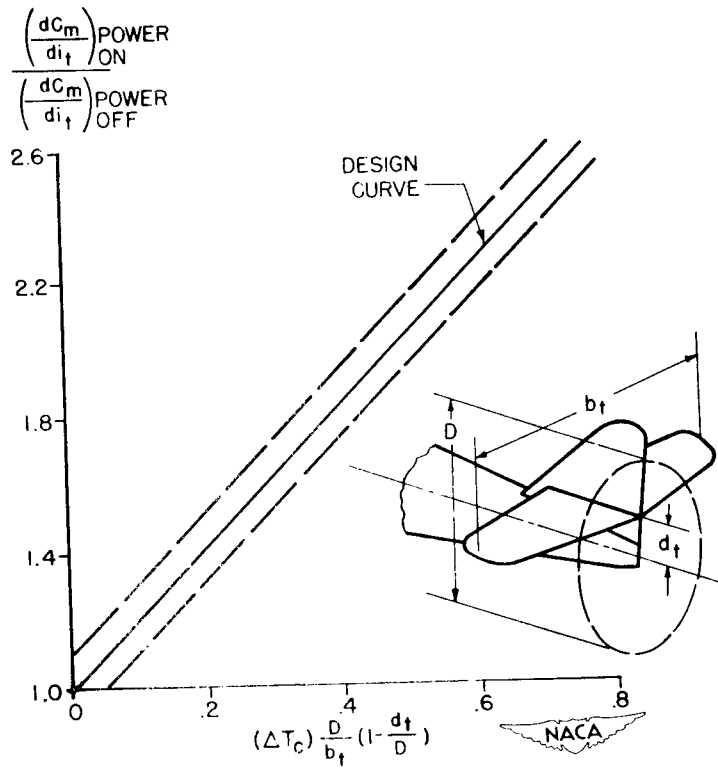


Figure 2.- Tail-effectiveness correlation for single-engine tractor airplanes.

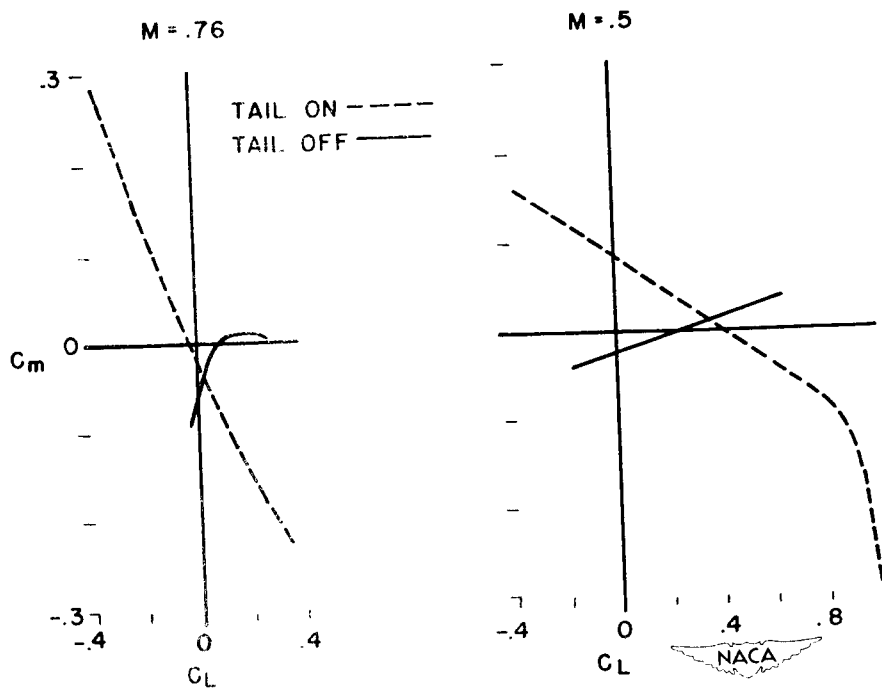


Figure 3.- Typical effect of compressibility on airplane pitching-moment characteristics.

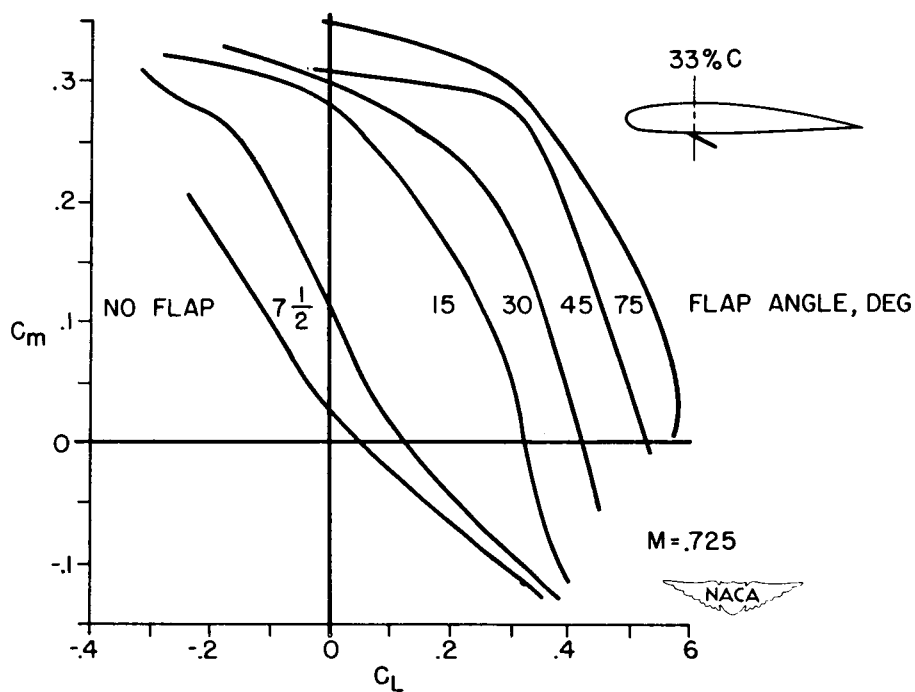


Figure 4.- Effect of dive-recovery flap on airplane pitching-moment characteristics.

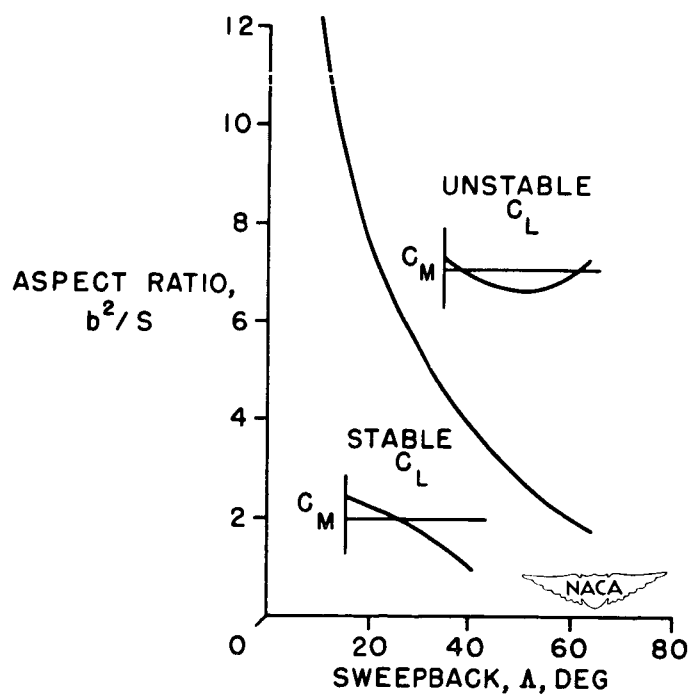


Figure 5.- Pitching-moment behavior of sweptback wings.

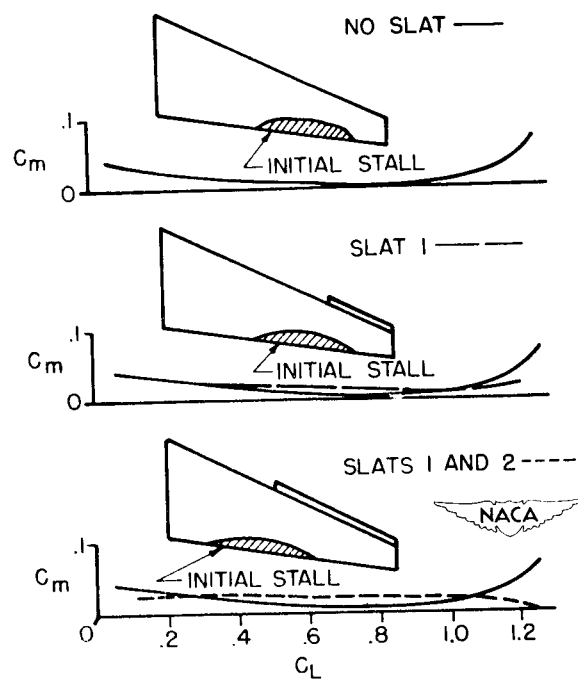


Figure 6.- Effect of leading-edge slats on pitching-moment characteristics.

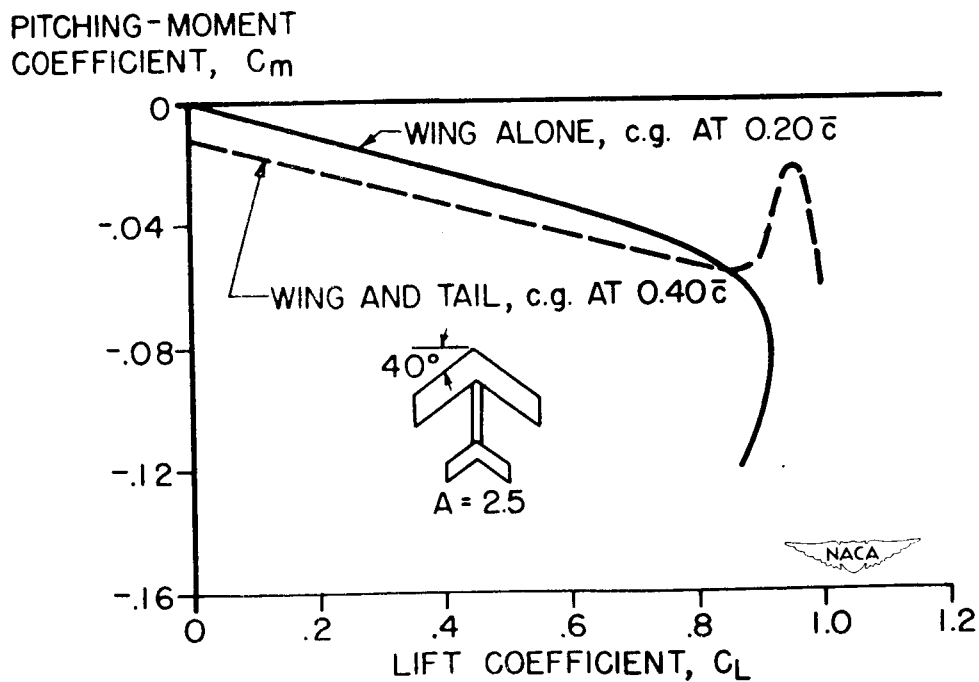


Figure 7.- Effect of tail on pitching-moment characteristics at stall.

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# RESEARCH MEMORANDUM

LOW-SPEED WIND-TUNNEL INVESTIGATION OF THE LONGITUDINAL  
STABILITY CHARACTERISTICS OF A MODEL EQUIPPED  
WITH A VARIABLE-SWEEP WING

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LOW-SPEED WIND-TUNNEL INVESTIGATION OF THE LONGITUDINAL  
STABILITY CHARACTERISTICS OF A MODEL EQUIPPED  
WITH A VARIABLE-SWEEP WING

By Charles J. Donlan and William C. Sleeman, Jr.

## SUMMARY

An investigation has been made to determine the longitudinal stability characteristics of a complete model equipped with a variable-sweep wing at angles of sweepback of  $45^\circ$ ,  $30^\circ$ ,  $15^\circ$ , and  $0^\circ$ . The investigation was directed toward the study of various wing modifications and an external-flap arrangement designed to minimize the shift in neutral point accompanying the change in sweep angle.

The results indicated that stability at the stall was obtained at a sweep angle of  $15^\circ$  without recourse to stall-control devices. The basic neutral-point movement accompanying the change in sweep angle from  $45^\circ$  to  $15^\circ$  amounted to 56 percent of the mean aerodynamic chord (at zero sweep angle) and the most effective modification investigated only reduced this change to 47 percent of the chord. It appears, therefore, that for designs in which the fuselage is the major load-carrying element some relative movement between the wing and center of gravity will be required to assure satisfactory stability at all sweep angles.

## INTRODUCTION

The use of swept wings on high-speed airplanes has introduced serious longitudinal- and lateral-stability problems at low speeds. Many high-lift and stall-control devices have been investigated in an attempt to improve the low-speed characteristics of highly swept wings but no completely satisfactory solution has been found. One obvious method for avoiding the low-speed problems associated with highly swept wings would be to employ a wing whose sweep angle could be varied in flight. Thus, for maximum high-speed flight and optimum cruising performance, the wing could be adjusted to any desired sweep angle; whereas, for the landing condition, the sweep angle could be decreased to an angle that would assure satisfactory low-speed characteristics without recourse to stall-control devices.

The present paper presents the results of a wind-tunnel investigation of a complete model equipped with a wing whose sweep angle could be varied for the purpose of studying various wing modifications designed to decrease

the large forward movement of the neutral point that was found in reference 1 to accompany the decrease in sweepback. Much of the basic data is presented in reference 1 but in that paper the pitching-moment coefficients are based on the mean aerodynamic chord associated with each sweep angle and the assumed pitching-moment reference axis was at 25 percent of the mean aerodynamic chord for each sweep angle. In the present paper a common chord and common reference axis were used in computing the pitching-moment coefficients for all sweep angles investigated.

### COEFFICIENTS AND SYMBOLS

The results of the tests are presented as standard NACA coefficients of forces and moments. Pitching-moment coefficients are given about the center-of-gravity location shown in figure 1 (25 percent of the mean aerodynamic chord at  $0^\circ$  sweep). The data are referred to the stability axes; the positive directions and angular displacements are shown in figure 2.

The coefficients and symbols are defined as follows:

|           |                                                                       |
|-----------|-----------------------------------------------------------------------|
| $C_L$     | lift coefficient ( $Lift/qS$ )                                        |
| $C_{L_0}$ | tail-off lift coefficient                                             |
| $C_X$     | longitudinal-force coefficient ( $X/qS$ )                             |
| $C_m$     | pitching-moment coefficient ( $M/qSc'$ )                              |
| $q$       | free-stream dynamic pressure, pounds per square foot ( $\rho V^2/2$ ) |
| $S$       | wing area without cutout, square feet (varies with angle of sweep)    |
| $S_t$     | horizontal-tail area, square feet                                     |
| $c$       | airfoil section chord, feet                                           |
| $c'$      | wing mean aerodynamic chord (1.181 ft for $\Lambda = 0^\circ$ )       |
| $c_t$     | wing tip chord measured parallel to plane of symmetry, feet           |
| $c_r$     | wing root chord at plane of symmetry, feet                            |
| $b$       | wing span, feet                                                       |
| $V$       | air velocity, feet per second                                         |

|            |                                                                                                   |
|------------|---------------------------------------------------------------------------------------------------|
| $\rho$     | mass density of air, slugs per cubic foot                                                         |
| $\alpha$   | angle of attack of fuselage center line, degrees                                                  |
| $\Lambda$  | angle of sweepback of quarter-chord line of wing, degrees                                         |
| $\epsilon$ | downwash angle, degrees                                                                           |
| $i_t$      | angle of stabilizer with respect to fuselage center line, degrees                                 |
| $\lambda$  | wing taper ratio ( $c_t/c_r$ )                                                                    |
| $\delta_f$ | flap deflection, degrees                                                                          |
| $n_o$      | tail-off aerodynamic-center location, percent wing mean aerodynamic chord for $\Lambda = 0^\circ$ |
| $n_p$      | neutral-point location, percent wing mean aerodynamic chord for $\Lambda = 0^\circ$               |

### MODEL AND APPARATUS

The model used in this investigation (fig. 1) had wing panels which could be rotated about a point on the quarter-chord line to angles of sweepback of  $45^\circ$ ,  $30^\circ$ ,  $15^\circ$ , and  $0^\circ$ . At  $45^\circ$  sweep the wing tips were parallel to the plane of symmetry. The geometric characteristics of the model are tabulated in table I. The model is shown mounted on a single-support strut in the Langley 300 MPH 7- by 10-foot tunnel in figure 3. Details of the various wing modifications investigated are given in figure 4 and table I. Photographs of the model with the external air-foil flaps installed ( $\delta_f = 40^\circ$ ) are presented in figure 5.

Structural limitations of the model wing determined the maximum chordwise dimensions of the cutout which was completely enclosed within the fuselage at  $45^\circ$  sweepback.

### TESTS

#### Test Conditions

The tests were made at a dynamic pressure of 30 pounds per square foot, which corresponds to an airspeed of about 108 miles per hour. The test Reynolds number was approximately  $1.2 \times 10^6$  based on the wing chord of 1.181 feet ( $c'$  at  $\Lambda = 0^\circ$ ). The degree of turbulence of the

tunnel is not known, but is believed to be small because of the large contraction ratio (14 to 1).

The aerodynamic coefficients for all configurations were based on the wing area without cutout. All pitching-moment data were based on a chord of 1.181 feet ( $c'$  at  $\Lambda = 0^\circ$ ).

### Corrections

The data have not been corrected for tares caused by the model-support system inasmuch as, with the arrangement used, the tares are believed to be small. Jet-boundary corrections have been applied to the angles of attack, the drag coefficients, and the tail-on pitching-moment coefficients. The corrections were computed by use of reference 2, which unpublished calculations have indicated to be satisfactory for sweep angles up to  $45^\circ$ .

All forces and moments were corrected for blocking by the method given in reference 3. An increment of longitudinal-force coefficient has been applied to account for the horizontal buoyancy.

### RESULTS AND DISCUSSION

The basic aerodynamic characteristics of the model are presented in figures 6 to 9. The longitudinal-stability parameters are presented in figures 10 to 15. For convenient reference, an outline of the summary figures presenting the results is given as follows:

#### Figure

#### Variable sweep:

- (a) Effect of sweep . . . . . 10
- (b) Effect of faired cutout . . . . . 11, 12, and 13
- (c) Effect of vertical location of  
horizontal tail . . . . . 12(a), 12(b), 12(c), 12(d), and 14

#### Modifications to model with $15^\circ$ sweep:

- (a) Basic tail position
  - 1. Effect of cutout profile and size . . . . . 12(a)
  - 2. Effect of flap deflection . . . . . 12(c)
  - 3. Effect of sharp leading edge . . . . . 12(e)
  - 4. Effect of wing vane . . . . . 12(f)
- (b) Alternate tail position
  - 1. Effect of faired cutout . . . . . 12(c), 12(d)
  - 2. Effect of flap deflection . . . . . 12(d)

#### Comparison of basic configuration with most favorable

- modifications . . . . . 15

## Basic Configurations

Effect of sweep angle, wing without cutouts.— The destabilizing movement of the neutral-point location as the wing sweep angle was decreased amounted to about 2 percent of the chord ( $c'$  at  $\Lambda = 0^\circ$ ) per degree change in sweep angle (fig. 10). Inasmuch as the parameters affecting the tail contribution to stability show relatively minor variations with sweep angle, it would appear that the shift in neutral point is primarily associated with the geometric movement of the wing-aerodynamic-center position as the wing is rotated. Up to about  $35^\circ$  of sweep angle the experimental rate of variation of neutral point with sweep angle is in good agreement with that estimated from the simple geometric consideration that the centroid of lift on each wing panel acts at 25 percent of the mean aerodynamic chord of the wing at each sweep angle.

For the  $30^\circ$  and  $45^\circ$  sweep configurations, an unstable variation of pitching-moment coefficient with lift coefficient at the high lift coefficients is indicated (figs. 8 and 9); whereas, for  $0^\circ$  and  $15^\circ$  configurations, stable pitching-moment characteristics were obtained in the vicinity of the maximum lift coefficient.

Effect of sweep angle, wing with faired cutout.— If a cutout is allowed to develop at the juncture of the wing-root trailing edge and the fuselage as the wing sweep angle is decreased (figs. 1 and 4), significant changes in the stability characteristics exhibited by the model can occur. It was anticipated that the cutout would move the wing-aerodynamic-center position forward somewhat but that the downwash field in the vicinity of the horizontal tail would be changed in such a manner that the over-all stability of the model would be increased. The extent to which these effects were manifested at various sweep angles is indicated in figures 11 to 13.

A study of these data indicates that the faired cutout afforded somewhat greater stability than the configurations having no cutout but the over-all effect on stability is small compared to the large changes produced by the geometric movement of the wing aerodynamic center. The effects on the stability parameters were greatest at  $0^\circ$  sweep and decreased as the sweep angle was increased. The tail contribution to stability at sweep angles of  $0^\circ$  and  $15^\circ$  was increased considerably because of favorable flow changes at the tail but this beneficial effect was partially canceled by the forward movement of the wing aerodynamic center caused by the cutouts.

The effect of the faired cutout on the neutral-point position for all sweep angles is summarized in figure 15.

Effect of vertical location of the horizontal tail.— Decreasing the height of the horizontal tail above the wing from the basic position to the alternate position (fig. 1) showed little effect on  $n_p$  for the configurations investigated (figs. 12(a), 12(b), and 14). The stability was slightly lower at the higher lift coefficients with the tail in the alternate position owing mainly to a less favorable downwash gradient.

#### Modifications to Model with $15^\circ$ Sweepback

Inasmuch as the configuration with  $15^\circ$  sweepback possessed favorable stability characteristics at the stall, this configuration was adopted as the basic low-speed arrangement and various modifications were investigated in an attempt to reduce the large ( $0.56c'$ ) basic shift in neutral point accompanying the reduction in sweep from  $45^\circ$  to  $15^\circ$ .

Effect of cutout profile.— The effects of various cutout arrangements are presented in figure 12(a). From stability considerations, the faired cutout appeared to be superior to the unfaired cutouts and this arrangement was used for the majority of tests with cutouts.

Effect of external airfoil flaps.— In an effort to compensate for the forward movement of the wing aerodynamic center caused by the cutout and at the same time introduce a field of upwash in the vicinity of the cutout, tests were made of a configuration employing essentially full-span external flaps (figs. 4 and 5). The results obtained for the various arrangements tested are presented in figures 7(c), 7(d), 12(c), and 12(d). A comparison of these results with those for the configuration without flaps (figs. 7(a), 7(b), 12(a), and 12(b)) indicate that the flap caused an additional rearward movement of the neutral point of only about  $0.02c'$  ( $\Lambda = 0^\circ$ ). The flap arrangement which was used in this investigation was not a particularly effective one, however, as is indicated by the rather low lift and pitching-moment increments produced by the flap. (Compare figs. 7(a) with 7(c).) It is possible that, with a well-designed extensible-slotted-flap arrangement, the rearward neutral-point movement resulting from the deflected flap for the configuration with the wing cutout would be considerably increased.

Sharp leading edge and wing vane.— Several sharp-leading-edge sections and wing vanes mounted on the inboard sections of the wing panel were investigated in an attempt to reduce the lift on this portion of the wing and thereby increase the stability by reducing the downwash gradient. None of these modifications changed the stability characteristics appreciably. Typical results obtained with the sharp leading edge and wing vane are presented in figures 12(e) and 12(f).

### Design Considerations

The variation of the static longitudinal stability characteristics with sweepback and model configuration is summarized in figure 15. It is evident that a combination of cutouts and flaps can aid in minimizing the forward neutral-point movement as the sweep angle is decreased to  $15^\circ$  but that translation of the wing is required to compensate for the greater portion of the neutral-point movement. For the sweep range investigated it would be necessary to translate the wing rearward roughly 0.5c' ( $\Lambda = 0^\circ$ ) as the sweep angle is decreased to  $15^\circ$  in order to maintain a constant location of the neutral point. It is probable that a sweep angle greater than  $15^\circ$  (but less than  $30^\circ$ ) would be satisfactory from low-speed stability considerations. If, for a particular design, the extent of longitudinal wing translation is limited, the maximum sweep angle for which adequate low-speed stability characteristics are attainable should be determined from wind-tunnel experiments.

Although the incorporation of a wing capable of translation as well as rotation affords formidable structural problems, the potential aerodynamic rewards incident to their solution are significant. The ability to adjust the sweep angle in flight not only makes it possible to utilize the most efficient sweep angles for high speed and cruising performance but assures stability in the landing configuration without recourse to wing slots or other stall-control devices. The more efficient moderately swept higher-aspect-ratio wing used for the landing condition can also be equipped with conventional high-lift devices and thus provide minimum landing speeds. The wing sweep angle could be adjusted in flight for optimum cruising configuration, and for the highest sweep angles the wing can be translated to compensate partly for the stability changes usually encountered at the higher Mach numbers with swept wings.

It appears from low-speed stability data that the cutout formed at the wing-fuselage juncture as the wing is rotated forward is beneficial. If high speeds are contemplated with intermediate sweep angles, however, model tests at higher Mach numbers will be required to evaluate the effect of these cutouts.

The amount of wing translation required is also dependent on the mass distribution of the airplane and the location of the wing pivot point. The weight of the wings, the location of wing fuel tanks as well as fuel tanks in other parts of the airplane, and the plan for emptying the fuel tanks in flight must be considered in evaluating the stability of the airplane. In the case of a variable-sweepback flying wing, for example, the center of gravity would move almost as much as the aerodynamic center of the wing. A study of the unlimited configurations that could utilize center-of-gravity movements created by expendable fuel and moving structural elements is, however, beyond the scope of this paper.

## CONCLUSIONS

The results of a low-speed wind-tunnel investigation of a complete model having a variable-sweep wing which was tested at  $45^\circ$ ,  $30^\circ$ ,  $15^\circ$ , and  $0^\circ$  sweepback indicated the following conclusions:

1. Stability at the stall was obtained for the configuration with  $15^\circ$  sweepback without recourse to stall-control devices.

2. The shift in neutral point as the sweep was varied from  $45^\circ$  to  $15^\circ$  was decreased from 56 percent of the chord ( $c'$  at  $\Lambda = 0^\circ$ ) in the original case to 47 percent by the most effective combination of the modifications tested.

3. It seems unlikely that satisfactory stability for all flight conditions can be achieved with a variable-sweep wing without recourse to relative translation between the wing and the center of gravity of the airplane.

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National Advisory Committee for Aeronautics  
Langley Air Force Base, Va.

## REFERENCES

1. Spearman, M. Leroy, and Comisarow, Paul: An Investigation of the Low-Speed Static Stability Characteristics of Complete Models Having Sweptback and Sweptforward Wings. NACA RM No. L8H31, 1948.
2. Gillis, Clarence L., Polhamus, Edward C., and Gray, Joseph L., Jr.: Charts for Determining Jet-Boundary Corrections for Complete Models in 7- by 10-Foot Closed Rectangular Wind Tunnels. NACA ARR No. L5G31, 1945.
3. Thom, A.: Blockage Corrections in a Closed High-Speed Tunnel. R. & M. No. 2033, British A.R.C., 1943.



TABLE I

## PHYSICAL CHARACTERISTICS OF THE VARIABLE-SWEEP MODEL

Center of gravity, all sweep angles, percent chord ( $c'$  at  $\Lambda = 0^\circ$ ) . 25

## Wing:

Root and tip sections . . . . . NACA 65<sub>1</sub>-110( $a = 1.0$ )

Incidence (root chord to center line of fuselage), degrees . . . . 0

| Sweep,<br>$\Lambda$<br>(deg) | Area<br>(sq ft) | Span<br>(ft) | c<br>(ft)     | Aspect<br>ratio | Taper<br>ratio | Cutout area<br>(sq ft) |       |
|------------------------------|-----------------|--------------|---------------|-----------------|----------------|------------------------|-------|
|                              |                 |              |               |                 |                | Faired                 | Large |
| 0                            | 8.67            | 8.35         | 1.181( $c'$ ) | 8.04            | 0.50           | 0.24                   | 0.37  |
| 15                           | 8.40            | 7.75         | 1.181         | 7.15            | .49            | .16                    | .25   |
| 30                           | 8.13            | 6.76         | 1.181         | 5.62            | .48            | .06                    | .12   |
| 45                           | 7.80            | 5.37         | 1.181         | 3.69            | .46            | ----                   | ----  |

## Horizontal tail:

Airfoil section . . . . . NACA 65-008

Total area, sq ft . . . . . 1.625

Span, ft . . . . . 2.85

Aspect ratio . . . . . 5.0

Taper ratio . . . . . 0.50

## External airfoil flap:

Airfoil section . . . . . 23012

Span, percent wing span at  $\Lambda = 15^\circ$  . . . . . 64.5

Chord, percent  $c'$  ( $\Lambda = 0^\circ$ ) . . . . . 14.1





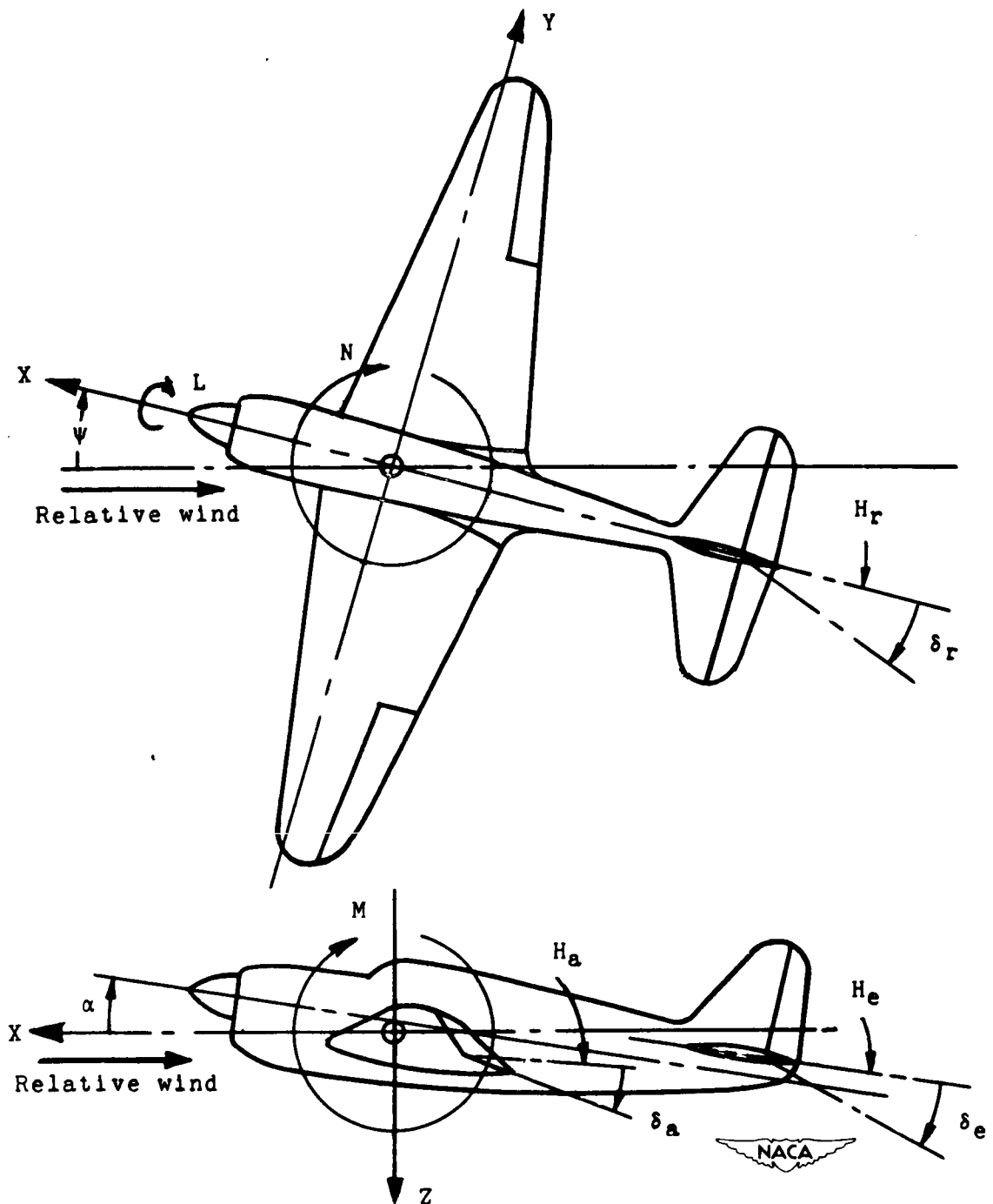


Figure 2.— System of axes and control-surface hinge moments and deflections. Positive values of forces, moments, and angles are indicated by arrows. Positive values of tab hinge moments and deflections are in the same directions as the positive values for the control surfaces to which the tabs are attached.



Figure 3.— Variable-sweep model mounted on single-support strut in 300 MPH 7- by 10-foot tunnel.  
 $\Lambda = 45^{\circ}$ ; rear view.

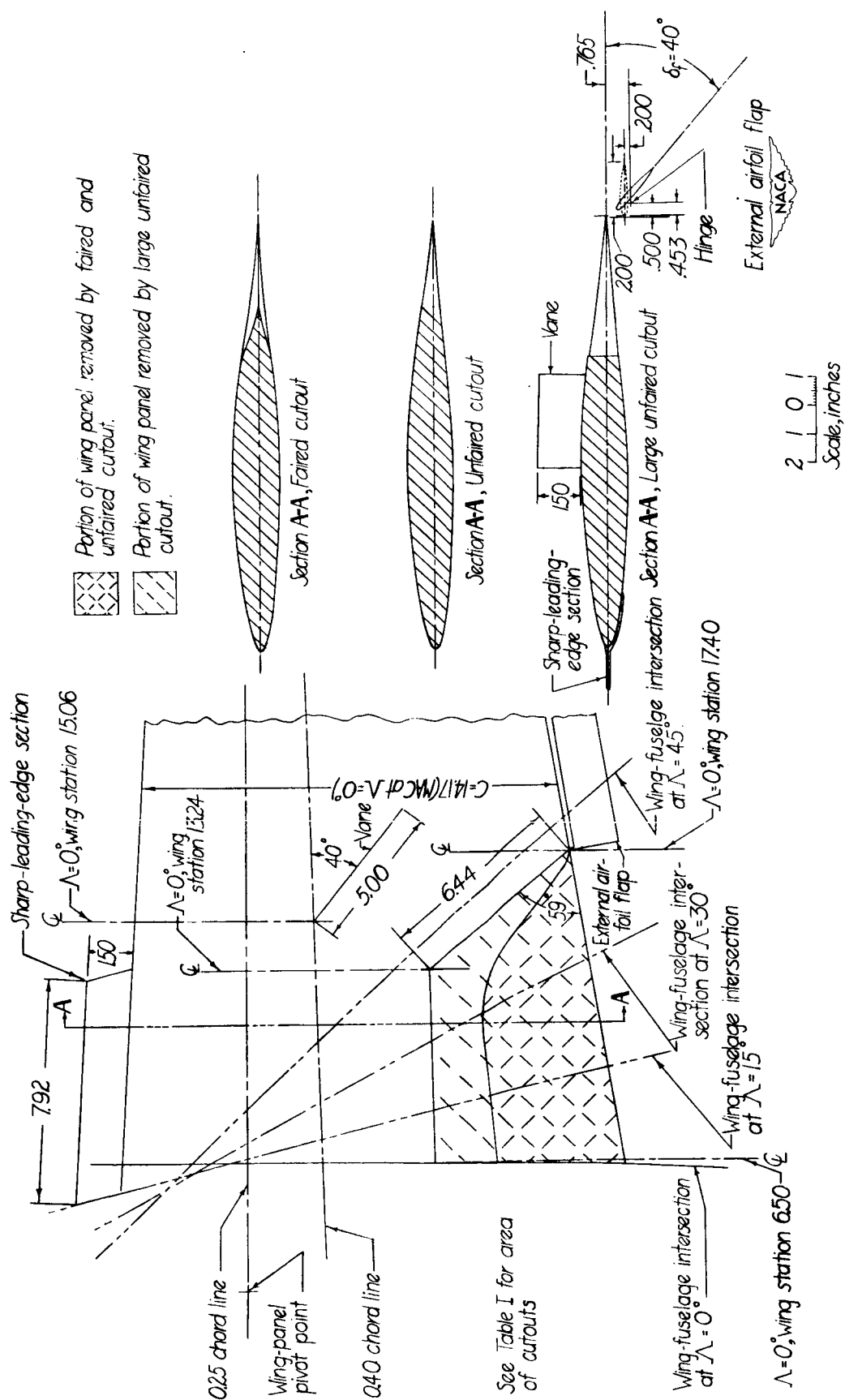
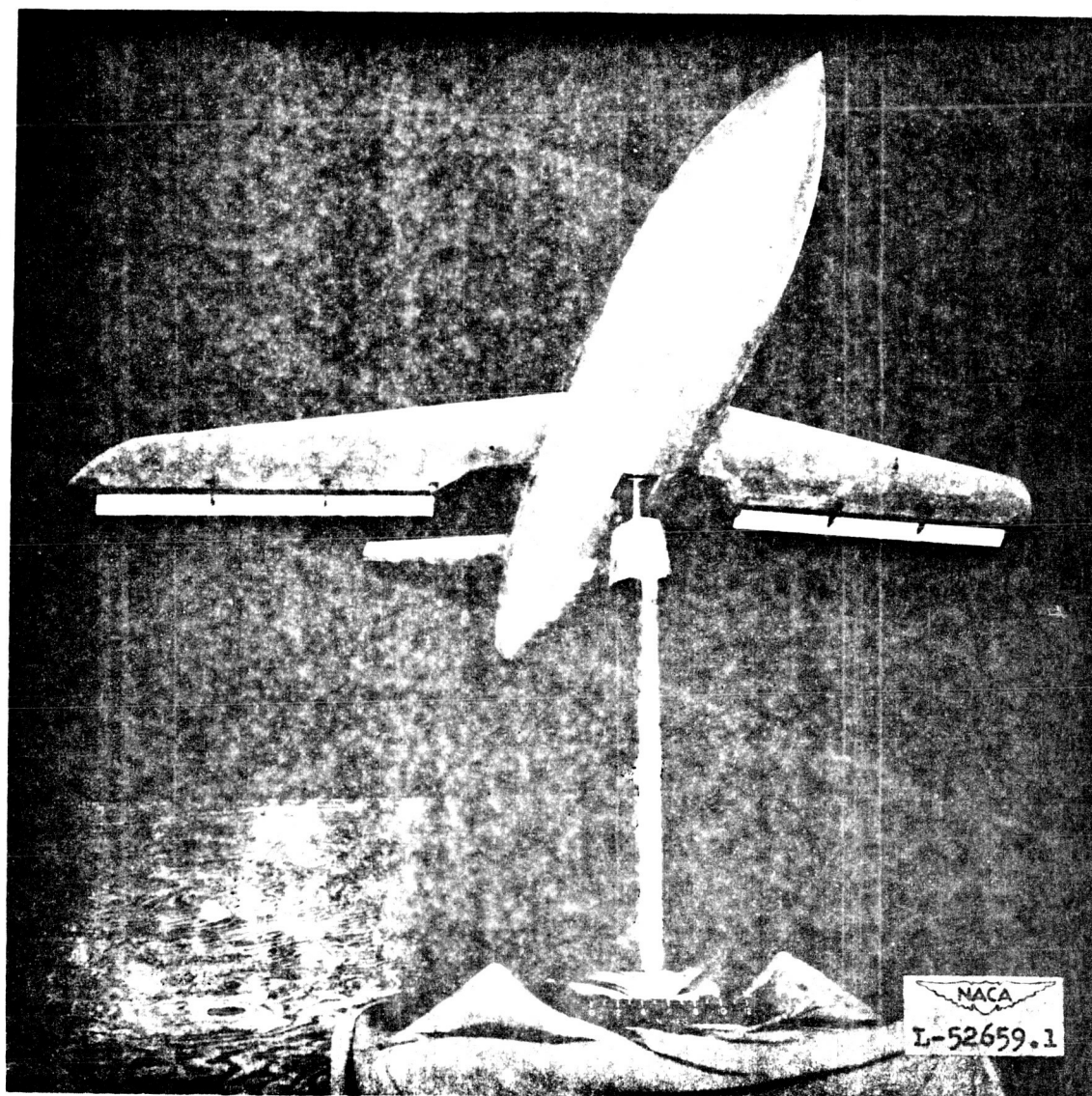
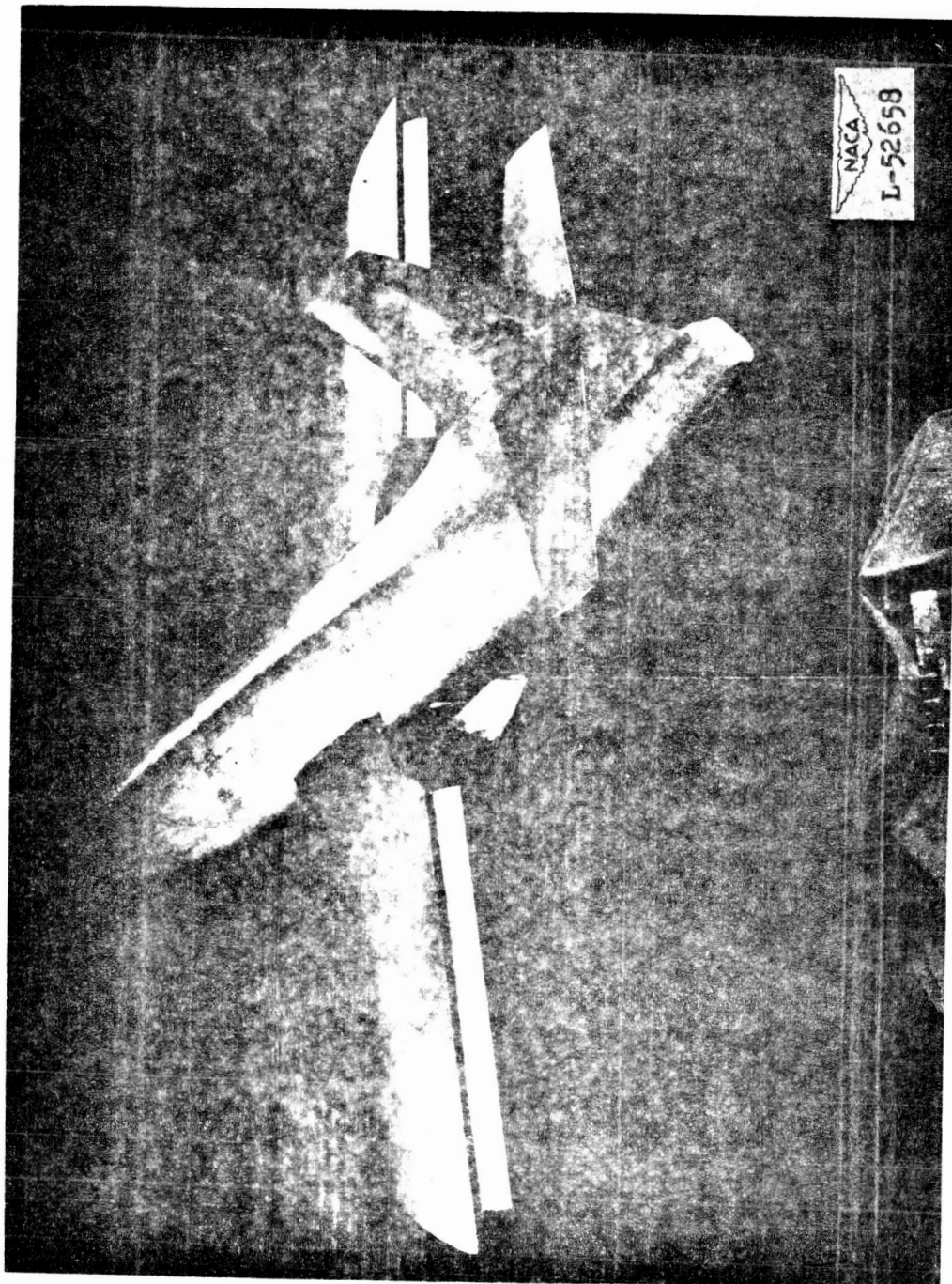


Figure 4.— Details of wing modifications.



(a) Front view.

Figure 5.- Variable-sweep model mounted on single-support strut with faired cutout and external airfoil flaps.  $\Lambda = 15^\circ$ .



(b) Rear view.

Figure 5.- Concluded.

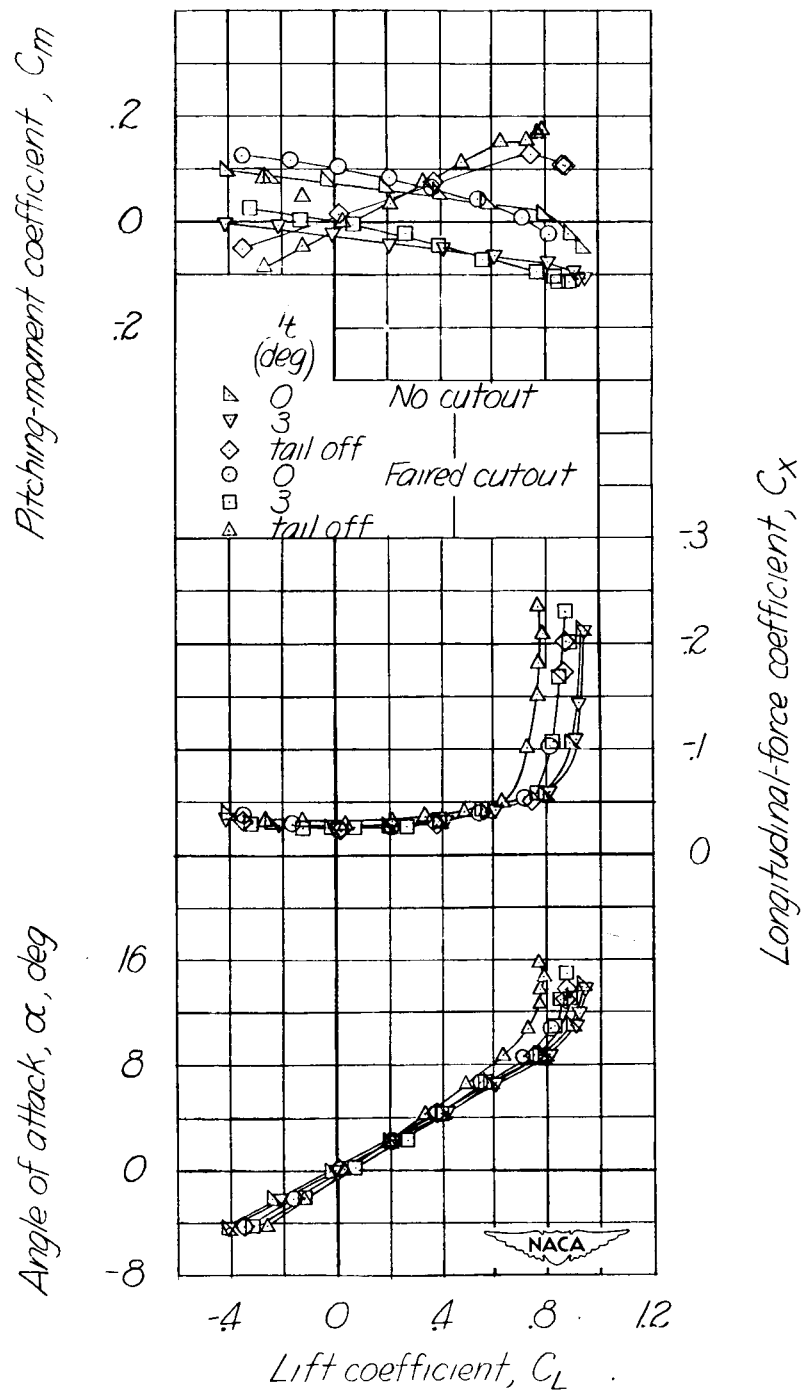
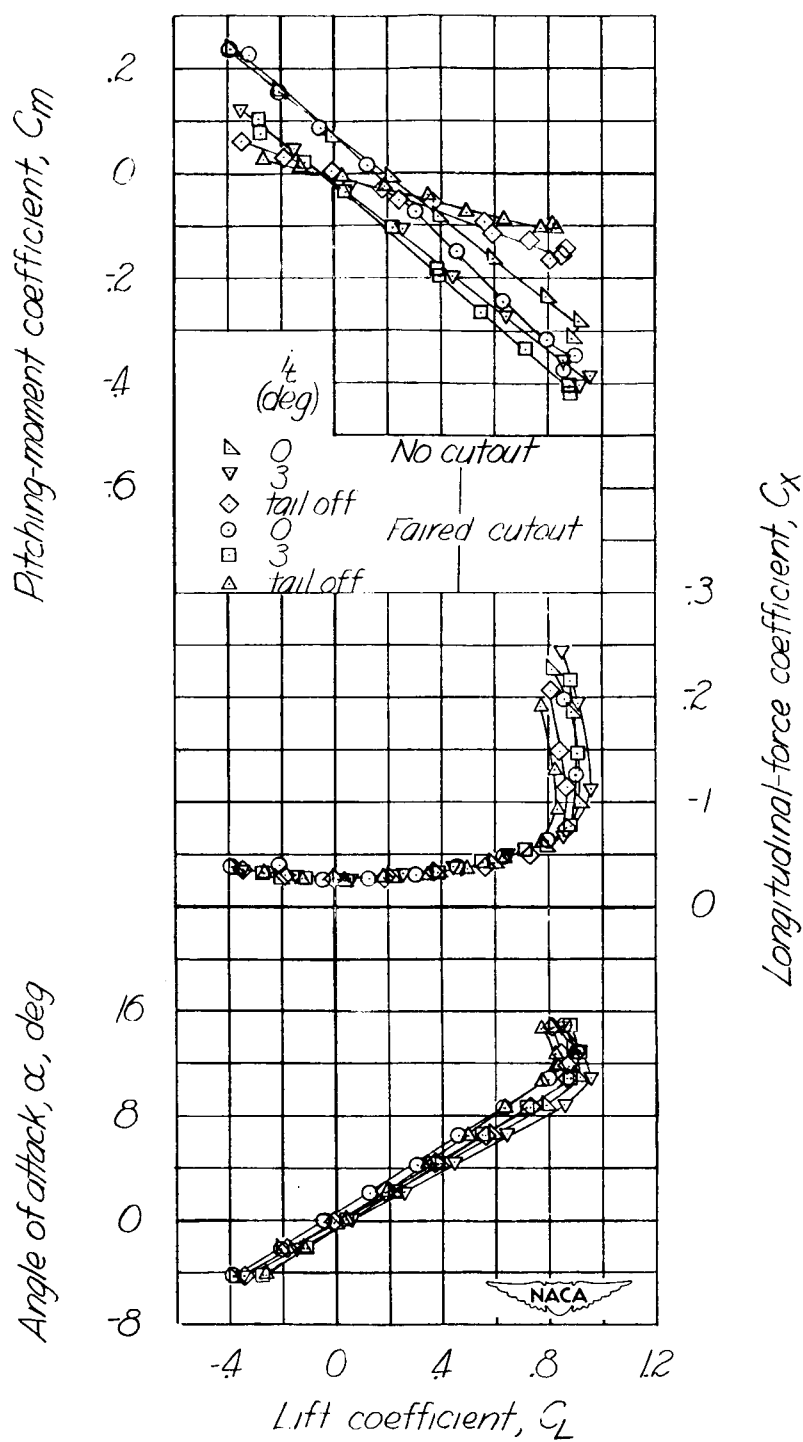


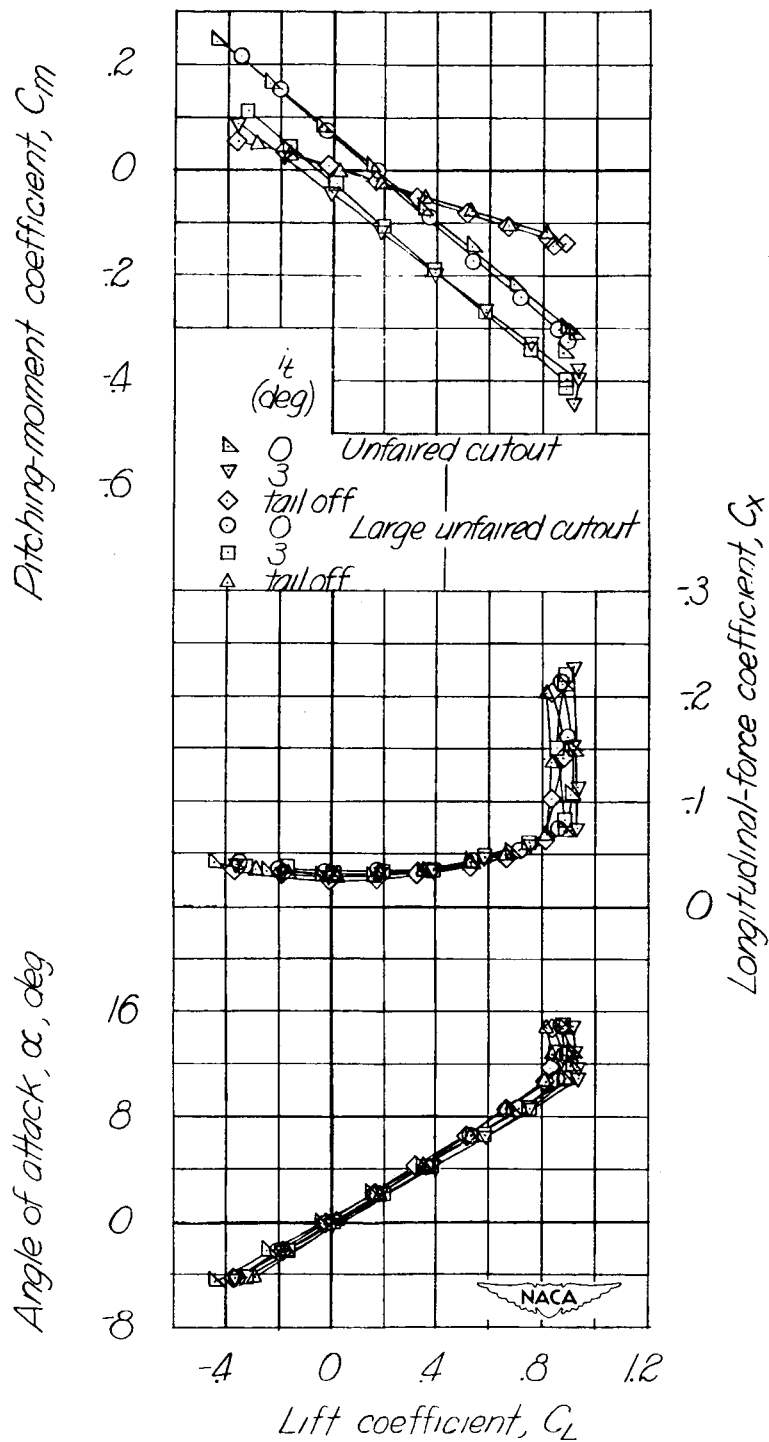
Figure 6.— Aerodynamic characteristics of a variable-sweep model with and without faired wing cutout.  $\Lambda = 0^\circ$ .





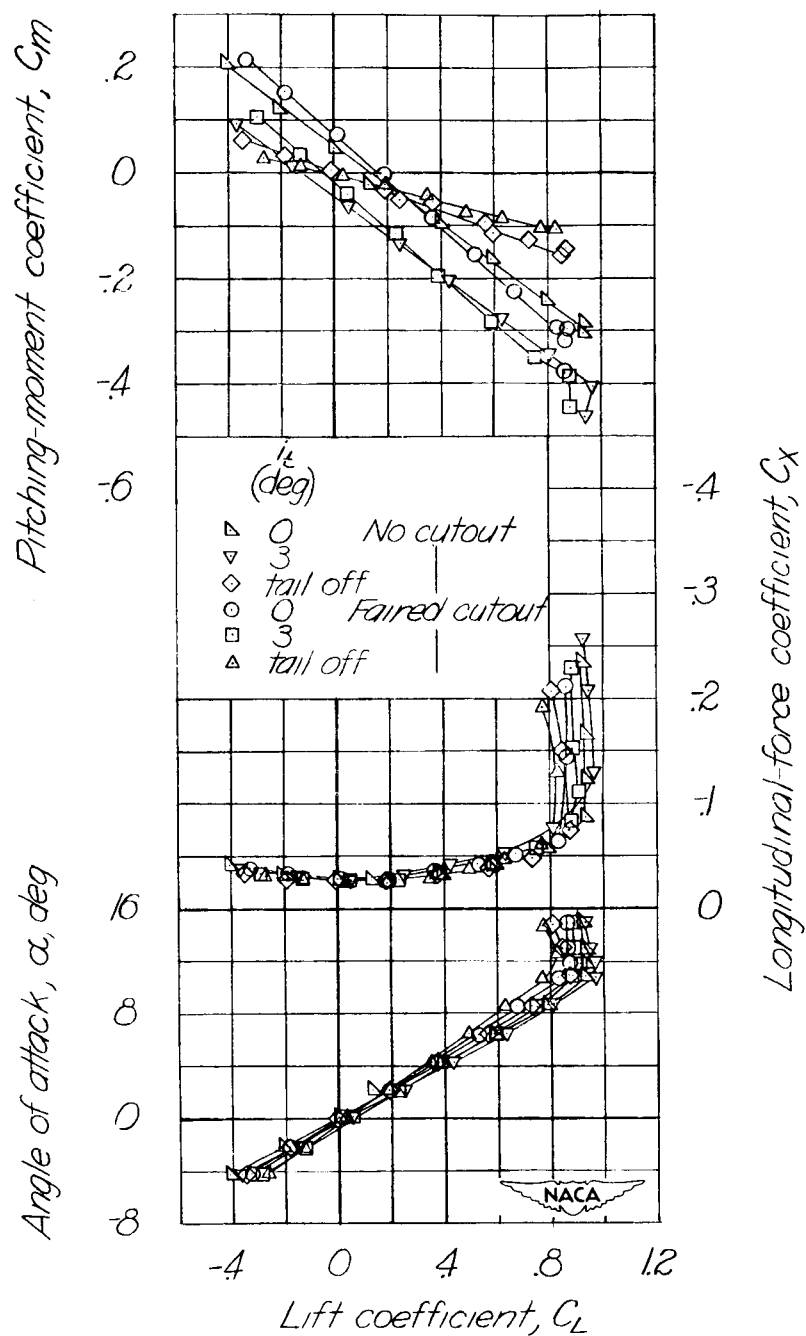
(a) Basic tail position.

Figure 7.- Aerodynamic characteristics of a variable-sweep model with and without wing cutouts.  $\Lambda = 15^\circ$ .



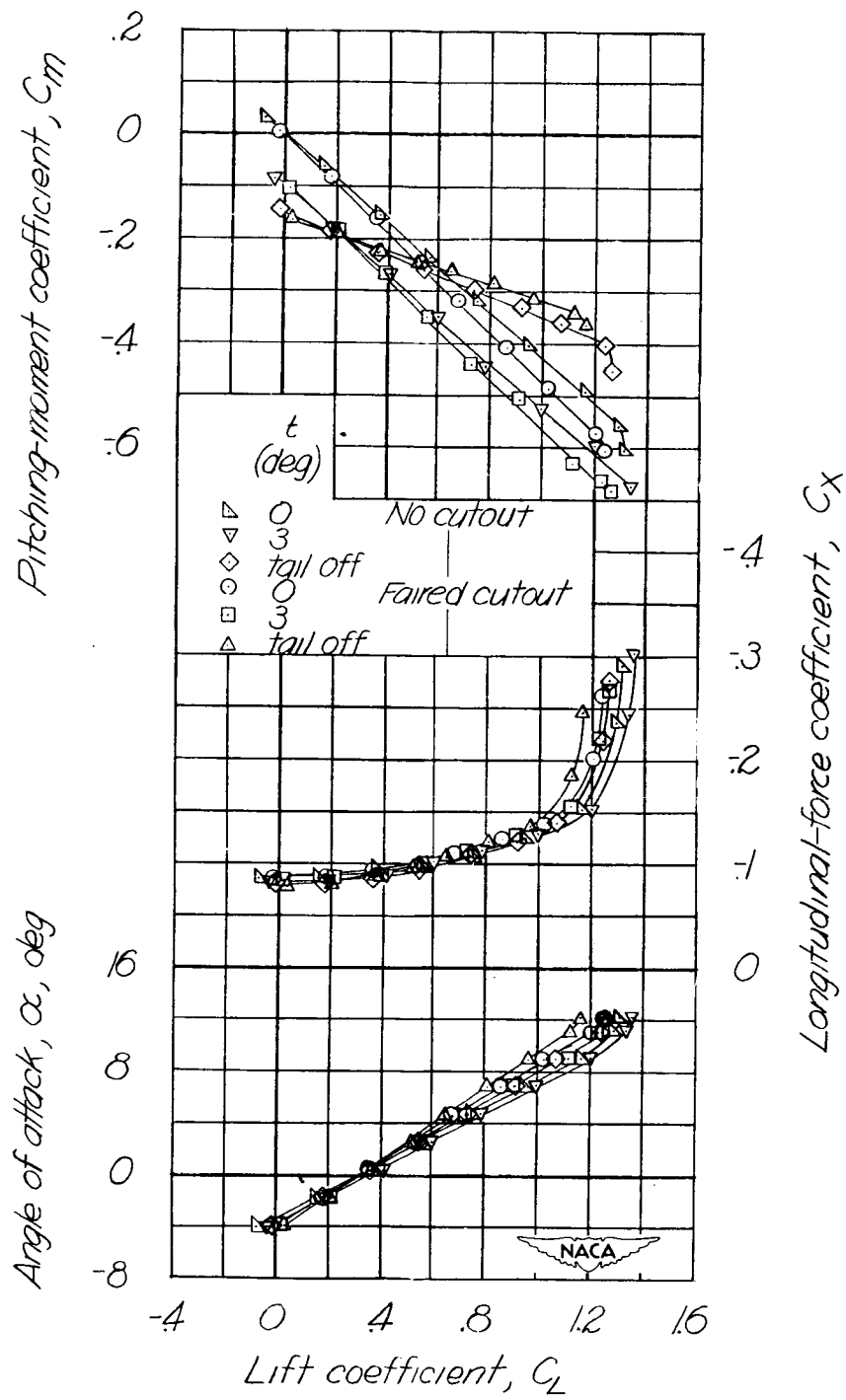
(a) Concluded.

Figure 7.- Continued.



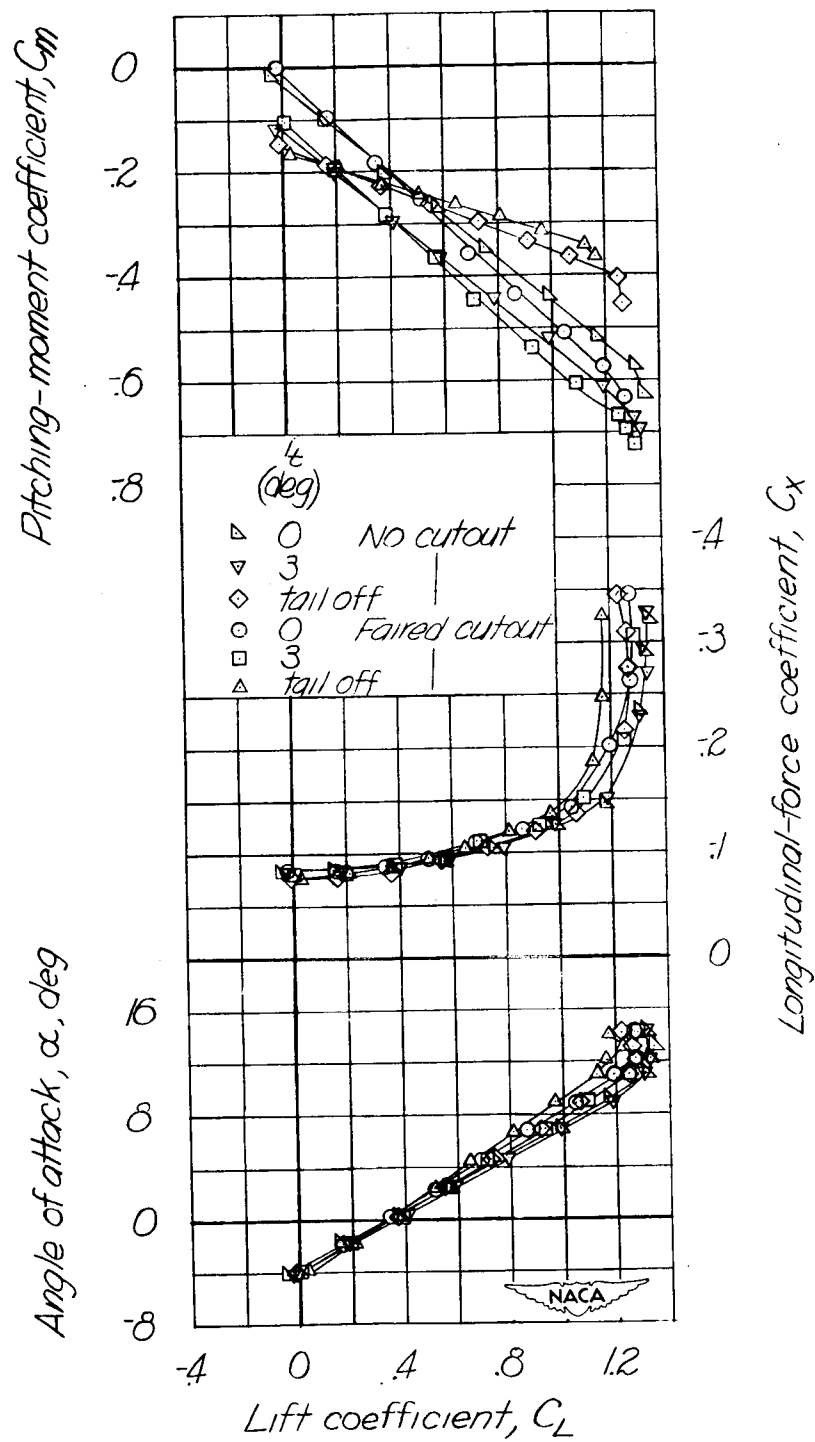
(h) Alternate tail position.

Figure 7.- Continued.



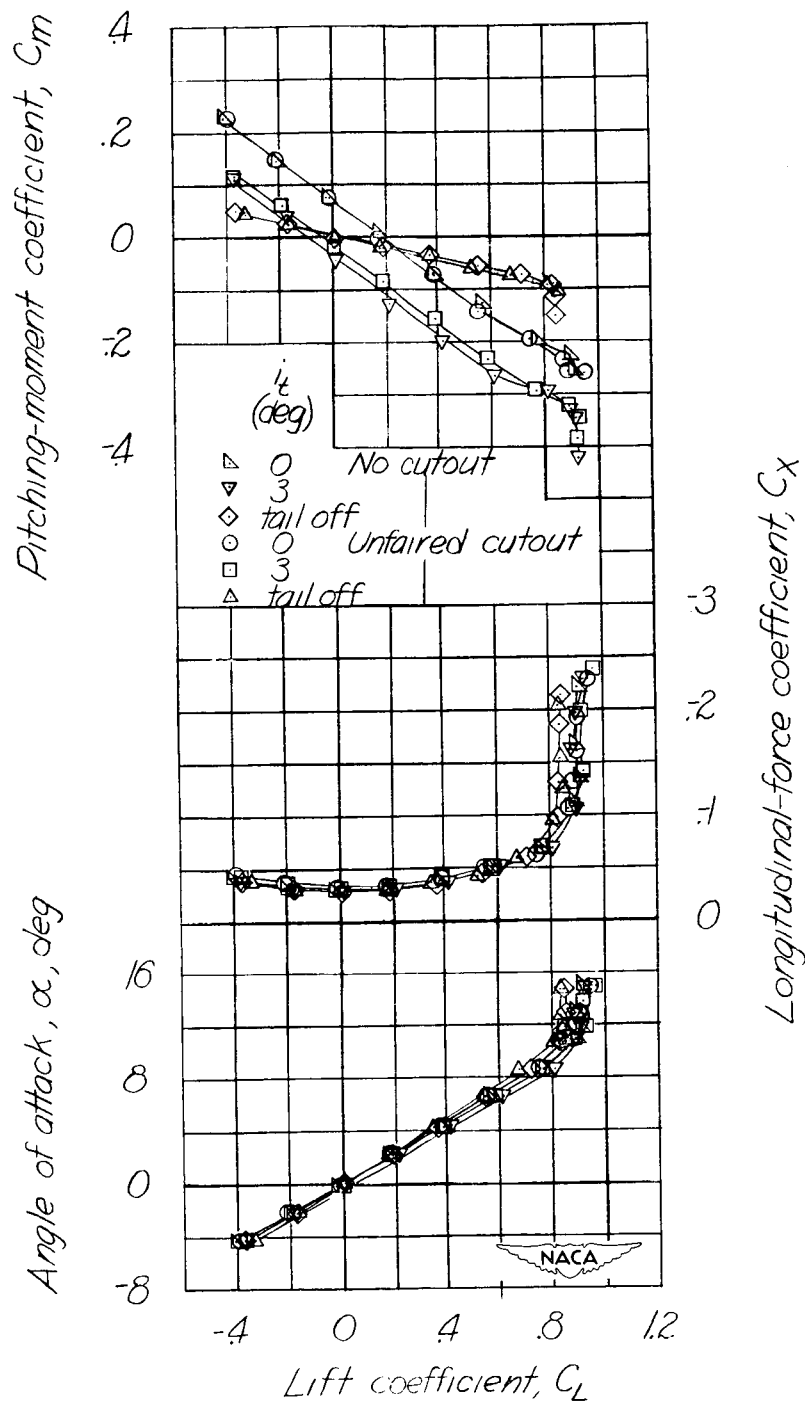
(c) External airfoil flaps.

Figure 7.- Continued.



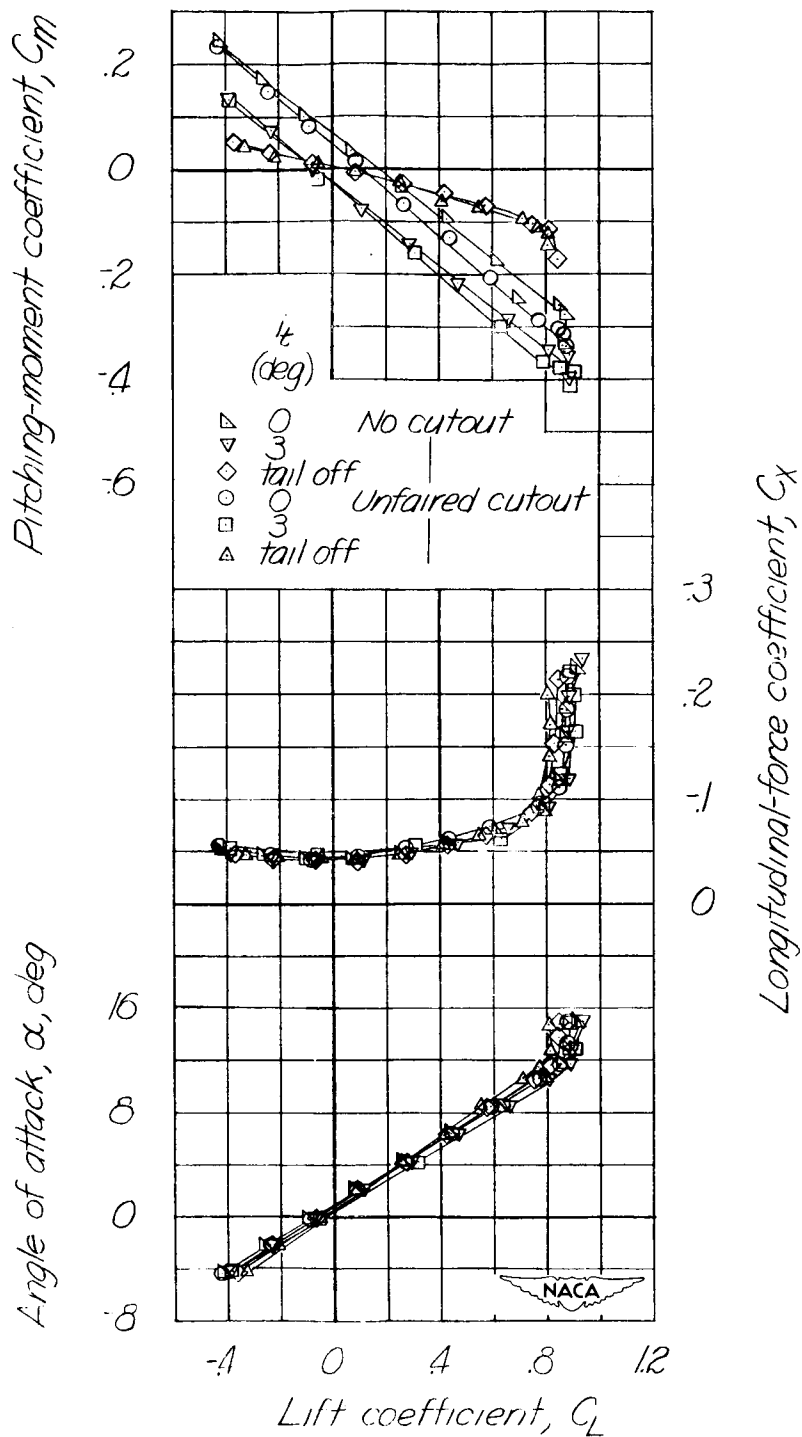
(d) External airfoil flaps, alternate tail position.

Figure 7.- Continued.



(e) Sharp-leading-edge wing section.

Figure 7.- Continued.



(f) Wing vane.

Figure 7.- Concluded.

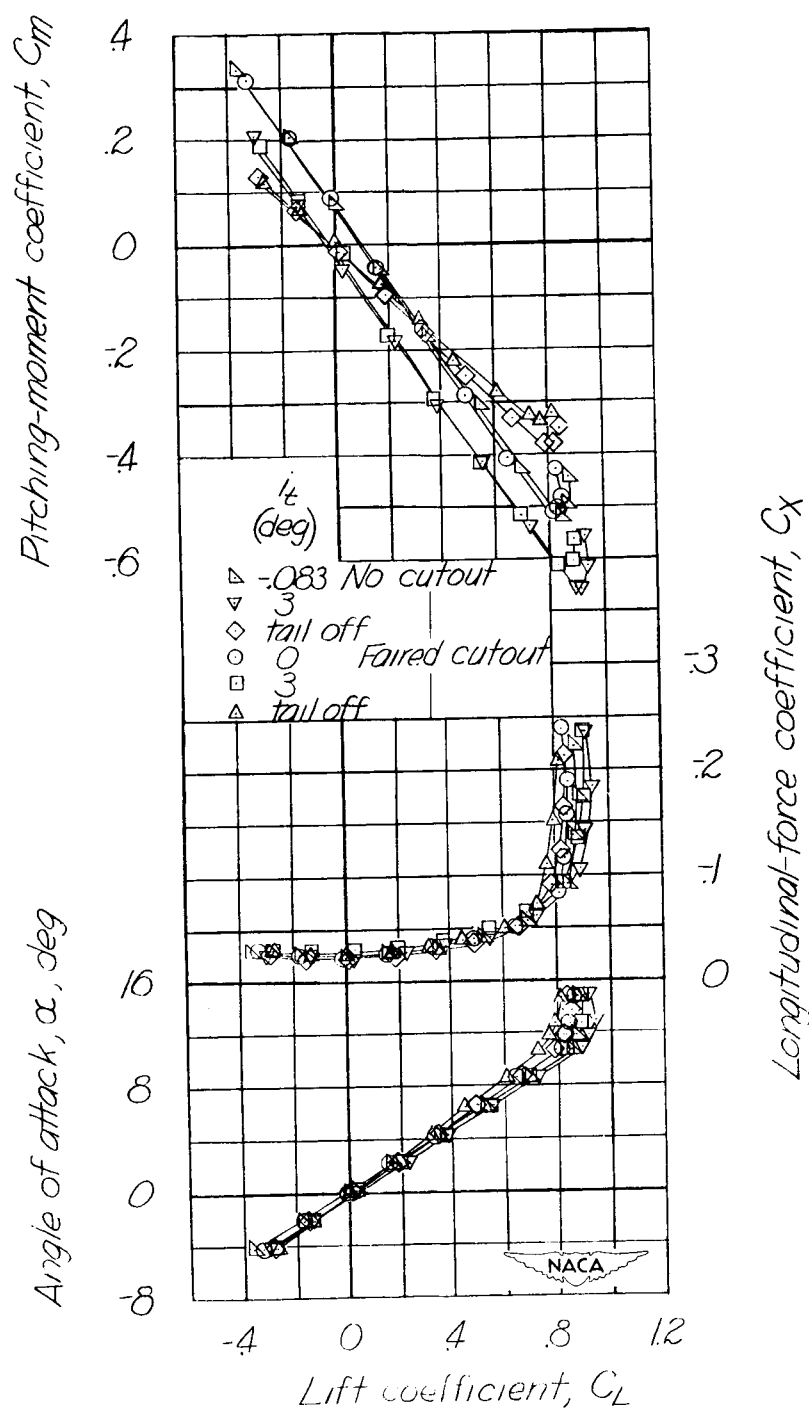
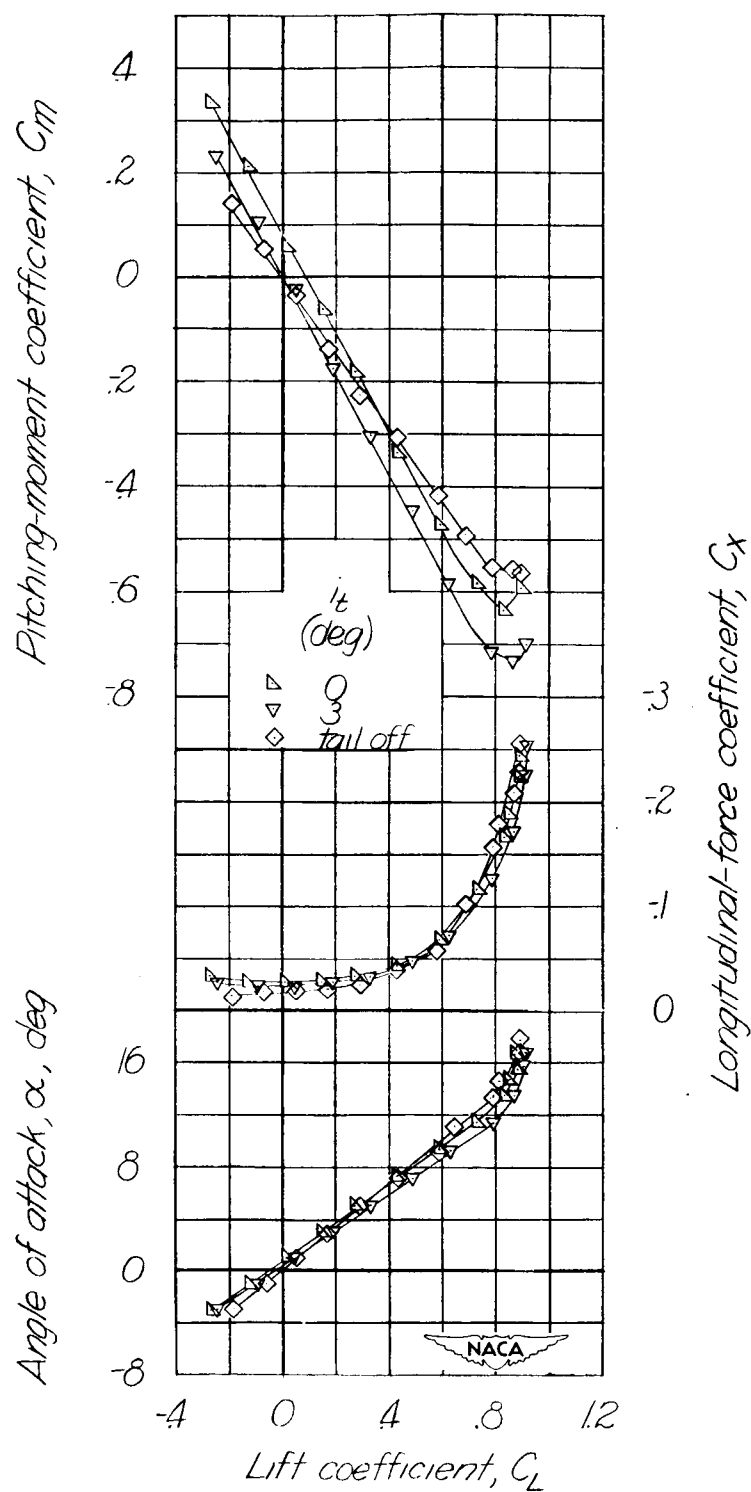


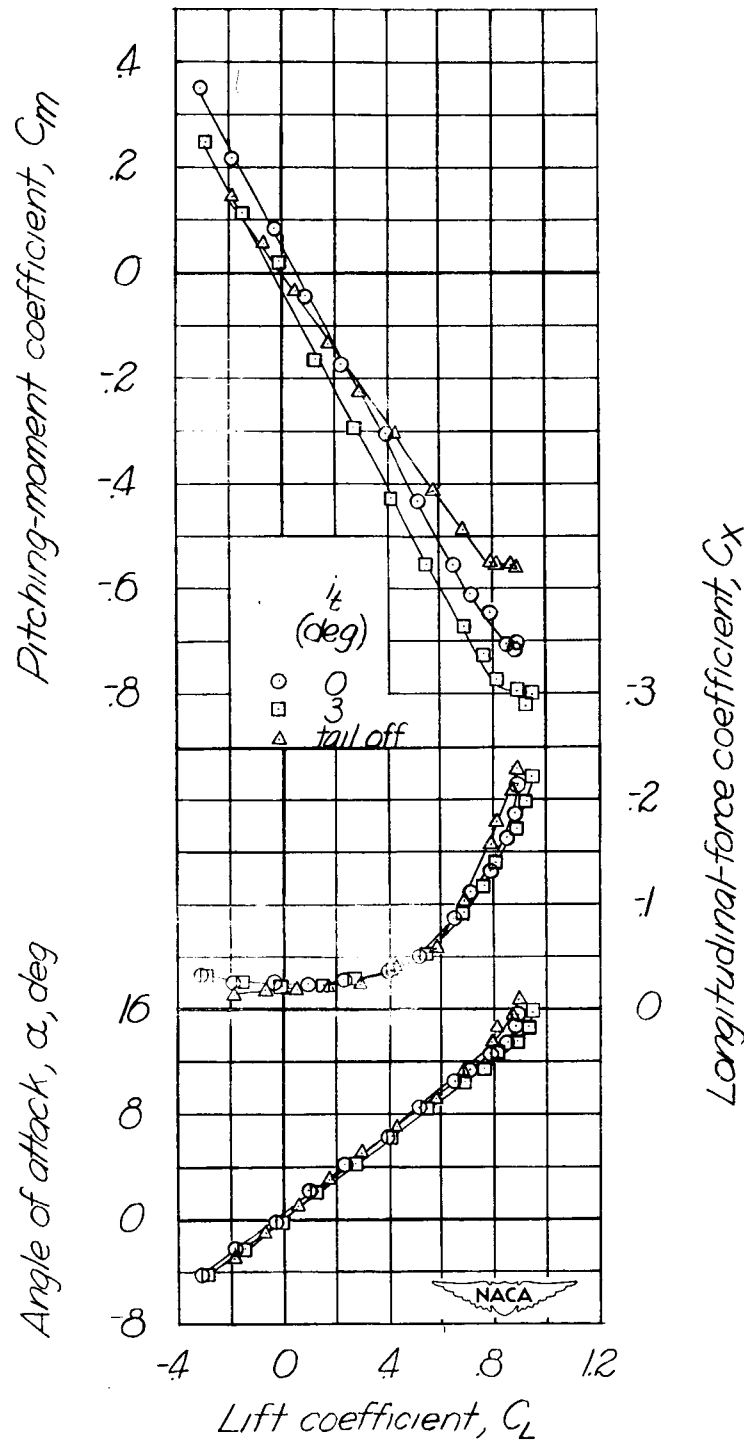
Figure 8.- Aerodynamic characteristics of a variable-sweep model with and without faired wing cutout.  $\Lambda = 30^\circ$ .





(a) Basic tail position.

Figure 9.- Aerodynamic characteristics of a variable sweep model with two horizontal-tail positions.  $\Lambda = 45^\circ$ .



(b) Alternate tail position.

Figure 9.- Concluded.

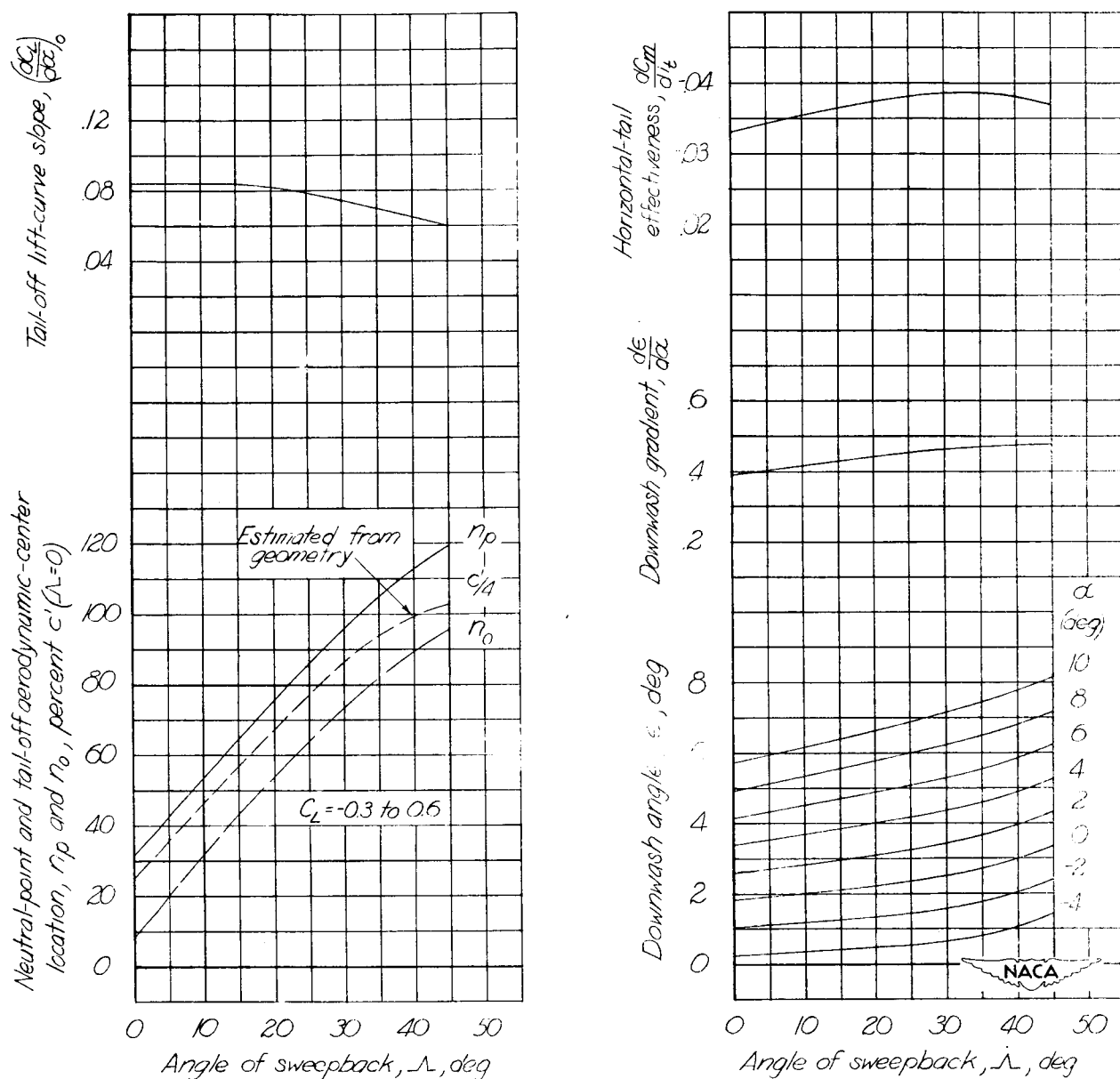


Figure 10.— Variation of longitudinal-stability parameters with angle of sweepback for a variable-sweep model. Basic tail position; wing without cutout.

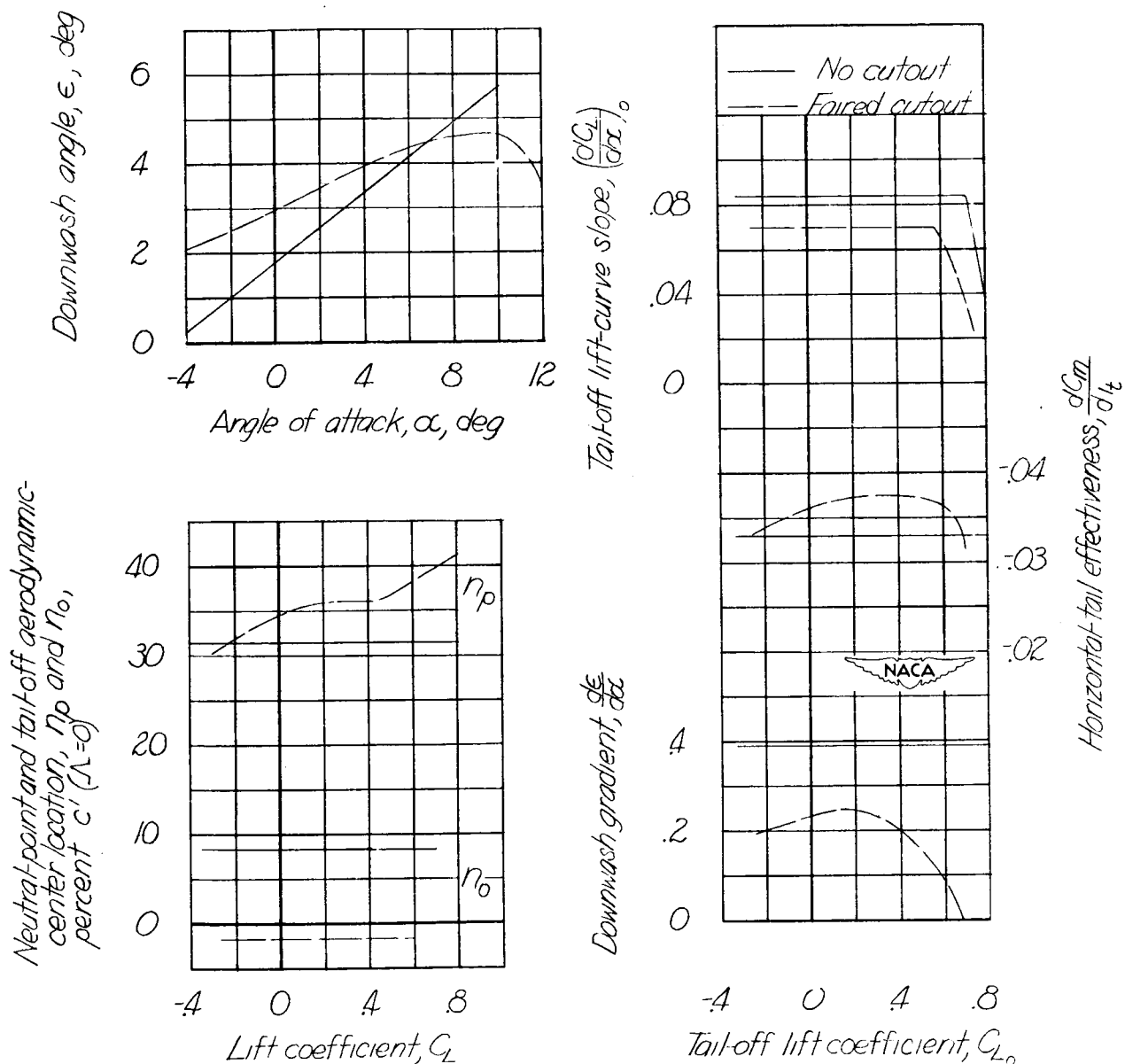
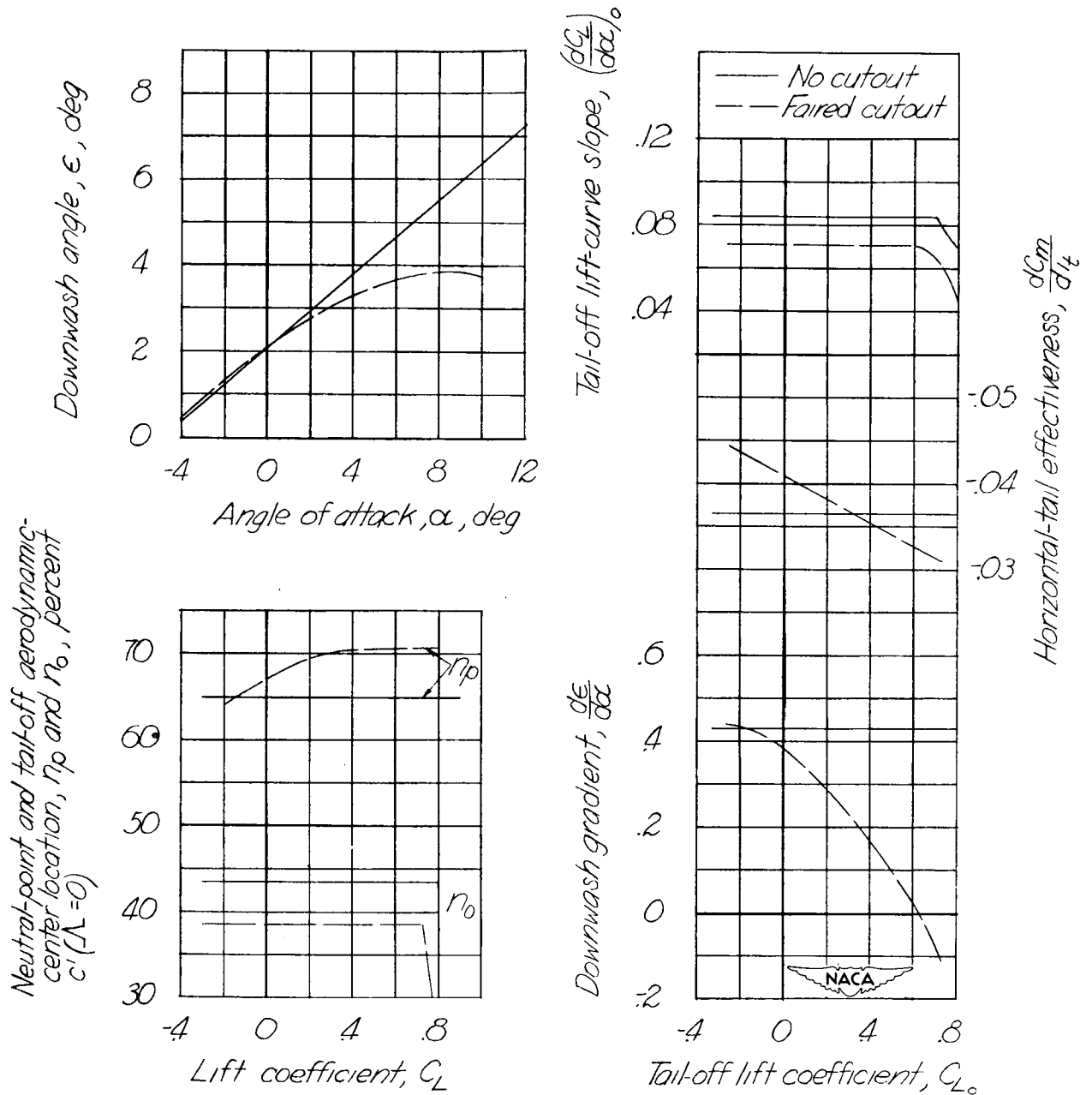
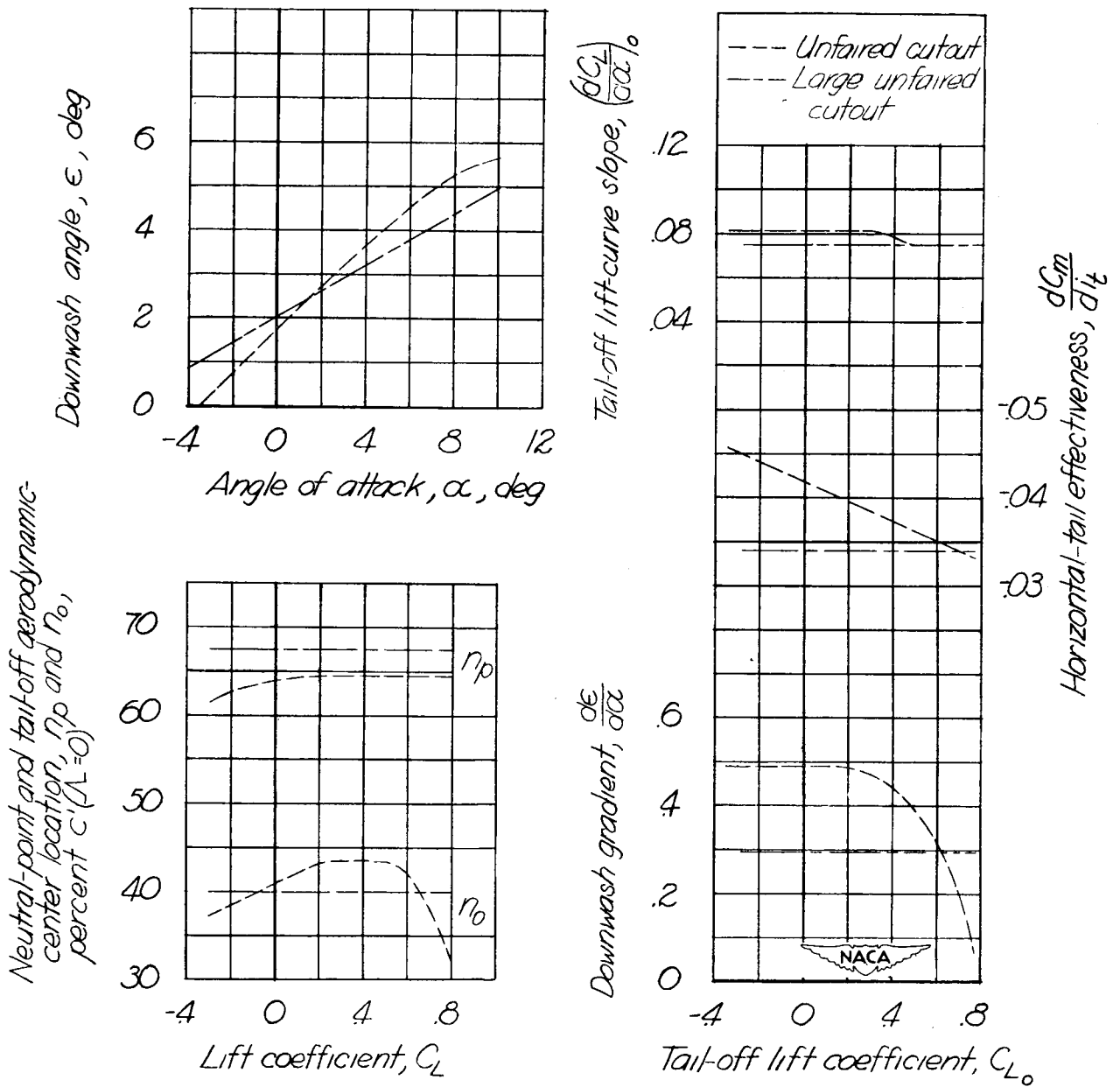


Figure 11.— Longitudinal-stability parameters of a variable-sweep model with and without faired wing cutout.  $\Lambda = 0^\circ$ .



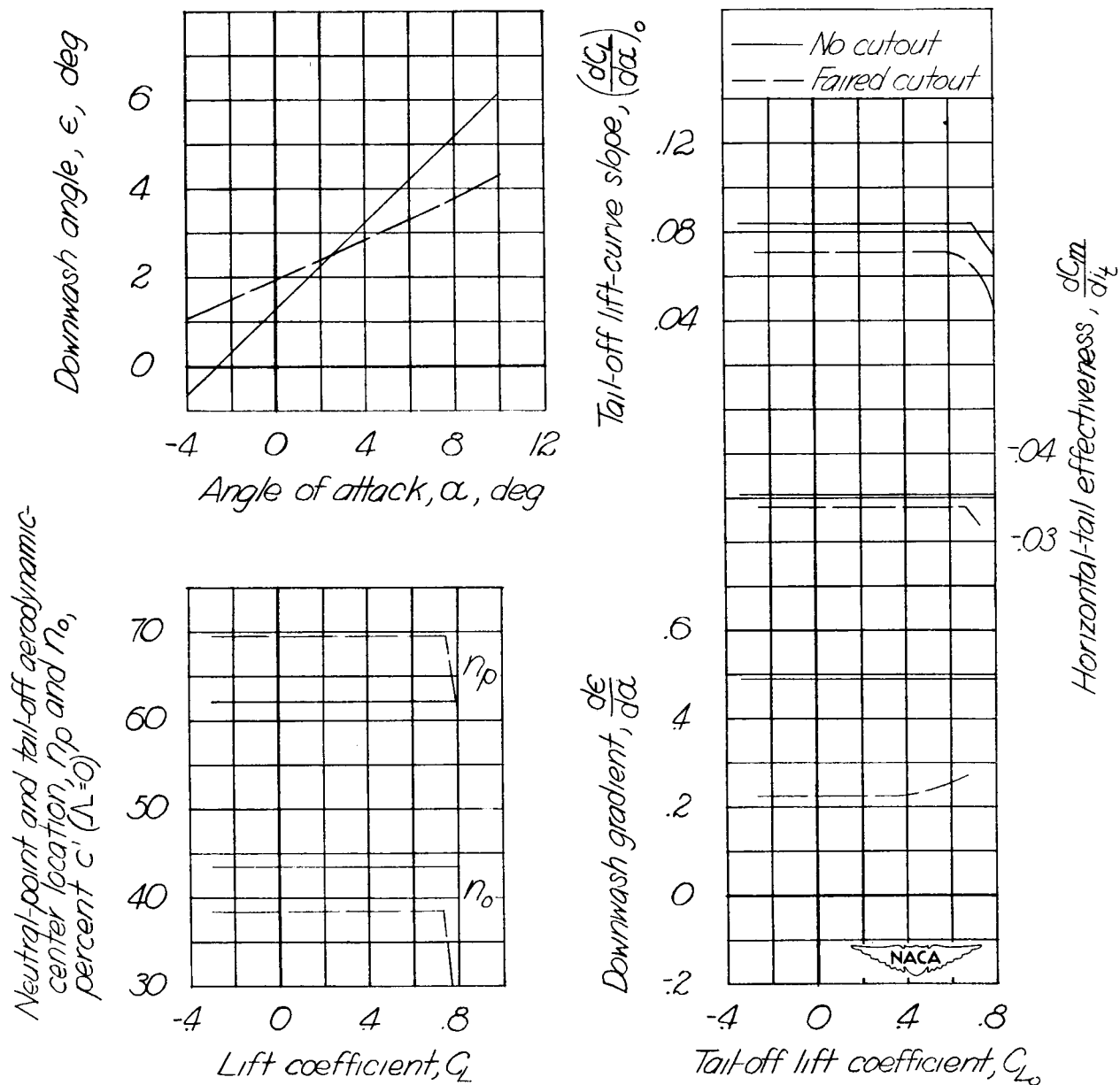
(a) Basic tail position.

Figure 12.— Longitudinal-stability parameters of a variable-sweep model with and without wing cutouts.  $\Lambda = 15^\circ$ .



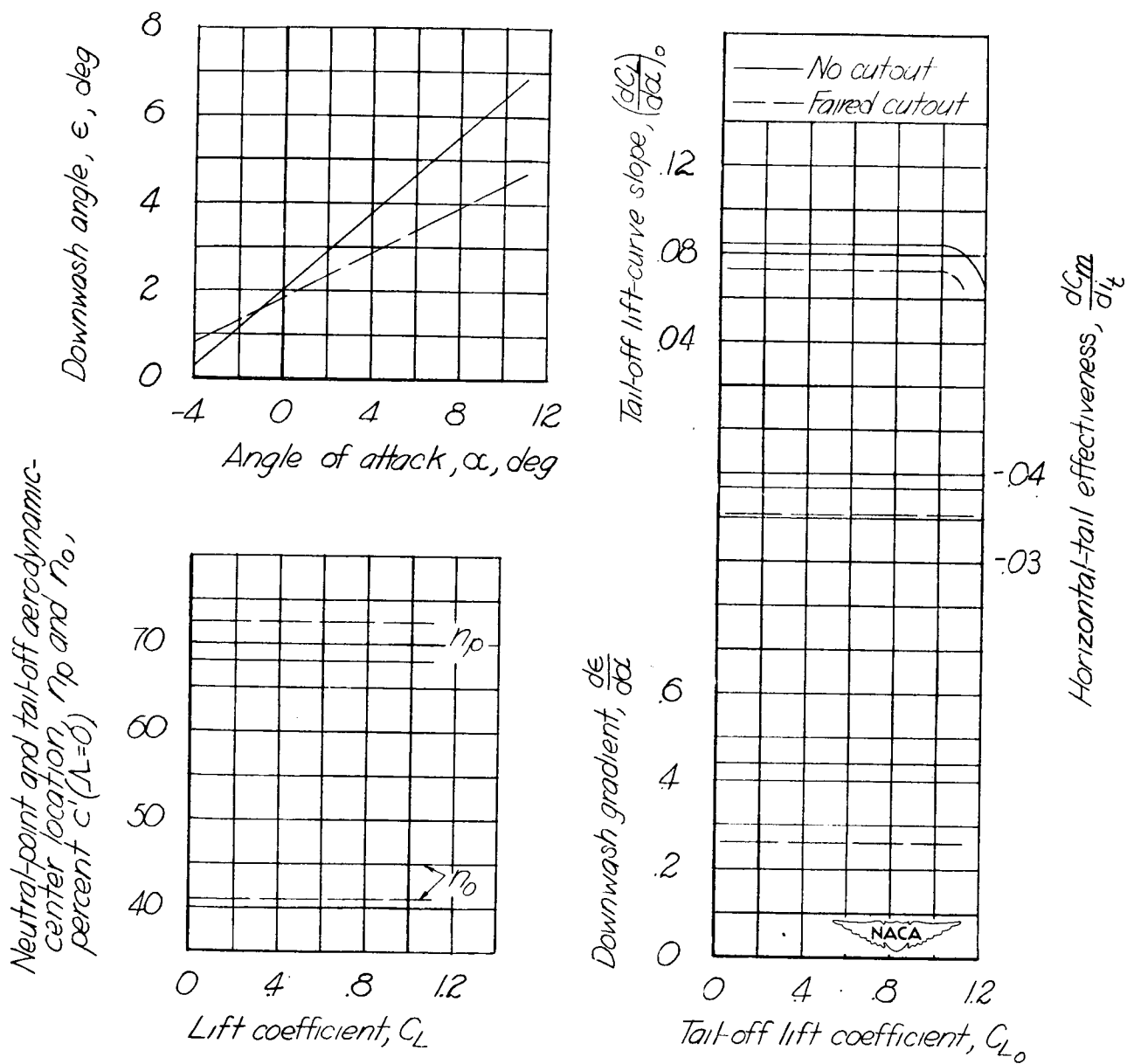
(a) Concluded.

Figure 12.- Continued.



(b) Alternate tail position.

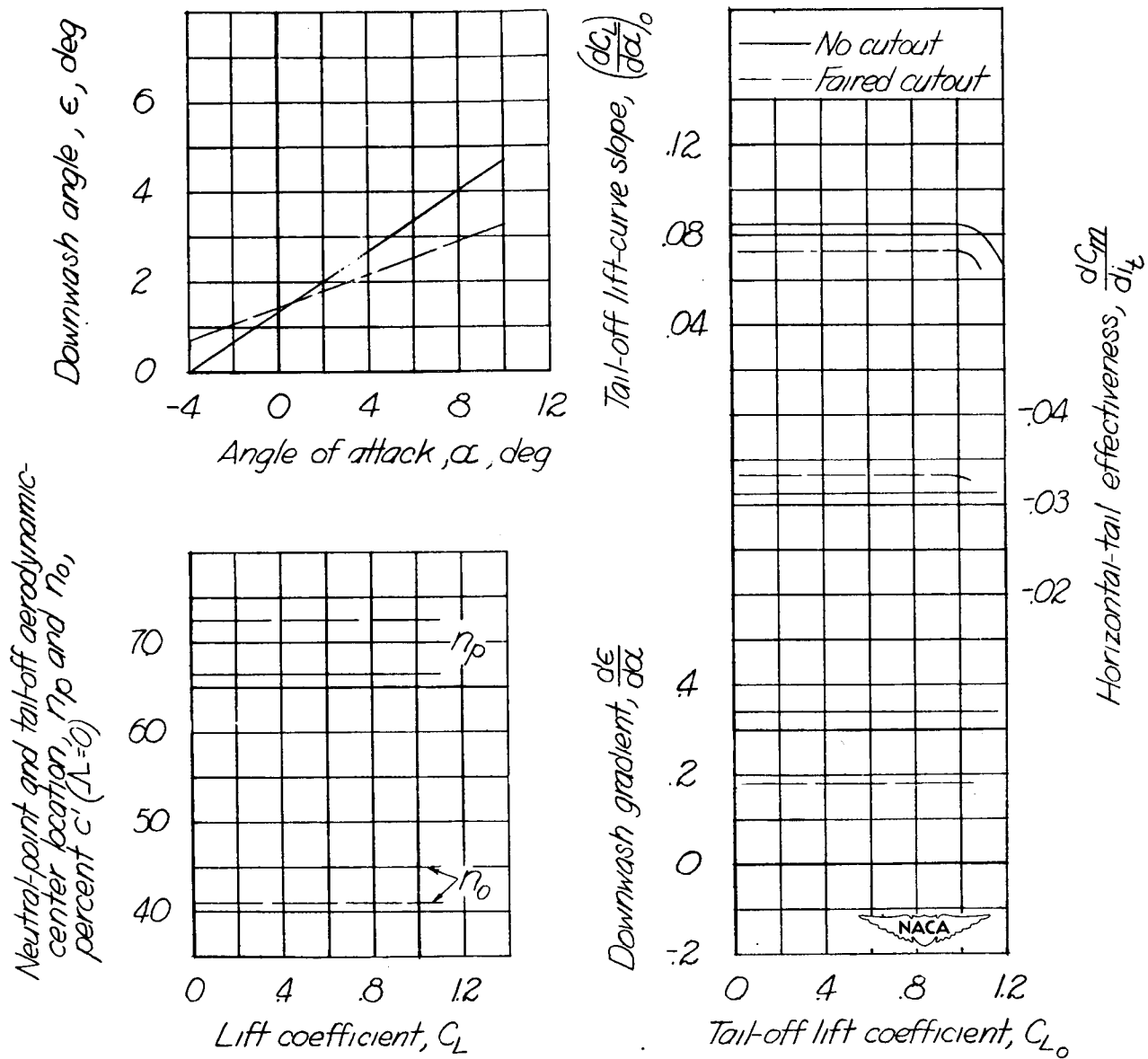
Figure 12.— Continued.



(c) External airfoil flaps.

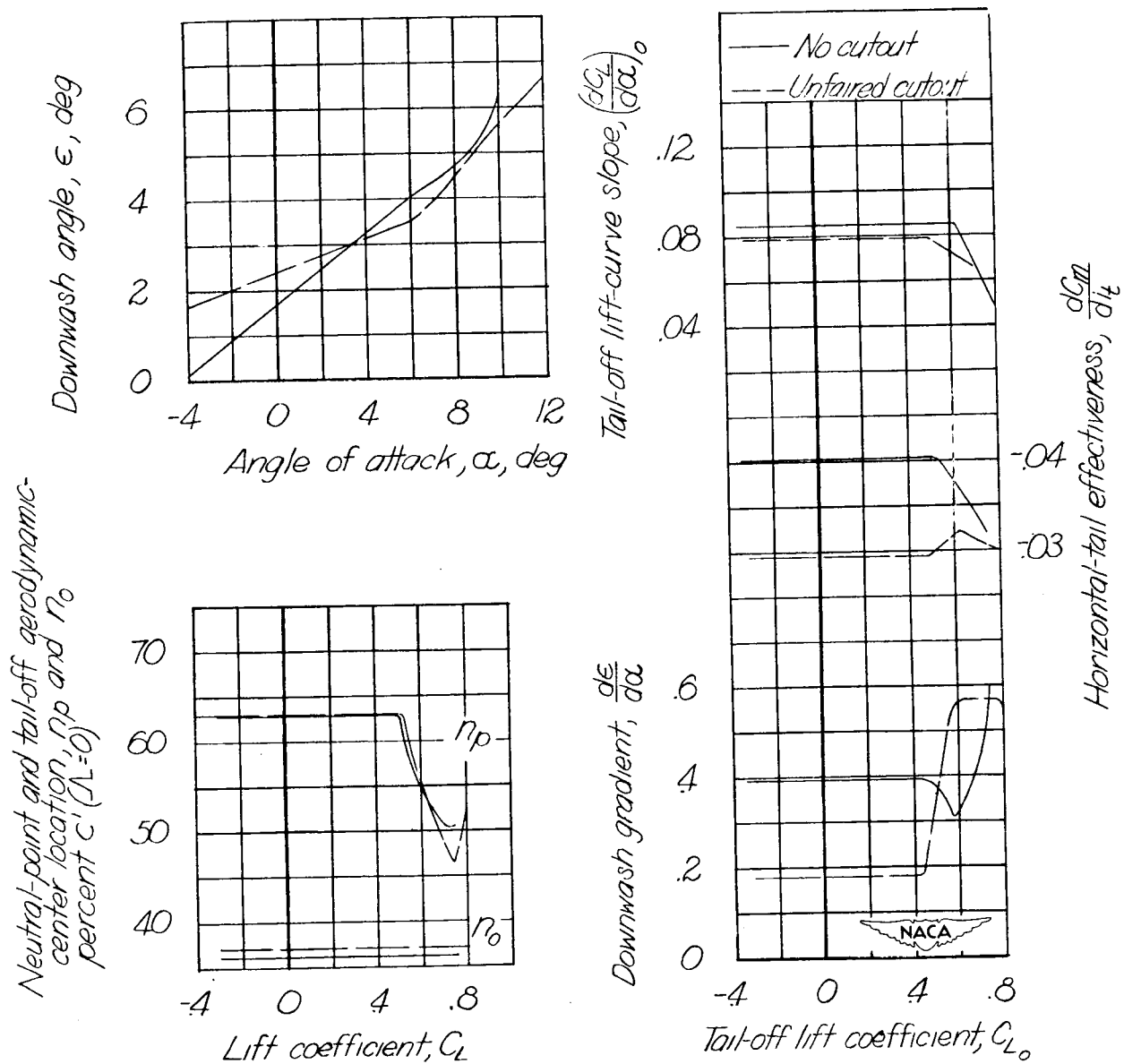
Figure 12.- Continued.





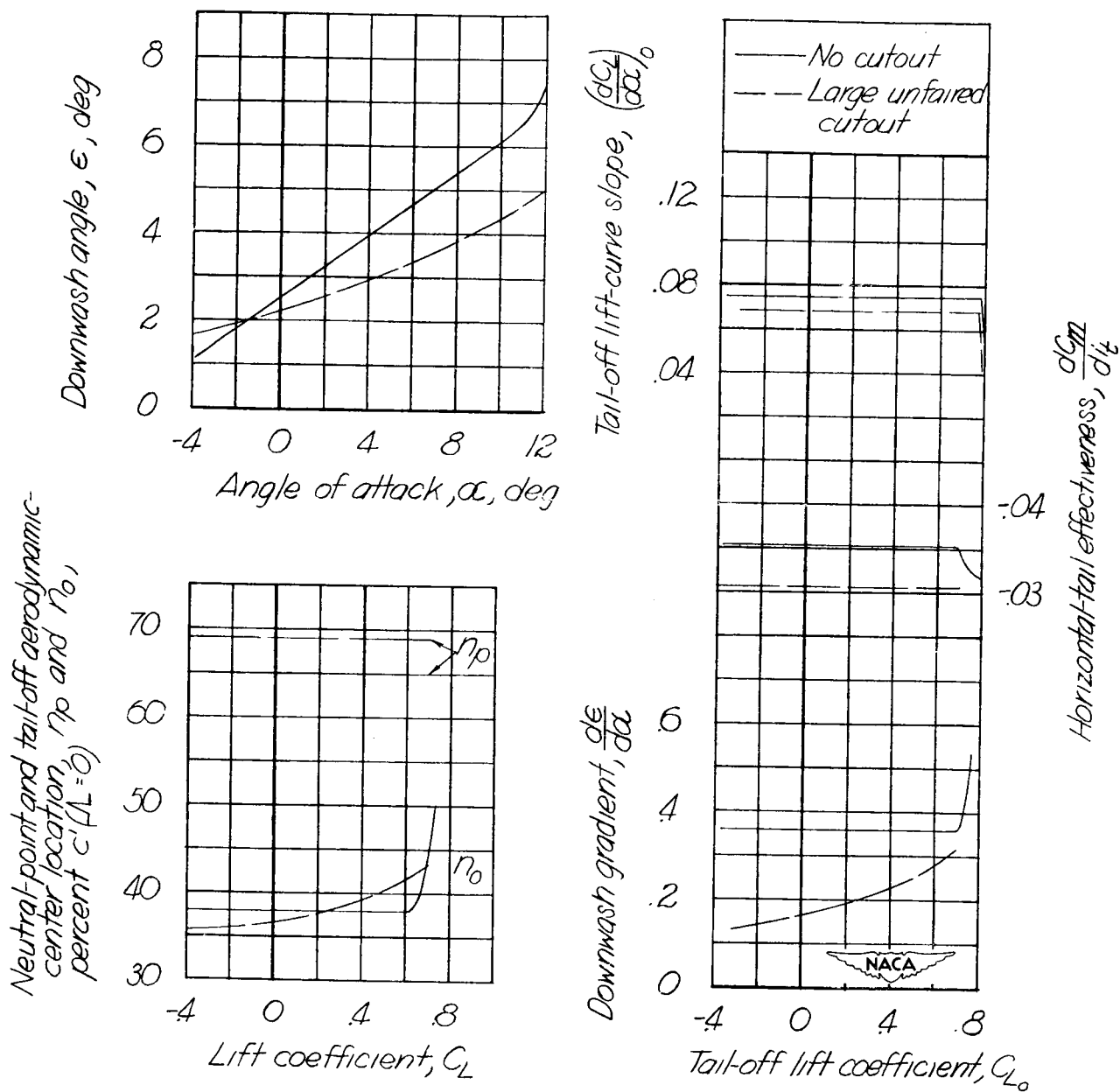
(d) External airfoil flaps, alternate tail position.

Figure 12.- Continued.



(e) Sharp-leading-edge wing section

Figure 12.- Continued.



(f) Wing vane.

Figure 12.- Concluded.

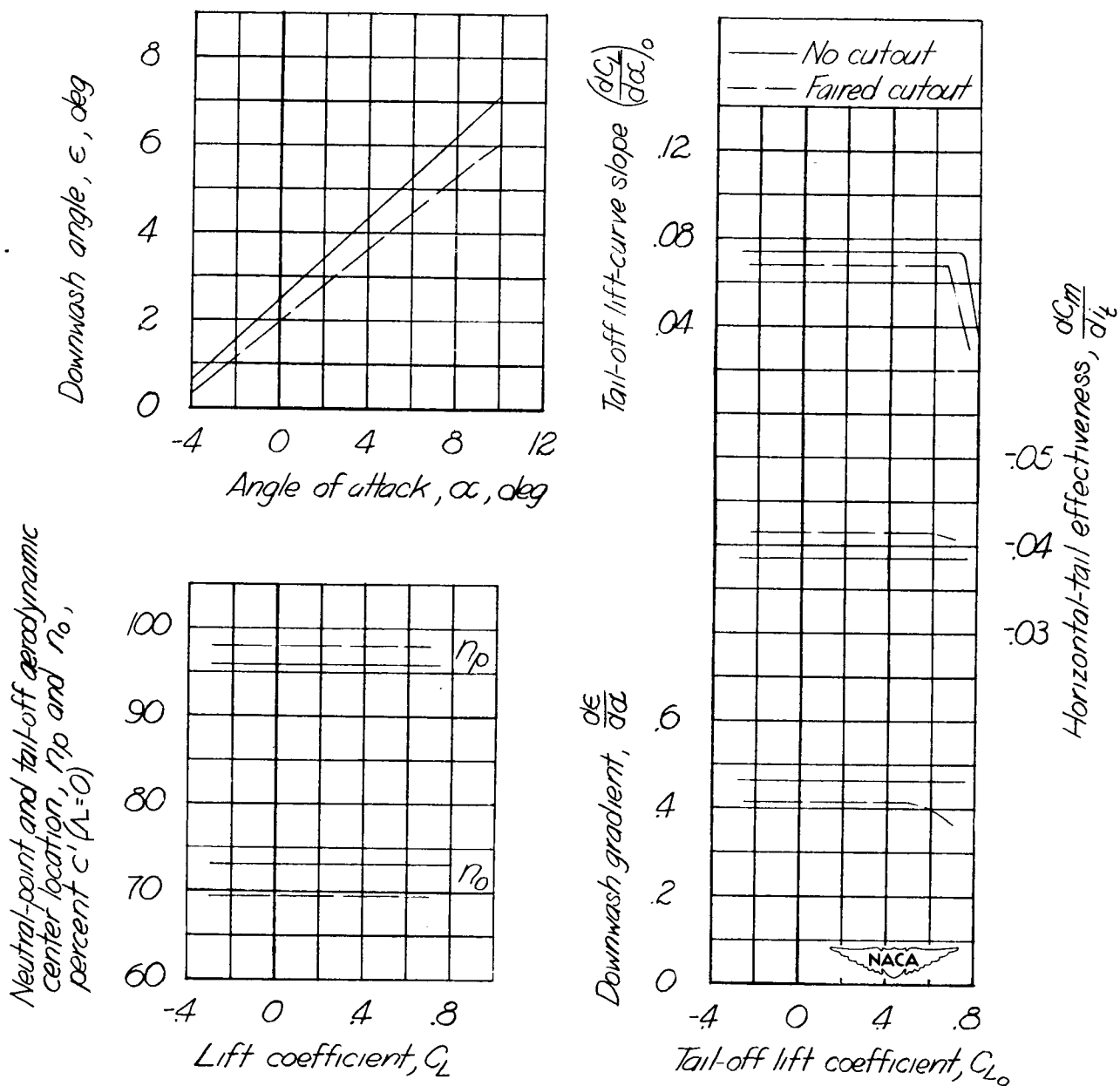


Figure 13.— Longitudinal-stability parameters of a variable-sweep model with and without faired wing cutout.  $\Lambda = 30^\circ$ .

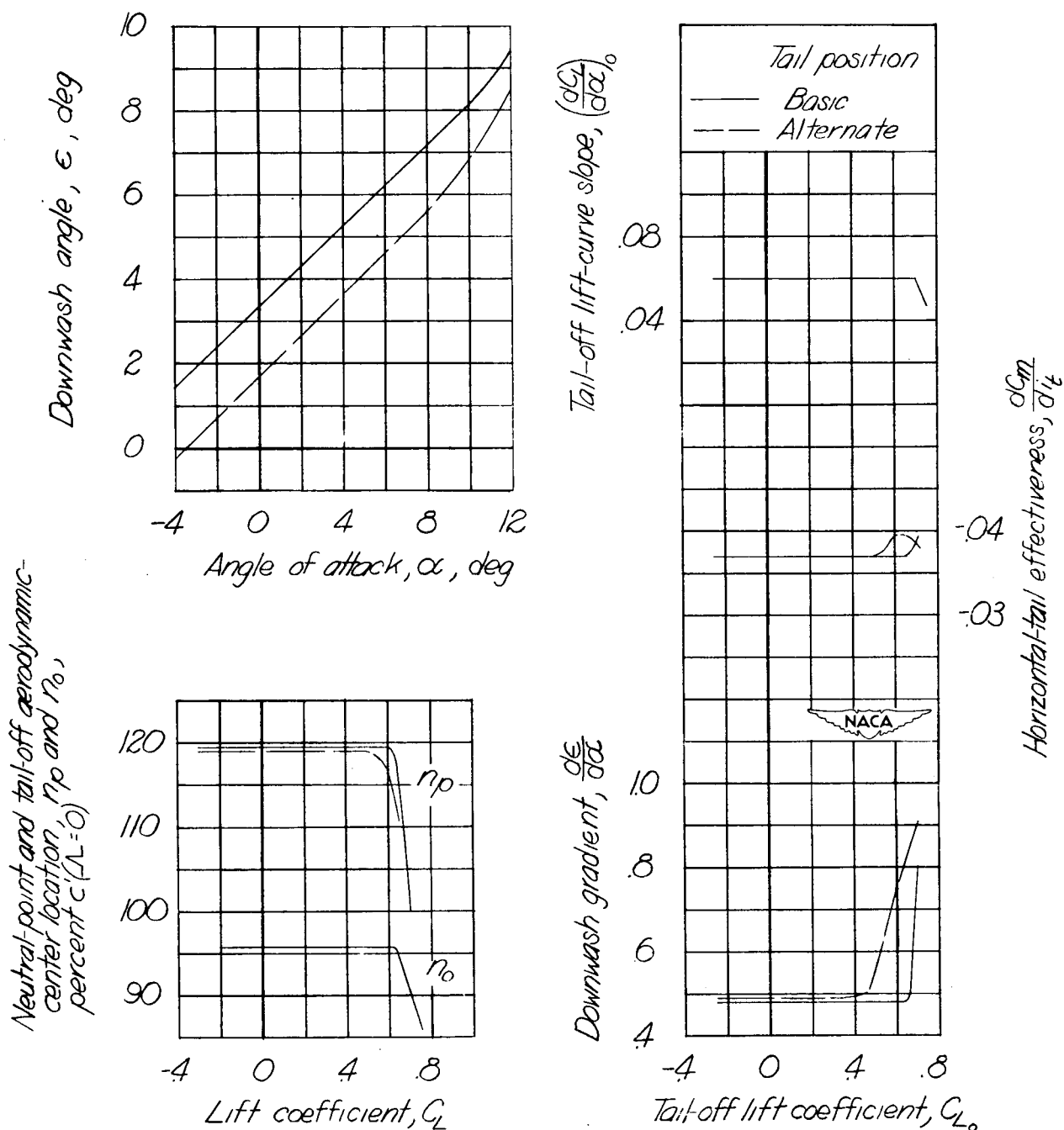


Figure 14.— Longitudinal-stability parameters of a variable-sweep model with two horizontal-tail positions.  $\Lambda = 45^\circ$ .

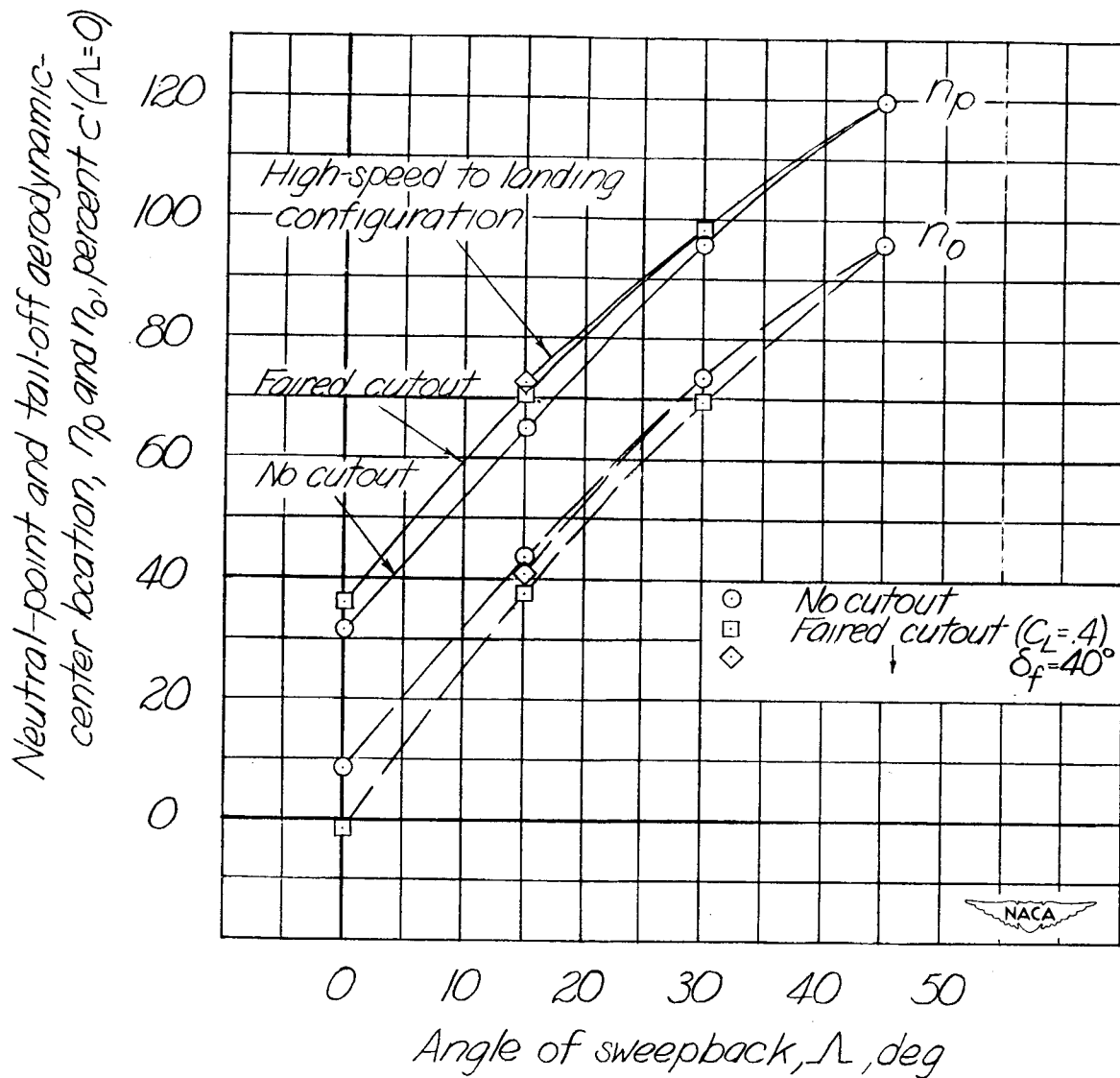


Figure 15.— Variation of longitudinal stability with sweepback and model configuration.

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# RESEARCH MEMORANDUM

ESTIMATED TRANSONIC FLYING QUALITIES OF A TAILLESS  
AIRPLANE BASED ON A MODEL INVESTIGATION

By

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NATIONAL ADVISORY COMMITTEE  
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WASHINGTON

June 8, 1949

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## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## RESEARCH MEMORANDUM

## ESTIMATED TRANSONIC FLYING QUALITIES OF A TAILLESS

## AIRPLANE BASED ON A MODEL INVESTIGATION

By Charles J. Donlan and Richard E. Kuhn

## SUMMARY

An analysis of the estimated flying qualities of a tailless airplane with the wing quarter-chord line swept back  $35^\circ$  in the Mach number range from 0.40 to 0.91 has been made, based on tests of a model of this airplane in the Langley high-speed 7- by 10-foot tunnel.

The analysis indicates longitudinal-control position instability at transonic speeds but the accompanying trim changes are not large. Control-position maneuvering stability, however, is present for all speeds. Longitudinal and lateral control appear adequate, but the damping of the short-period longitudinal and lateral oscillations at high altitudes is poor and would probably require artificial damping.

## INTRODUCTION

Stability and control tests of a tailless-type swept-wing airplane model have been conducted in the Langley high-speed 7- by 10-foot tunnel through the Mach number range from 0.40 to 0.91. The flying qualities that might be expected from such an airplane have been estimated from these data for assumed wing loadings of 24 and 34 pounds per square foot at sea level and at an altitude of 40,000 feet. All computations are based on a center-of-gravity position of 17 percent of the mean aerodynamic chord.

The estimated flying qualities of the airplane are presented in the body of the paper and in figures 1 to 23. A discussion of the wind-tunnel tests is presented in the Appendix and the data are presented in figures 24 to 41.

## COEFFICIENTS AND SYMBOLS

The system of axes employed, together with an indication of the positive forces, moments, and angles, is presented in figure 1. Pertinent symbols used in this paper are defined as follows:



|                   |                                                                                       |
|-------------------|---------------------------------------------------------------------------------------|
| $C_L$             | lift coefficient (Lift/qS)                                                            |
| $C_D$             | drag coefficient (Drag/qS)                                                            |
| $C_m$             | pitching-moment coefficient (Pitching moment/qS $\bar{c}$ )                           |
| $C_l$             | rolling-moment coefficient (Rolling moment/qSb)                                       |
| $C_Y$             | side-force coefficient (Side force/qS)                                                |
| $C_n$             | yawing-moment coefficient (Yawing moment/qSb)                                         |
| q                 | free-stream dynamic pressure, pounds per square foot ( $\frac{\rho V^2}{2}$ )         |
| S                 | wing area                                                                             |
| $\bar{c}$         | wing mean aerodynamic chord (M.A.C.) ( $\bar{c} = \frac{2}{S} \int_0^{b/2} c^2 db$ )  |
| c                 | chord, parallel to plane of symmetry                                                  |
| b                 | wing span                                                                             |
| V                 | air velocity, feet per second                                                         |
| P                 | rolling velocity, degrees or radians per second                                       |
| r                 | yawing velocity, radians per second                                                   |
| q                 | pitching velocity, radians per second                                                 |
| a                 | speed of sound, feet per second                                                       |
| M                 | Mach number (V/a)                                                                     |
| R                 | Reynolds number ( $\frac{\rho V \bar{c}}{\mu}$ )                                      |
| $\mu$             | absolute viscosity, pounds-seconds per square foot                                    |
| $\rho$            | mass density of air, slugs per cubic foot                                             |
| $\alpha$          | angle of attack, measured from X-axis to fuselage center line, degrees                |
| $\alpha_{static}$ | angle of attack of model under no-load conditions                                     |
| $\delta$          | control deflection, measured on chord line parallel to the plane of symmetry, degrees |
| $\psi$            | angle of yaw, degrees                                                                 |

|            |                                                                                                                                             |
|------------|---------------------------------------------------------------------------------------------------------------------------------------------|
| $\beta$    | angle of sideslip, radians                                                                                                                  |
| $\eta$     | angle of attack of principal longitudinal axis of airplane,<br>positive when principal axis is above flight path at the<br>nose, degrees    |
| $\epsilon$ | angle between fuselage center line and principal axis of<br>inertia, positive when fuselage center line is above<br>principal axis, degrees |
| $\gamma$   | angle of flight path to horizontal axis, positive in climb,<br>degrees                                                                      |
| $L/D$      | lift-drag ratio ( $C_L/C_D$ )                                                                                                               |
| $W/S$      | wing loading (Weight/S)                                                                                                                     |

$$C_{m\delta} = \frac{\partial C_m}{\partial \delta}$$

$$C_{l\psi} = \frac{\partial C_l}{\partial \psi}$$

$$C_{n\psi} = \frac{\partial C_n}{\partial \psi}$$

$$C_{Y\psi} = \frac{\partial C_Y}{\partial \psi}$$

$$C_{l_p} = \frac{\partial C_l}{\partial \left(\frac{pb}{2V}\right)}$$

$$C_{n_p} = \frac{\partial C_n}{\partial \left(\frac{pb}{2V}\right)}$$

$$C_{Y_p} = \frac{\partial C_Y}{\partial \left(\frac{pb}{2V}\right)}$$

$$C_{l_r} = \frac{\partial C_l}{\partial \left(\frac{rb}{2V}\right)}$$

$$C_{n_r} = \frac{\partial C_n}{\partial \left( \frac{rb}{2V} \right)}$$

$$C_{Y_r} = \frac{\partial C_Y}{\partial \left( \frac{rb}{2V} \right)}$$

$$C_{L_\alpha} = \frac{\partial C_L}{\partial \alpha}$$

$$C_{m_q} = \frac{\partial C_m}{\partial \left( \frac{q\bar{c}}{2V} \right)}$$

$k_{X_0}$  radius of gyration in roll about body axes, feet

$k_{Y_0}$  radius of gyration in pitch about body axes, feet

$k_{Z_0}$  radius of gyration in yaw about body axes, feet

Subscripts:

a aileron

l left

r right

#### MODEL AND AIRPLANE

The test model represented a tailless, swept-wing, jet-propelled, fighter-type airplane. The physical characteristics of the solid-steel model are presented in figure 2, and pictures of the model mounted on the sting-support systems used for this investigation are presented in figure 3. For the portions of analysis for which full-scale airplane dimensions were required, a model scale of 0.08 was assumed. The control surfaces, which are plain flaps with sealed gaps, are intended to be used for both longitudinal and lateral control. Rudders were not simulated on the model. Air flow through the jet-intake ducts was permitted for all tests, and one of the exhaust ports, together with its mirror image, can be seen in figure 3(a).

## BASIS OF ANALYSIS

The most recent specification for satisfactory flying qualities (reference 1) has been used as a guide in the present analysis. However, inasmuch as the analysis is restricted to the high-speed configuration without regard to control forces (no model hinge-moment data were obtained) and because much of the interest centers about the behavior of the airplane at speeds above those at which adverse compressibility effects are encountered, no detailed step-by-step comparison with the specifications has been attempted.

The estimated characteristics of the aircraft at each Mach number are based upon the results of tunnel tests at the same Mach number but at the test Reynolds number indicated in figure 4. The full-scale Reynolds numbers corresponding to flight at sea level and at an altitude of 40,000 feet are also shown in figure 4. No attempt was made to account for Reynolds number effects in interpreting the results. It is of interest to note, however, that a few unpublished tests made with transition fixed at the leading edge in order to simulate flow conditions at high Reynolds numbers were in good agreement with the basic free-transition tests. This indicates that, although the bulk of the data was obtained with free transition, the model data were not obtained in a critical range of Reynolds number.

## RESULTS AND DISCUSSION

## Performance

Flight conditions.-- The variations with Mach number of the lift coefficient required for level flight for the various wing loadings and altitudes considered in the analysis are given in figure 5 and the corresponding angle-of-attack variation is given in figure 6. Figure 6 is useful for estimating the inclination of the principal axes of inertia for the different flight conditions. It will be observed that the angle of attack for level flight at sea level for the lighter wing loading becomes slightly negative at the highest Mach numbers. This condition, of course, is a result of the shift in angle of zero lift effected by the deflected elevator required for balance.

Lift-drag ratios.-- The variation of the untrimmed lift-drag ratios at the various Mach numbers as a function of the lift coefficient is presented in figure 7. It will be observed that the lift coefficient for maximum  $L/D$  is essentially independent of Mach number, although the magnitude of the available  $L/D$  maximum drops rather rapidly above a Mach number of 0.80. The level-flight  $L/D$  values associated with the trimmed-flight conditions defined in figure 5 are presented in figure 8. The advantages to be gained by flying at high altitude are forcefully illustrated by this figure.

## Longitudinal Stability and Control

Strictly speaking the elevator deflections for the various configurations discussed in the following paragraphs are slightly in error (about  $1/3$  of a deg too much down elevator) because the data used in the analysis were not corrected for the additional pitching-moment correction discussed in the appendix.

Static longitudinal stability.- The static longitudinal stability of the airplane is presented in figure 9 in the form of the variation of the elevator position required for trim with Mach number. Control-position instability is first manifested at a Mach number of 0.90 at sea level and at a Mach number of 0.85 at an altitude of 40,000 feet. The causes of the control-position instability exhibited above these Mach numbers are traceable to the rapid changes occurring in the basic untrimmed pitching-moment coefficient (fig. 10(a)) and to the changes in control effectiveness (fig. 10(b)). The resultant changes in trim, however, appear to be relatively gradual and of moderate magnitude, at least to a Mach number of 0.91, and may not be objectionable.

A rigorous evaluation of the neutral-point location (center-of-gravity position for which  $\frac{d\delta}{dM} = 0$ ) at these Mach numbers would indeed indicate that the control-fixed neutral point moves well ahead of the center-of-gravity position. However, the utility of the neutral-point concept largely vanishes when irregular and rapid changes in trim occur. The desired information on static longitudinal stability appears to be most directly conveyed through charts like figure 9.

Maneuvering stability.- For tailless aircraft which possess very little damping in pitch, the factor  $\left(\frac{\partial C_m}{\partial C_L}\right)_M$  very nearly defines the stick-fixed "maneuver margin" - the distance, expressed as a fraction of the chord, that the center of gravity is ahead of the "maneuver point." (The maneuver point is the center-of-gravity position for which the rate of change of control deflection with normal acceleration vanishes.)

The variation of the maneuver-point location with Mach number is presented for several lift coefficients in figure 11. It is evident that the maneuver point moves rearward, in general, at the higher Mach numbers. However, because of the nonlinearities involved in the evaluation of the maneuver point, its influence can be studied more conveniently in conjunction with the evaluation of the effectiveness of the longitudinal control.

Longitudinal-control effectiveness.- The amount of elevator control required for various accelerated-flight conditions is presented in figure 12. For flight at sea level (figs. 12(c) and 12(d)), only

about  $1^\circ$  of elevator is required to produce a 6g acceleration at a Mach number of 0.85. The elevator must always be moved in the desired direction, however, as would be expected from the maneuver-point movement previously discussed (fig. 11). The minimum degree of stick-position maneuvering stability that can be tolerated will depend on the associated stick-force gradient. A small stick-position gradient, however, may make it difficult to design the control system to supply an adequate force gradient and still keep the maximum control force for other conditions within the capabilities of the pilot. At altitude of 40,000 feet (figs. 12(a) and 12(b)), much larger control deflections are required for the accelerated-flight conditions which makes the design of the control system even more critical.

Dynamic stability.— The characteristics of the stick-fixed short-period longitudinal oscillation are presented in figures 13 to 16. The computations are based on the formulas of reference 2 and the appropriate parameters in table I. While it is desirable that the short-period oscillation be damped to one-tenth amplitude in one cycle, it is obvious from figure 16 that this tailless design would not meet such a requirement at altitude. For the altitude case, it is seen that an oscillation of about 40 percent of the original amplitude still persists after one complete oscillation. At sea level, on the other hand, the damping of the oscillation appears to be adequate.

The damping characteristics have been evaluated for the control-fixed condition although the specifications are based upon free controls. However, if an irreversible control system were used on this airplane, the fixed-control characteristics would dictate the behavior of the aircraft.

### Lateral Stability and Control

Lateral stability parameters.— Because of the absence of any rudder data from which trimmed yawed conditions could be evaluated, the directional and lateral stability will be adjudged from the stability parameters presented in figure 17. In general, the data indicate adequate static lateral stability. It will be noted, however, that the speed brakes decrease the directional stability and produce a slight negative dihedral effect (negative  $C_{l_\psi}$ ) at the highest Mach numbers.

Lateral control.— The lateral-control characteristics of the airplane are presented in figure 18 in the form of the variation with Mach number of the wing-tip helix angle  $\frac{pb}{2V}$  obtained with various total aileron deflections. The helix angle was computed from the simple relation  $\frac{pb}{2V} = -\frac{C_l}{C_{l_p}}$  using the aileron rolling-moment data presented in

figure 34 and the damping characteristics given in figure 18. The damping coefficients were estimated by the method of reference 3. Some unpublished experimental  $C_{lp}$  data indicate that for this wing plan form the theoretical values are in good agreement with experiment.

The rate of roll expressed in degrees per second is presented in figure 19. Aeroelastic distortion effects would undoubtedly decrease the rates of roll from those indicated in figure 19, but in any event the rates of roll should be extremely high. It will be noted that, as in the case of longitudinal control, lateral-control effectiveness begins to decrease rapidly at the highest Mach numbers.

It is evident from the extremely rapid rates of roll possible on this airplane that the limiting rate of roll will probably be conditioned by the pilot's ability to withstand the angular accelerations imposed.

Dynamic stability.- Using the parameters presented in table I, the characteristics of the control-fixed lateral oscillations have been evaluated by the method of reference 4 and are presented in figures 20 to 23. The values of  $C_{lp}$  presented in this table are slightly different from those given in figure 18, but the effect of this difference on the dynamic stability characteristics was found to be negligible.

It will be noted from figure 23 that the damping of the oscillation is marginal for the sea-level conditions and is definitely unsatisfactory for the altitude conditions according to the desired damping criterion set forth in reference 1. If flight tests on airplanes of this type subsequently demonstrate the real need for additional damping, the simplest way to provide for it would be to introduce artificial damping into the system in the form of rudder control coupled to a gyroscope sensitive to yawing velocity as discussed, for example, in reference 5.

A check on spiral stability was also made for the conditions stated in figure 20. It was found that spiral instability was present at a Mach number above 0.9, but the degree of spiral instability was so slight that the time required for the angle of bank to increase 10 percent was of the order of 1 minute at an altitude of 40,000 feet and 4 minutes at sea level.

### CONCLUSIONS

An analysis of the transonic flying qualities to be expected from a tailless airplane in the Mach number range from 0.40 to 0.91 based on a model investigation indicates the following conclusions:

1. The airplane would exhibit longitudinal control-position instability at transonic speeds but the accompanying trim changes at these speeds should not be large.

2. Control-position maneuvering stability would be present at all speeds investigated although the control-position gradient may be as high as 6g's per degree of elevator deflection at low altitudes.

3. The damping of the short-period longitudinal oscillation at high altitudes would be less than desired.

4. The damping of the lateral oscillation at high altitude would be very poor and would probably require artificial damping.

5. Longitudinal and lateral control appear to be adequate at all speeds investigated.

Langley Aeronautical Laboratory  
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## APPENDIX

## WIND-TUNNEL INVESTIGATION

## Tests

Scope.- The tests covered a Mach number range of 0.40 to 0.91 and an angle-of-attack range of  $0^\circ$  to  $10^\circ$ . Yaw tests were conducted through  $\pm 4^\circ$  at  $0^\circ$  and  $6^\circ$  angle of attack. Longitudinal-control tests were conducted for  $-4.4^\circ$  to  $9.5^\circ$  elevator deflection through the angle-of-attack and Mach number range, and aileron-control tests covered  $-1.8^\circ$  to  $18.9^\circ$  deflection of the left aileron through the angle-of-attack and Mach number range. The effect of the fins, canopy, and speed brakes on the longitudinal and lateral stability and control was also investigated.

The variation of test Reynolds number with Mach number for average test conditions is presented in figure 4. The size of the model used in the present investigation resulted in a corrected tunnel choking Mach number of about 0.94. Experience has indicated that, with this value of choking Mach number, the data should be reliable up to a corrected Mach number of about 0.91.

Support system.- The model was supported by a sting extending from the rear of the fuselage to a vertical strut located behind the model. A photograph of the model supported on this system is shown in figure 3(a). The tare forces and moments produced by the center sting were determined by mounting the model on two wing supports which were also attached to the vertical strut and testing the model with and without the center sting (fig. 3(b)). For wing-alone tests the method that was employed to obtain pitching-moment tares was found to give unreliable results. Consequently, no pitching-moment data for the wing alone are presented in this report. Angles of attack and yaw were changed by the use of interchangeable couplings in the stings behind the model. Deflections of the support system under load were determined from static-loading tests.

Corrections.- The test results have been corrected for tare forces and moments produced by the support system. However, there are small additional corrections to the pitching-moment and rolling-moment coefficients which have not been incorporated in the data. These corrections, which are inherent in the balance system, were determined subsequent to the completion of the present investigation, but the data of this paper can be corrected as follows:

$$(C_m)_{\text{corrected}} = (C_m)_{\text{presented}} - 0.003$$

$$(C_l)_{\text{corrected}} = (C_l)_{\text{presented}} - 0.0008$$

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The jet-boundary corrections to the lift and drag were computed by the method of reference 6. The jet-boundary corrections to other components were considered negligible.

The drag has been corrected for the buoyancy produced by the small longitudinal static-pressure gradient in the tunnel. All coefficients and Mach numbers were corrected for blocking by the model and its wake by the method of reference 7.

## RESULTS AND DISCUSSION

The results of the wind-tunnel tests are presented in the following figures. The pitching-moment coefficients are presented about a center of gravity located at 17 percent of the mean aerodynamic chord.

### Basic Force Data:

Figure

#### Longitudinal -

|                                                                                                      |            |
|------------------------------------------------------------------------------------------------------|------------|
| Pitch tests, effect of control deflection, speed brakes, fins, canopy, and wing-alone data . . . . . | 24, 25, 26 |
| Lift-curve slope . . . . .                                                                           | 27         |
| Curves of $\left(\frac{\partial C_m}{\partial C_{L,M}}\right)$ . . . . .                             | 28         |
| Control effectiveness parameter . . . . .                                                            | 29         |

#### Lateral -

|                                                                                |        |
|--------------------------------------------------------------------------------|--------|
| Yaw tests, effect of speed brakes, fins, canopy, and wing-alone data . . . . . | 30, 31 |
| Lift coefficient of yaw tests . . . . .                                        | 32     |
| Lateral-stability derivatives . . . . .                                        | 33     |
| Lateral-control tests . . . . .                                                | 34     |
| Effect of aileron deflection on drag . . . . .                                 | 35     |
| Effect of fins on aileron effectiveness . . . . .                              | 36     |

### Miscellaneous Data:

|                                                     |    |
|-----------------------------------------------------|----|
| Tuft studies of flow over wing . . . . .            | 37 |
| Speed-brake configurations -                        |    |
| Drawing of fuselage brakes . . . . .                | 38 |
| Tuft studies . . . . .                              | 39 |
| Effect on lift, drag, and pitching moment . . . . . | 40 |
| Drag increments . . . . .                           | 41 |

Longitudinal stability and control.- The aerodynamic characteristics in pitch of the model and various components are presented in figures 24, 25, and 26. For these tests a cluster of static and total head tubes was installed in the right duct to measure the flow during tests. The inlet-velocity ratios measured were small compared to those which might be expected in flight; however, calculations have indicated that only a small pitching moment results from turning the inlet air through the angle of attack at the duct inlet.

Visual observation of tufts indicated no external flow separation from the duct inlets at any Mach number at low angles of attack. At the highest angle of attack, however, a local separation from the upper surface of the duct lip was observed at Mach numbers as low as 0.45 (fig. 37).

The elevator effectiveness parameter  $C_{m\delta}$  (fig. 10(b)) was determined from cross plots of the data from figure 24 and is defined as the slope of the pitching-moment coefficient plotted against elevator-deflection curve at zero elevator deflection. The pitching-moment coefficient was found to vary linearly with deflection through the deflection range at the lower Mach numbers. At large deflections the effectiveness was somewhat reduced at the higher Mach numbers.

The effectiveness parameter  $\left(\frac{\Delta C_m}{\Delta \delta}\right)$  (fig. 29) is based on data obtained from elevator deflections  $0^\circ$  to  $4.4^\circ$  and  $-4.4^\circ$  only.

Lateral stability.- The variation of lateral-stability characteristics with Mach number ( $\alpha_{static} = 0^\circ$  and  $6^\circ$ ) for several configurations of the model are presented in figures 30 and 31. During the test runs in which these data were obtained, the lift coefficient varied as indicated by the curves in figure 32. The angle-of-attack change from the wind-off static values ( $\alpha_{static} = 0^\circ$  and  $6^\circ$ ) was caused by the deflection of the support system under aerodynamic load and is indicated by the values of the actual angle of attack shown in figure 32.

Lateral control.- Most of the test results presented are for the complete model configuration consisting of the wing, fuselage, canopy, and vertical tails (figs. 34 and 35). Several tests, however, were made with the vertical tails removed (fig. 36) and these data are uncorrected for the small changes in angle of attack of the model caused by deflection of the sting-support system. The data, however, can be compared with those of figure 34 inasmuch as the lateral characteristics are not particularly sensitive to angle of attack in this range.

It is of interest to note that at low angles of attack there is an appreciable favorable yawing moment accompanying the large negative aileron deflections at all Mach numbers and that this yawing moment decreases with increase of angle of attack. A study of the data indicates that this favorable yawing moment is attributable to the side force on the vertical fins induced by the deflected aileron. The decrease in yawing moment with increase in angle of attack is probably caused by the variation with angle of attack of the incremental-drag coefficient produced by the aileron. (See fig. 35.)

Speed-brake modifications.- Tuft studies of the flow over the model with the original speed brakes (fig. 39(a)) indicated bad separation of the flow over the vertical fins, particularly the inboard surface, over

most of the Mach number range. In an effort to improve this condition, other speed-brake configurations (fig. 38) were tested. On the basis of these tuft observations (figs. 39(b), 39(c), and 39(d)), it appeared that all the modifications tested eliminated the poor flow conditions evident at the vertical fin with the original configuration.

The effect of these speed-brake configurations on the aerodynamic characteristics in pitch is presented in figure 40 for a static angle of attack of  $1.8^\circ$ . The variation of the drag increments ( $\Delta C_D$ ), produced by the various speed brakes, with Mach number is presented in figure 41. It is evident from these data that the modified wing brakes produced considerably larger drag increments than the fuselage brakes.

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TABLE I  
PARAMETERS USED IN DYNAMIC STABILITY COMPUTATIONS  

$$\left[ \epsilon = 0; \gamma = 0; C_{Y_p} = 0; C_{Y_r} = 0 \right]$$

| W/s | Altitude    | M    | $\eta$ | $C_L$ | Longitudinal |                |                     |                     |                     | Lateral   |           |           |           |               |               |               |
|-----|-------------|------|--------|-------|--------------|----------------|---------------------|---------------------|---------------------|-----------|-----------|-----------|-----------|---------------|---------------|---------------|
|     |             |      |        |       | $C_{m,q}$    | $C_{m,\alpha}$ | $\frac{k_{Y_0}}{c}$ | $\frac{k_{X_0}}{b}$ | $\frac{k_{Z_0}}{b}$ | $C_{l,p}$ | $C_{l,r}$ | $C_{n,p}$ | $C_{n,r}$ | $C_{l,\beta}$ | $C_{n,\beta}$ | $C_{Y,\beta}$ |
| 24  | Sea level   | 0.4  | 1.75   | 0.1   | -0.724       | -0.312         | 0.575               | 0.145               | 0.243               | -0.240    | 0.0228    | -0.0055   | -0.0604   | -0.0344       | 0.0573        | -0.372        |
| 24  | Sea level   | .5   | .85    | .067  | -.756        | -.313          | .575                | .145                | .243                | -.245     | .0152     | -.0037    | -.0653    | -.0172        | .0516         | -.401         |
| 24  | Sea level   | .6   | .4     | .046  | -.787        | -.314          | .575                | .145                | .243                | -.251     | .0104     | -.0026    | -.0704    | -.0086        | .0487         | -.430         |
| 24  | Sea level   | .7   | .25    | .038  | -.835        | -.322          | .575                | .145                | .243                | -.256     | .0085     | -.0024    | -.0757    | -.0057        | .0458         | -.458         |
| 24  | Sea level   | .8   | .1     | .025  | -.918        | -.341          | .575                | .145                | .243                | -.265     | .0055     | -.0018    | -.0810    | -.0040        | .0573         | -.487         |
| 24  | Sea level   | .85  | 0      | .021  | -.972        | -.3791         | .575                | .145                | .243                | -.269     | .0046     | -.0018    | -.0836    | -.0023        | .0630         | -.516         |
| 24  | Sea level   | .875 | .05    | .020  | -1.005       | -.447          | .575                | .145                | .243                | -.271     | .0044     | -.0018    | -.0848    | -.0011        | .0688         | -.516         |
| 24  | Sea level   | .9   | .1     | .019  | -1.036       | -.565          | .575                | .145                | .243                | -.275     | .0041     | -.0020    | -.0866    | 0             | .0716         | -.487         |
| 24  | Sea level   | .91  | 0      | .019  | -1.049       | -.654          | .575                | .145                | .243                | -.278     | .0041     | -.0021    | -.0881    | 0             | .0745         | -.487         |
| 24  | 40,000 feet | .6   | 4.23   | .243  | -.787        | -.314          | .575                | .145                | .243                | -.251     | .0547     | -.0139    | -.0730    | -.0688        | .0620         | -.430         |
| 24  | 40,000 feet | .7   | 2.90   | .181  | -.835        | -.322          | .575                | .145                | .243                | -.256     | .0404     | -.0113    | -.0769    | -.0487        | .0573         | -.458         |
| 24  | 40,000 feet | .8   | 1.95   | .138  | -.918        | -.341          | .575                | .145                | .243                | -.265     | .0305     | -.0101    | -.0818    | -.0372        | .0573         | -.487         |
| 24  | 40,000 feet | .85  | 1.60   | .122  | -.972        | -.379          | .575                | .145                | .243                | -.269     | .0268     | -.0102    | -.0841    | -.0258        | .0630         | -.516         |
| 24  | 40,000 feet | .875 | 1.45   | .115  | -1.005       | -.447          | .575                | .145                | .243                | -.271     | .0251     | -.0104    | -.0858    | -.0201        | .0688         | -.516         |
| 24  | 40,000 feet | .9   | 1.35   | .110  | -1.036       | -.565          | .575                | .145                | .243                | -.275     | .0239     | -.0113    | -.0879    | -.0143        | .0716         | -.487         |
| 24  | 40,000 feet | .91  | 1.40   | .107  | -1.049       | -.654          | .575                | .145                | .243                | -.278     | .0231     | -.0120    | -.0891    | -.0115        | .0745         | -.487         |
| 34  | Sea level   | .4   | 2.60   | .142  | -.724        | -.312          | .575                | .157                | .240                | -.240     | .0324     | -.0078    | -.0605    | -.0487        | .0573         | -.372         |
| 34  | Sea level   | .5   | 1.55   | .092  | -.756        | -.313          | .575                | .157                | .240                | -.245     | .0209     | -.0051    | -.0654    | -.0344        | .0516         | -.401         |
| 34  | Sea level   | .6   | .90    | .064  | -.787        | -.314          | .575                | .157                | .240                | -.251     | .0144     | -.0036    | -.0704    | -.0260        | .0487         | -.430         |
| 34  | Sea level   | .7   | .50    | .046  | -.835        | -.322          | .575                | .157                | .240                | -.256     | .0103     | -.0029    | -.0757    | -.0230        | .0458         | -.458         |
| 34  | Sea level   | .8   | .30    | .035  | -.918        | -.341          | .575                | .157                | .240                | -.265     | .0077     | -.0026    | -.0810    | -.0171        | .0573         | -.487         |
| 34  | Sea level   | .85  | .2     | .030  | -.972        | -.379          | .575                | .157                | .240                | -.269     | .0066     | -.0025    | -.0836    | -.0115        | .0630         | -.516         |
| 34  | Sea level   | .875 | .1     | .028  | -1.005       | -.447          | .575                | .157                | .240                | -.271     | .0061     | -.0025    | -.0848    | -.0056        | .0688         | -.516         |
| 34  | Sea level   | .9   | .05    | .026  | -1.036       | -.565          | .575                | .157                | .240                | -.275     | .0056     | -.0027    | -.0869    | -.0029        | .0716         | -.487         |
| 34  | Sea level   | .91  | .1     | .026  | -1.049       | -.654          | .575                | .157                | .240                | -.278     | .0056     | -.0029    | -.0881    | -.0029        | .0745         | -.487         |
| 34  | 40,000 feet | .6   | 6.20   | .345  | -.787        | -.314          | .575                | .157                | .240                | -.251     | .0776     | -.0197    | -.0759    | -.1003        | .0620         | -.430         |
| 34  | 40,000 feet | .7   | 4.30   | .253  | -.835        | -.322          | .575                | .157                | .240                | -.256     | .0564     | -.0158    | -.0784    | -.0688        | .0573         | -.458         |
| 34  | 40,000 feet | .8   | 2.85   | .195  | -.918        | -.341          | .575                | .157                | .240                | -.265     | .0431     | -.0143    | -.0829    | -.0516        | .0573         | -.487         |
| 34  | 40,000 feet | .85  | 2.35   | .173  | -.972        | -.379          | .575                | .157                | .240                | -.269     | .0381     | -.0144    | -.0848    | -.0372        | .0630         | -.516         |
| 34  | 40,000 feet | .875 | 2.14   | .163  | -1.005       | -.447          | .575                | .157                | .240                | -.271     | .0355     | -.0148    | -.0880    | -.0287        | .0688         | -.516         |
| 34  | 40,000 feet | .9   | 2.00   | .155  | -1.036       | -.565          | .575                | .157                | .240                | -.275     | .0336     | -.0158    | -.0890    | -.0201        | .0716         | -.487         |
| 34  | 40,000 feet | .91  | 2.00   | .151  | -1.049       | -.654          | .575                | .157                | .240                | -.278     | .0326     | -.0170    | -.0902    | -.0172        | .0745         | -.487         |

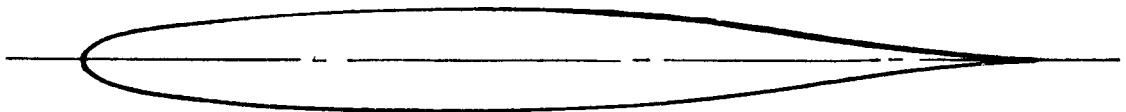
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TABLE II. COORDINATES OF SYMMETRICAL  
AIRFOIL SECTION

[All dimensions in percent of wing chord parallel  
to plane of symmetry of wing]

| Station  | Upper- and lower-surface<br>ordinate |
|----------|--------------------------------------|
| 0        | 0                                    |
| .5871    | 1.0958                               |
| .8803    | 1.3226                               |
| 1.4661   | 1.6687                               |
| 2.9264   | 2.2597                               |
| 5.8297   | 2.9981                               |
| 8.7103   | 3.4923                               |
| 11.5680  | 3.8626                               |
| 17.2154  | 4.3929                               |
| 22.7728  | 4.7516                               |
| 28.2409  | 4.9951                               |
| 33.6203  | 5.1488                               |
| 38.9118  | 5.2322                               |
| 44.1160  | 5.2200                               |
| 49.2336  | 5.1300                               |
| 54.2654  | 4.9088                               |
| 59.2118  | 4.5506                               |
| 64.0736  | 4.0784                               |
| 68.9587  | 3.5320                               |
| 73.5461  | 2.9550                               |
| 78.1583  | 2.3821                               |
| 82.6881  | 1.8395                               |
| 87.1366  | 1.3383                               |
| 91.5043  | .8757                                |
| 95.7921  | .4408                                |
| 100.0000 | .0206                                |

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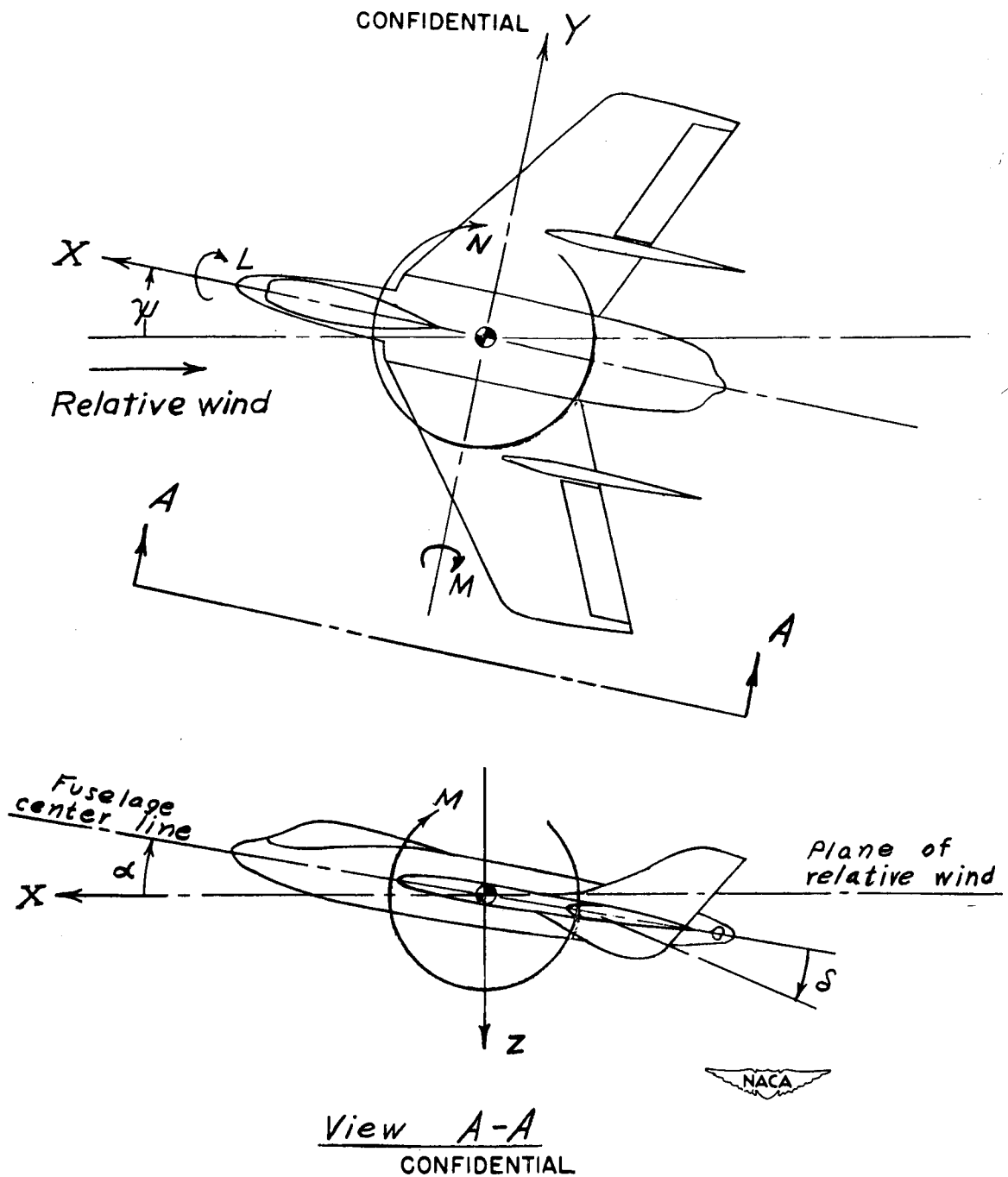


Figure 1.- System of axes and control-surface deflections. Positive values of forces, moments, and angles are indicated by arrows.



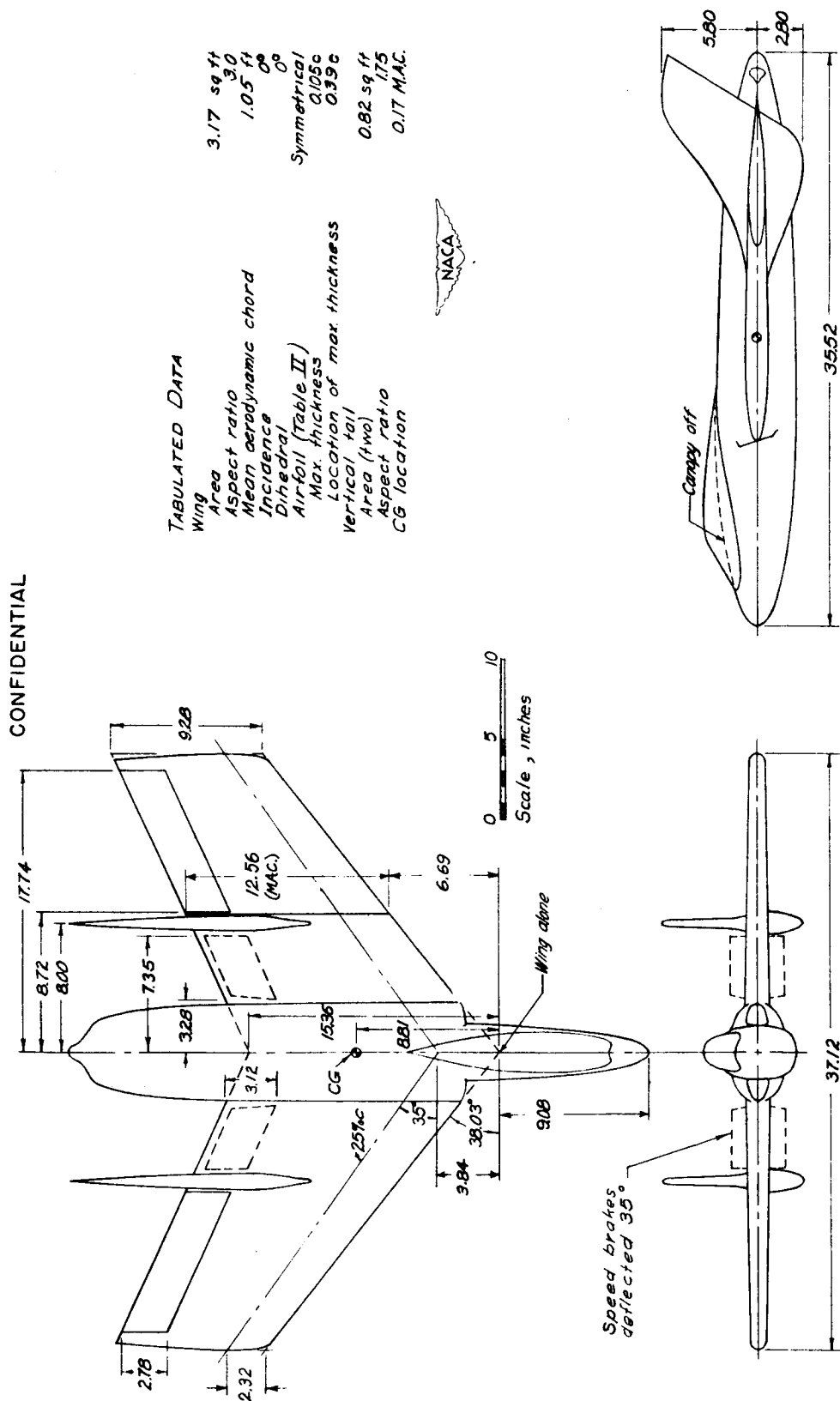


Figure 2.- General arrangement of the test model.

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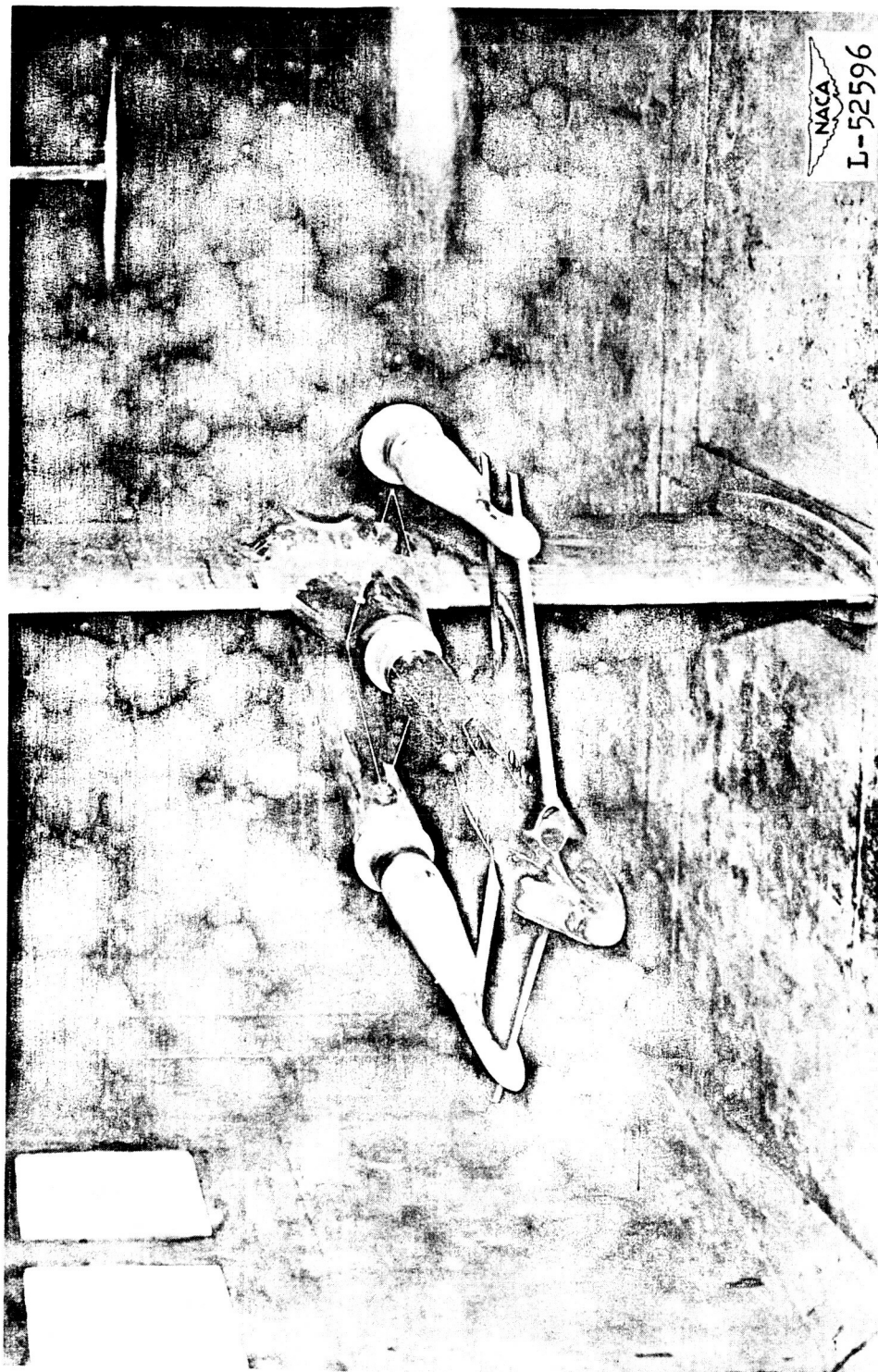


(a) Model mounted on the center sting.

Figure 3.- Photographs of the test model.

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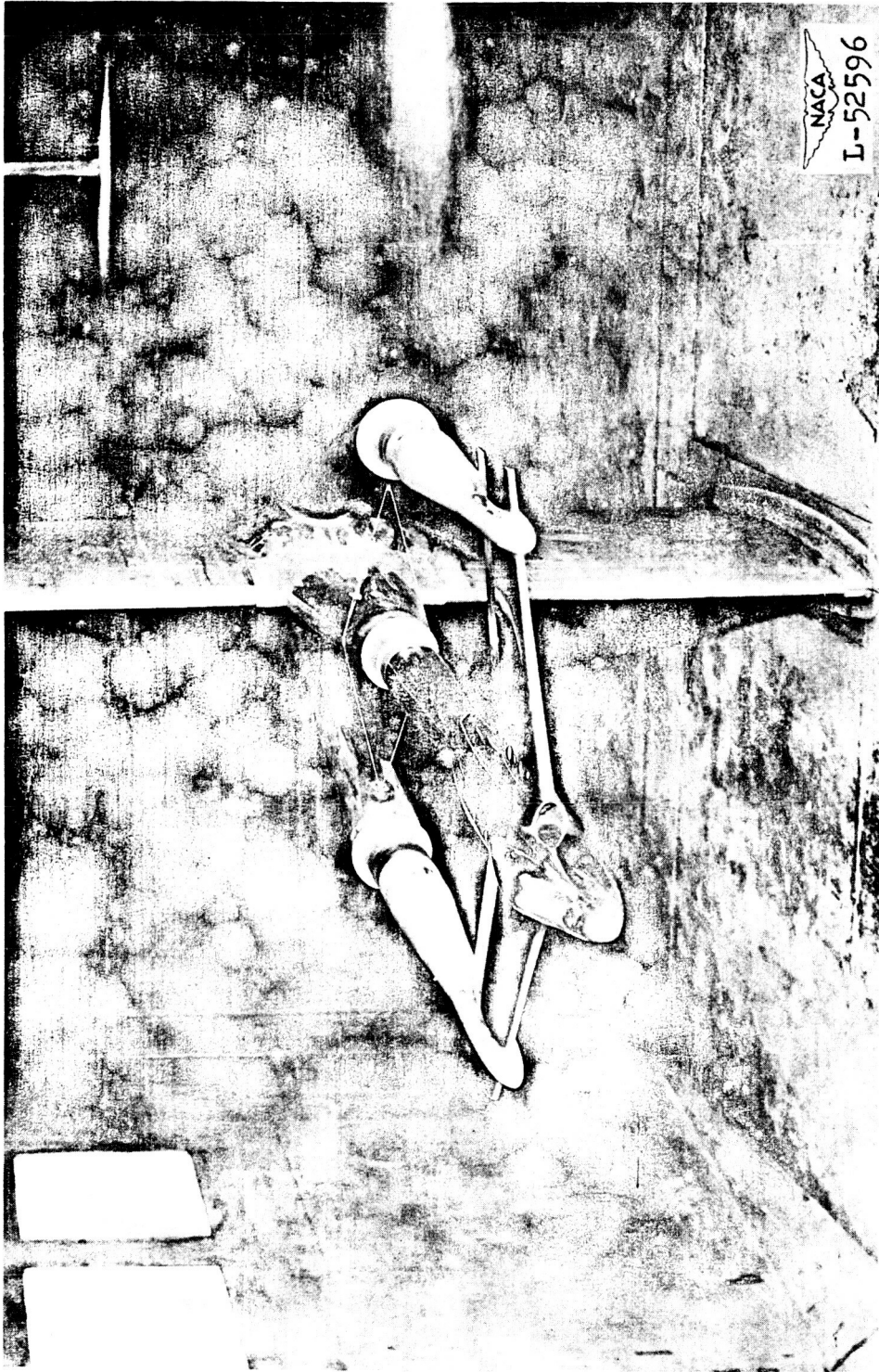


(b) Model mounted on the tare stings with the center sting in place.

Figure 3.- Concluded.

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(b) Model mounted on the tare stings with the center sting in place.

Figure 3.- Concluded.

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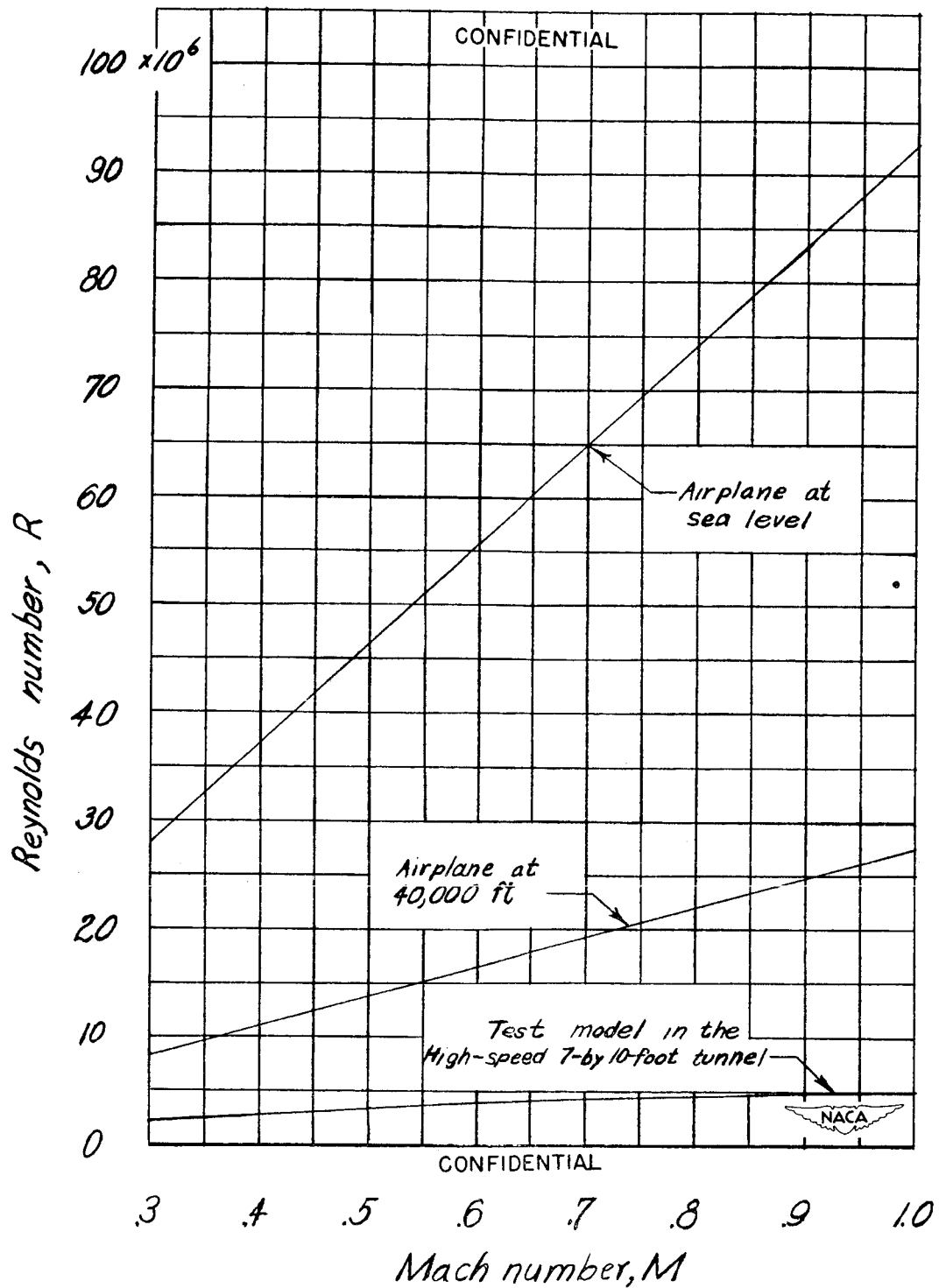


Figure 4.— Variation of Reynolds number with Mach number for flight of the assumed airplane at two altitudes and for wind-tunnel tests data. Based on the model M.A.C. of 1.046 feet and an airplane M.A.C. of 13 feet.

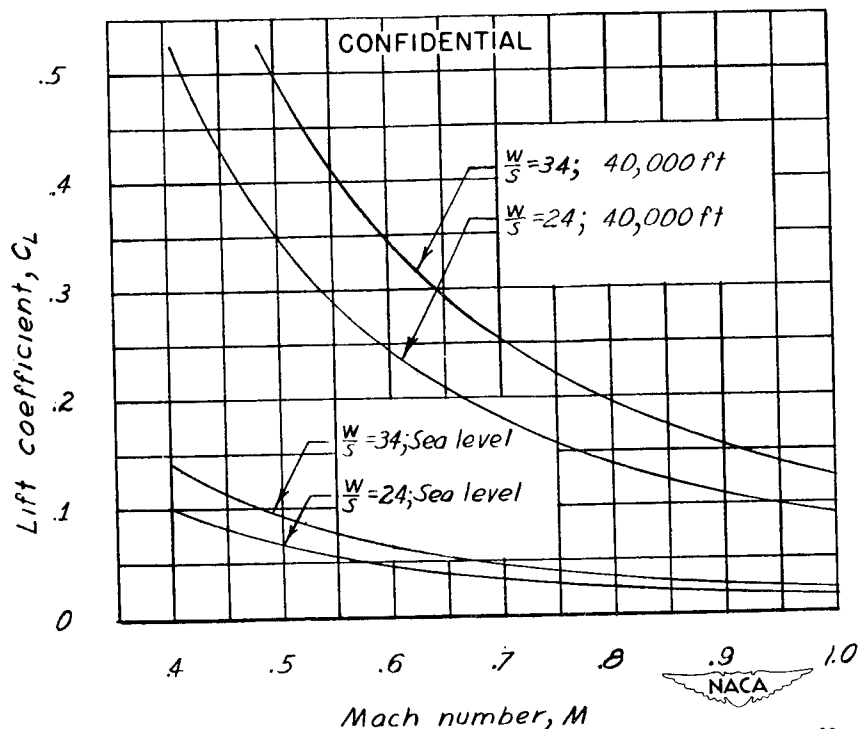


Figure 5.-Variation with Mach number of the lift coefficient required for level flight at various altitudes and wing loadings.

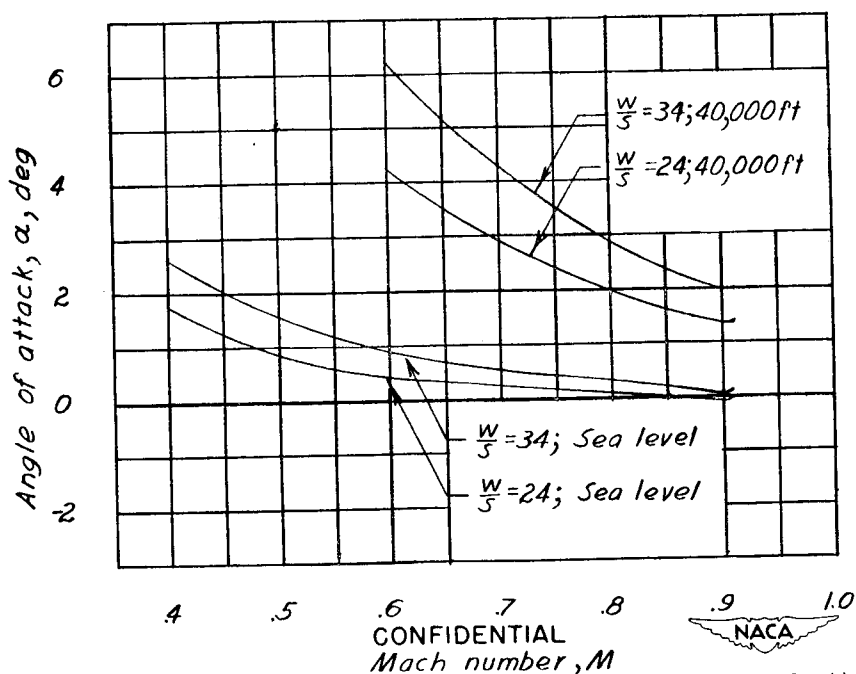


Figure 6.-Variation with Mach number of the angle of attack required for level flight at various altitudes and wing loadings.

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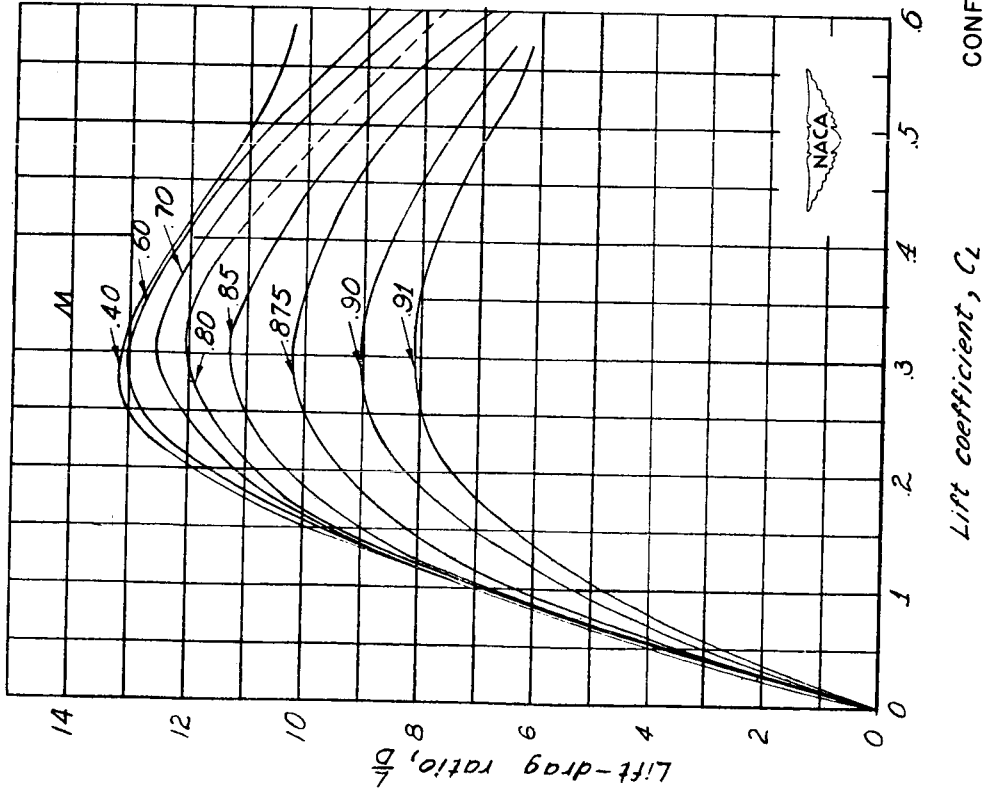


Figure 7.- Variation with lift coefficient of the lift-drag ratio at various Mach numbers.  $\delta_0 = 0^\circ$ .

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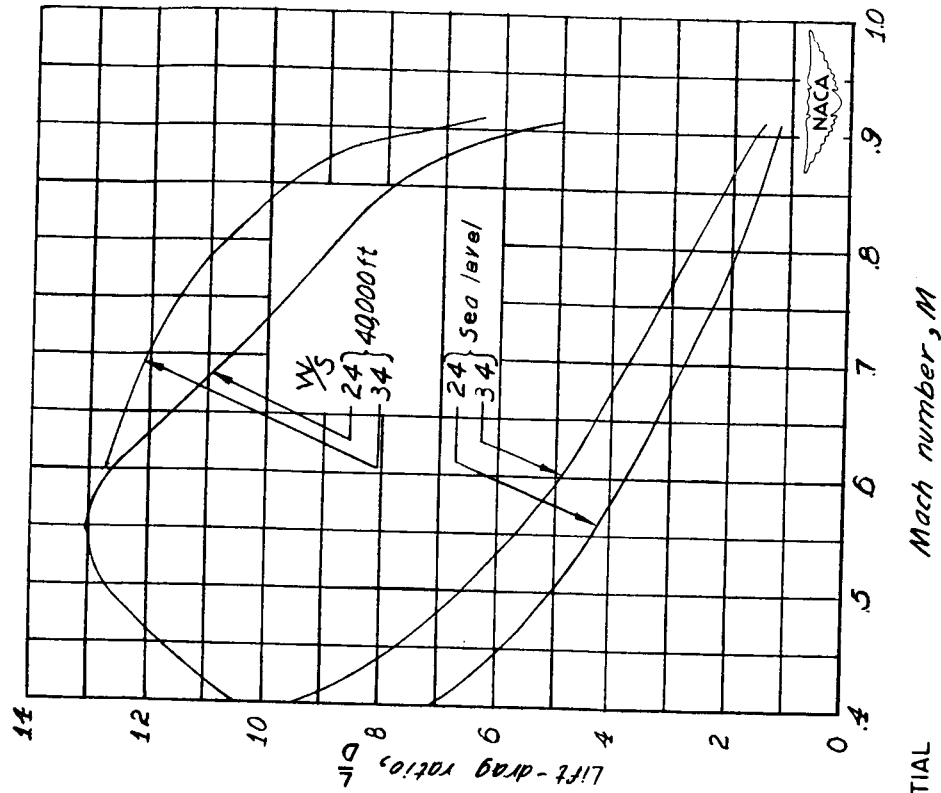


Figure 8.- Variation with Mach number of the lift-drag ratio in level flight at various altitudes and wing loadings. (Trimmed)

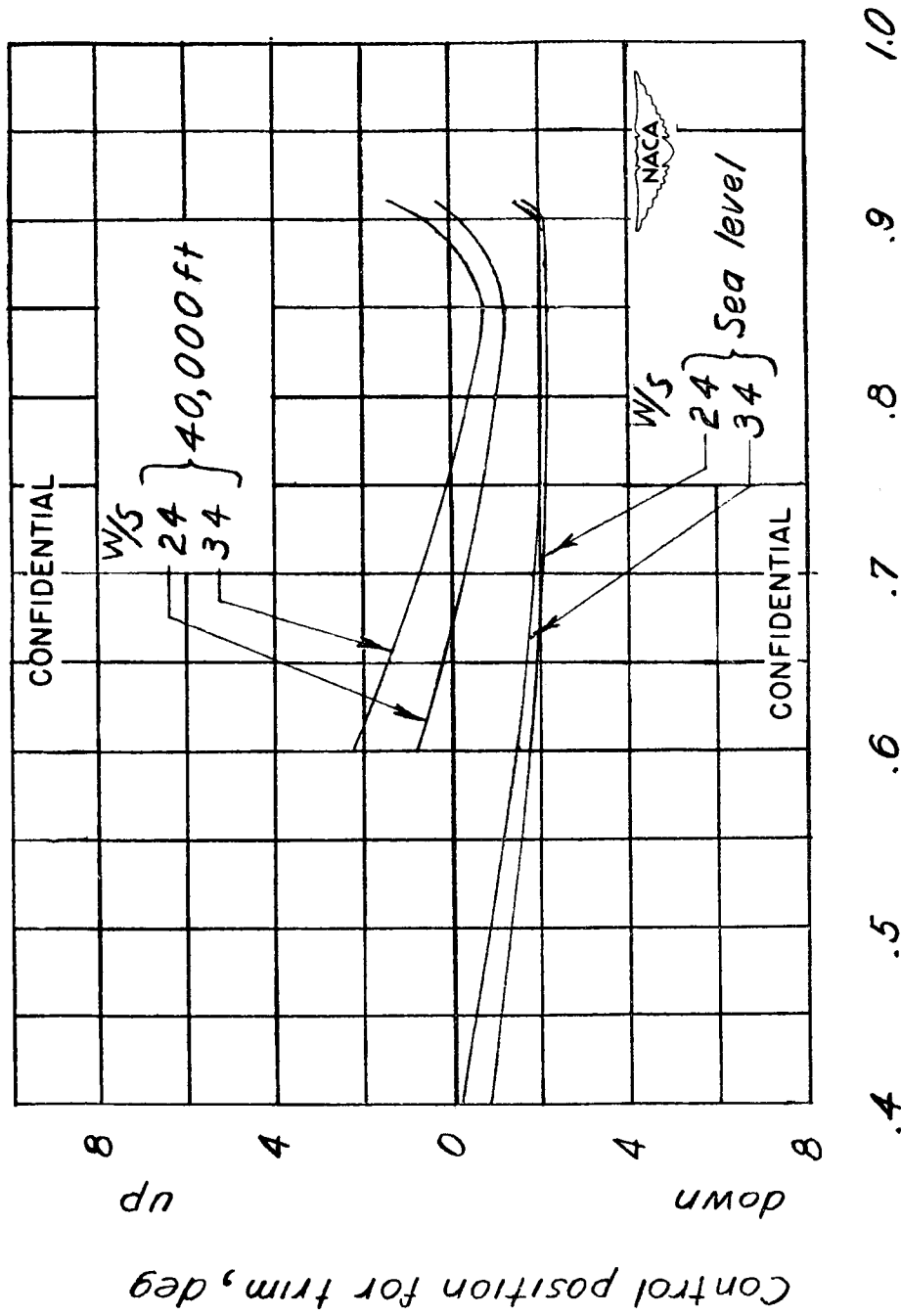
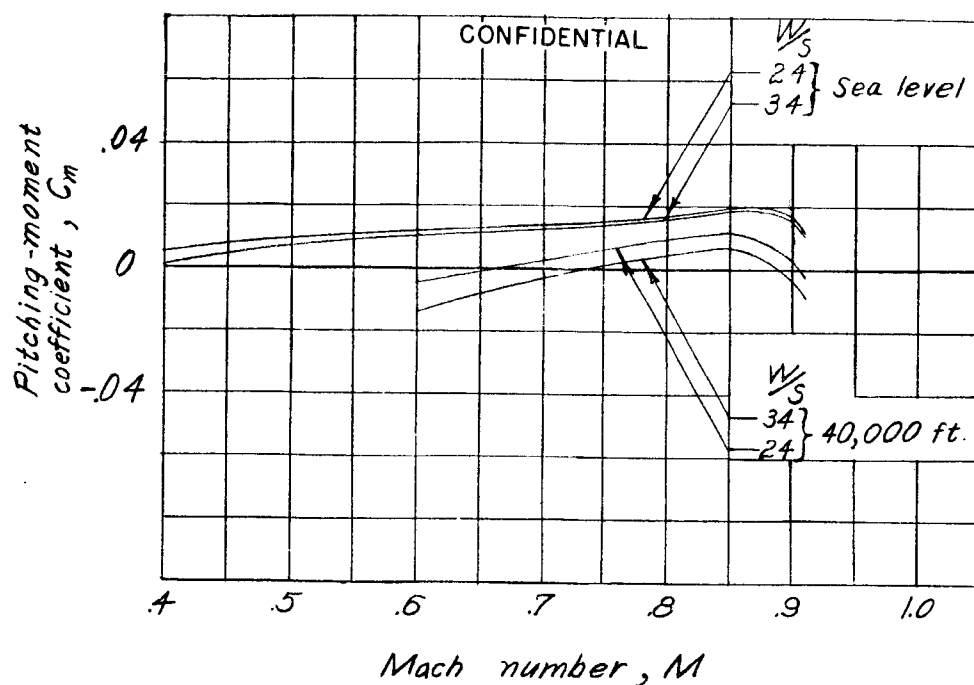
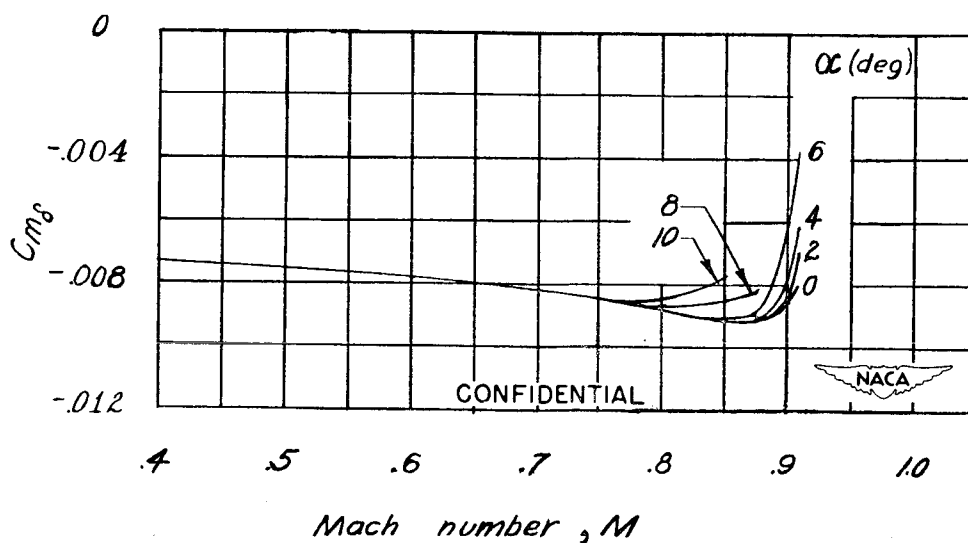


Figure 9. - Variation with Mach number of the control position required for trim in level flight at various altitudes and wing loadings.





(a) Variation with Mach number of the out-of-trim pitching moment of the model with zero control deflection.



(b) Variation of the control-effectiveness parameter ( $C_{m\delta}$ ) with Mach number.

Figure 10.— Variation of the out-of-trim pitching moment and the control effectiveness with Mach number.

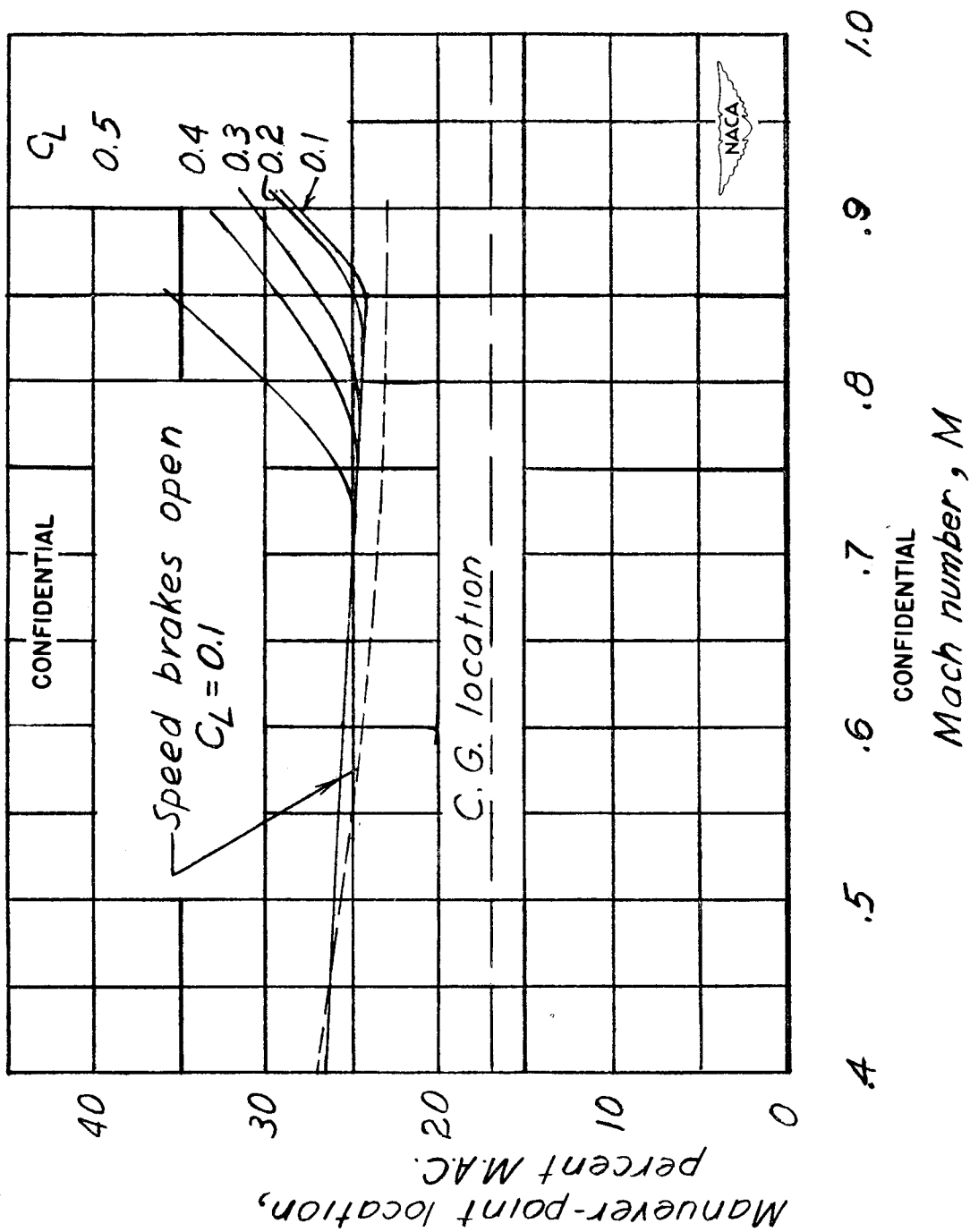
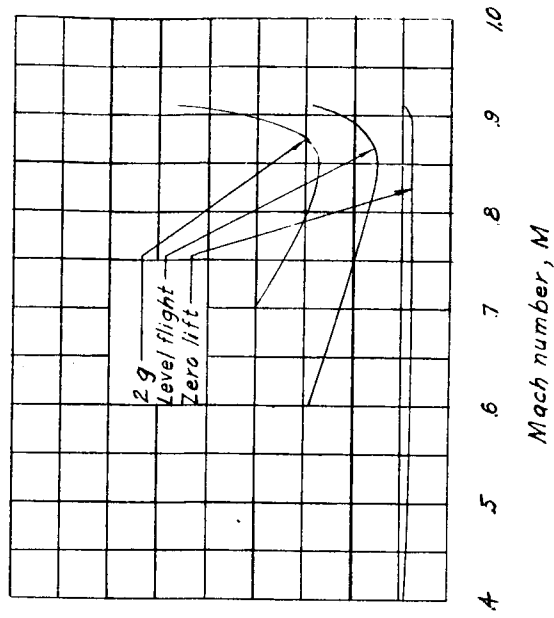
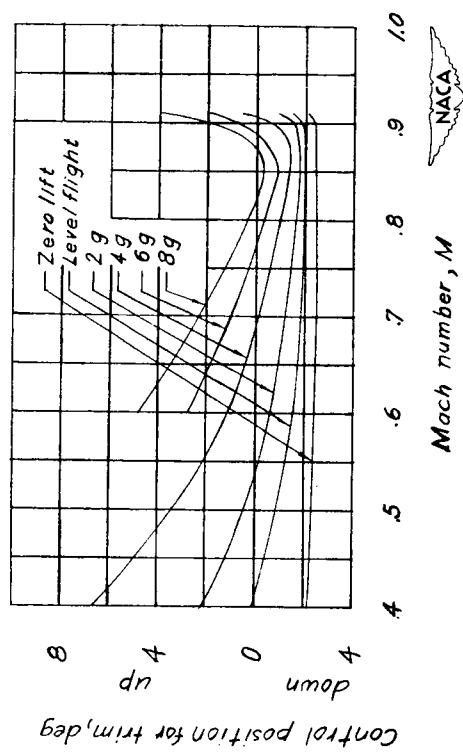


Figure 11.- Variation with Mach number of the maneuver-point location.

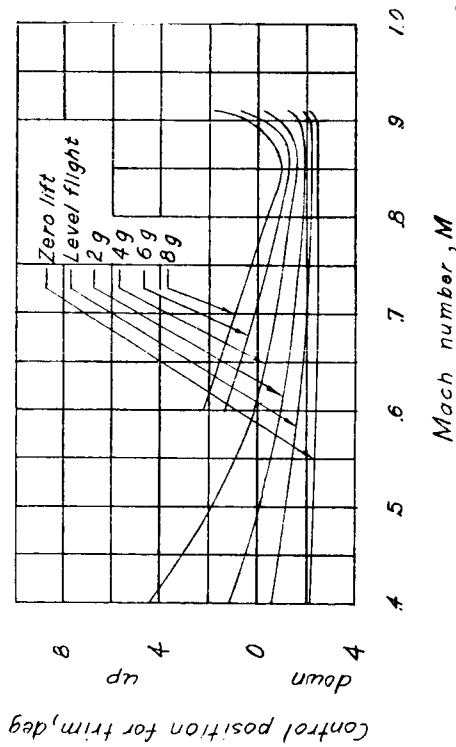
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(b) 40,000 ft.,  $\frac{W}{S} = 3.4$ .



(c) Sea Level,  $\frac{W}{S} = 2.4$ .



(d) Sea Level,  $\frac{W}{S} = 3.4$ .

Figure 12.- Variation with Mach number of the control position required for trim in level and accelerated flight at various altitudes and wing loadings.



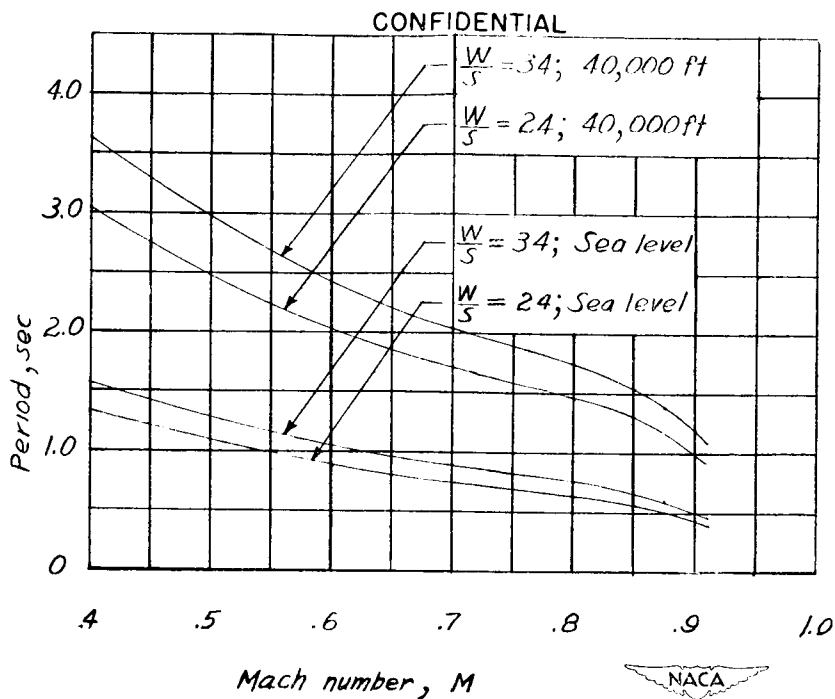


Figure 13.- Variation with Mach number of the period of the short-period longitudinal oscillation.

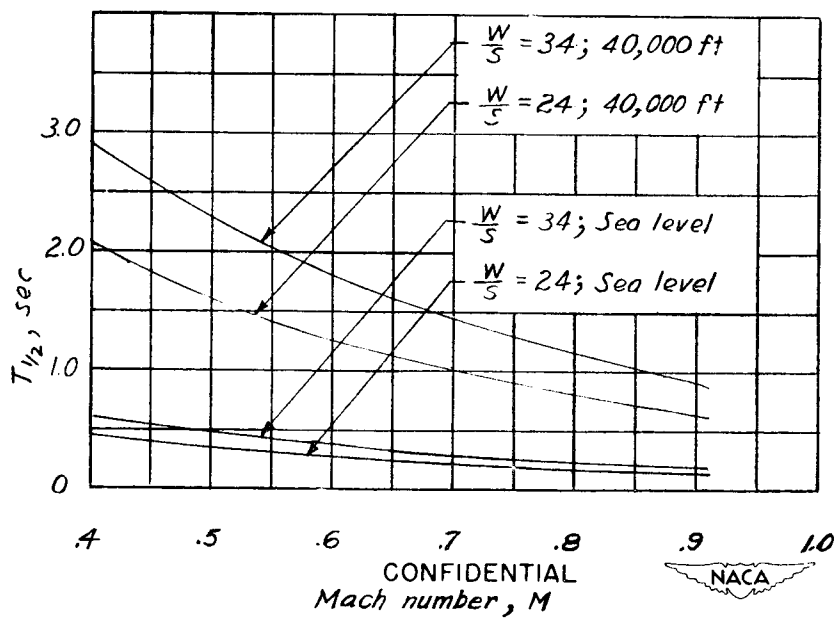


Figure 14.- Variation with Mach number of the time required for the short-period longitudinal oscillation to damp to one-half amplitude.

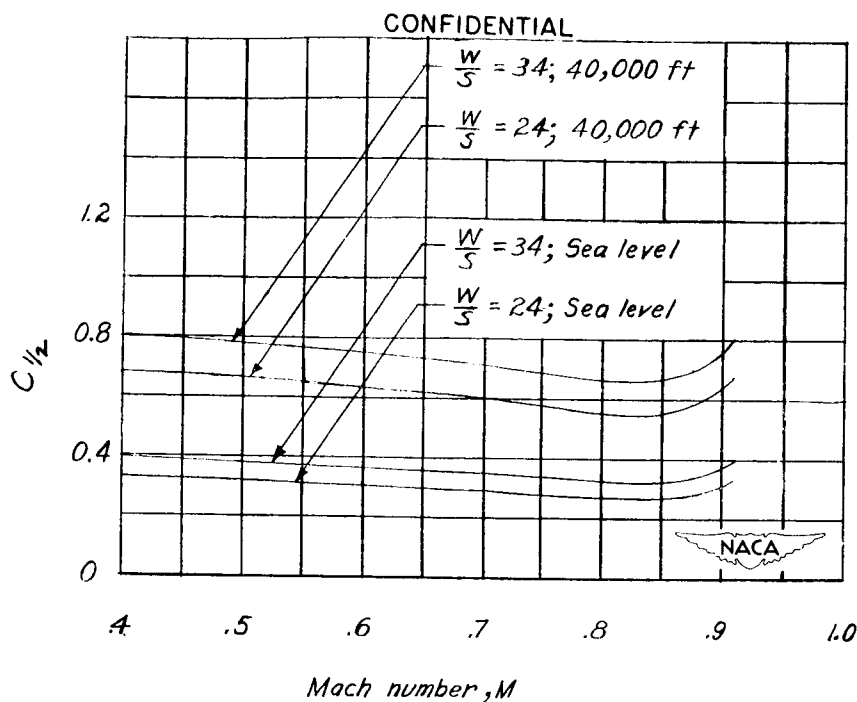


Figure 15.- Variation with Mach number of the number of cycles required for the short-period longitudinal oscillation to damp to one-half amplitude.

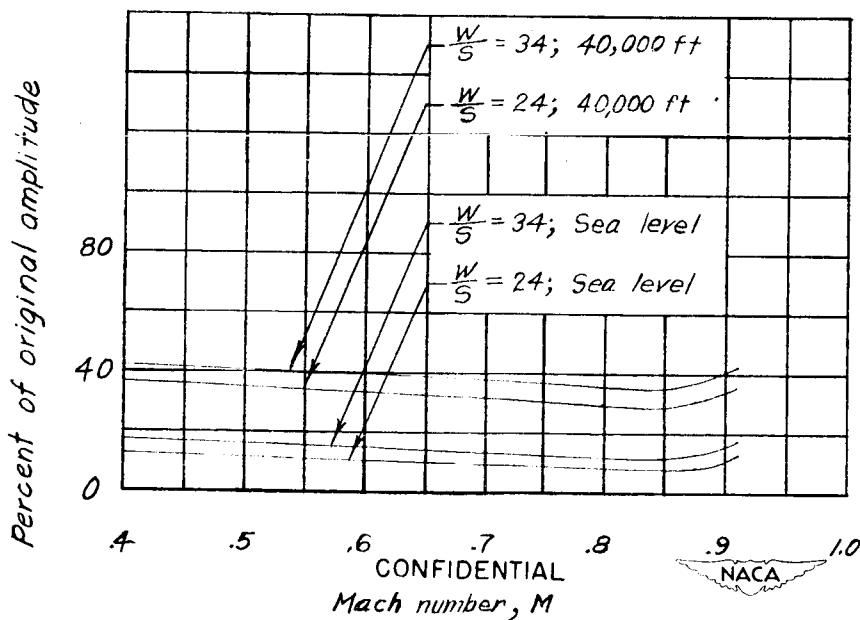


Figure 16.- Variation with Mach number of the amplitude of the short-period longitudinal oscillation after one cycle in percent of the original amplitude.

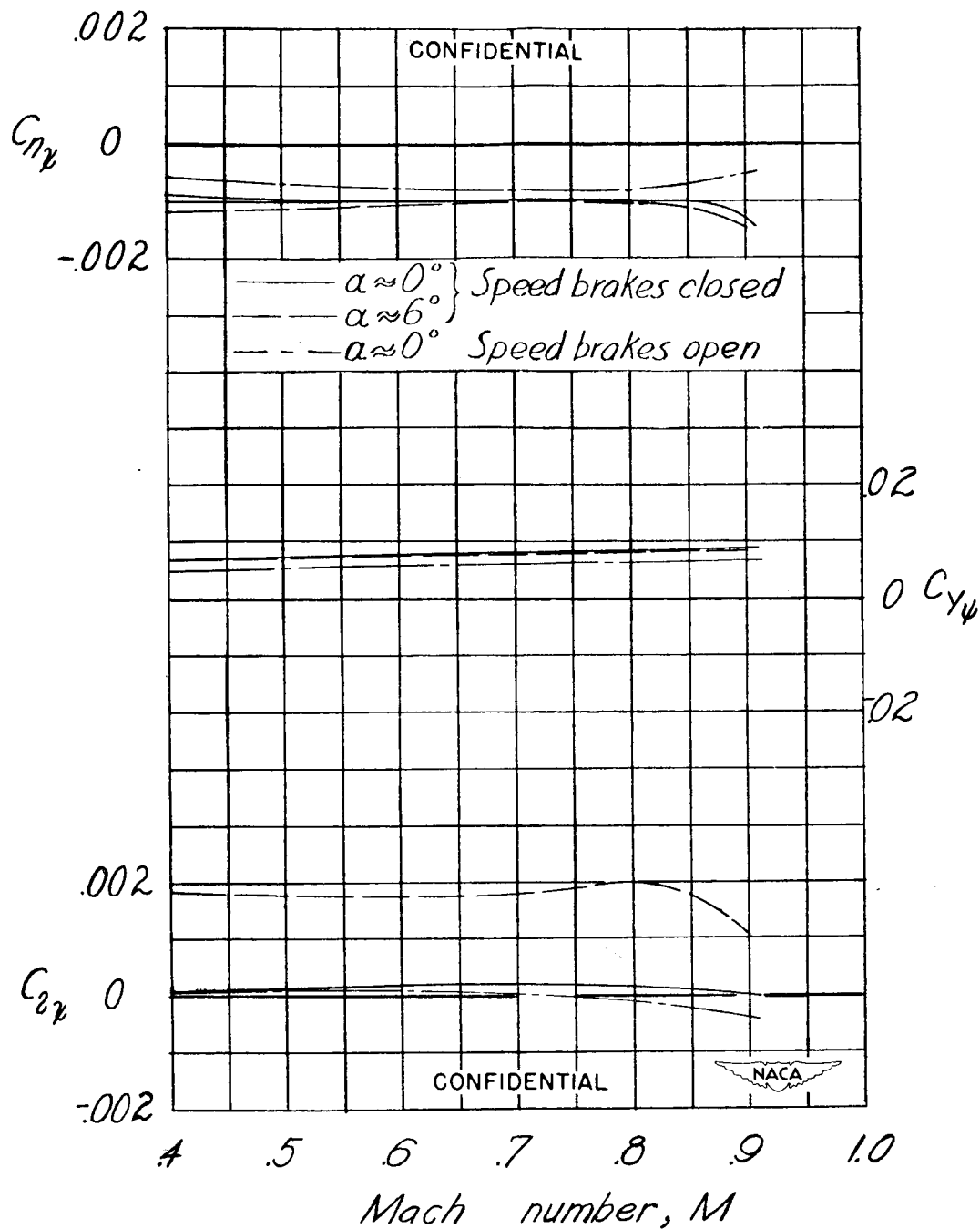


Figure 17.- Variation with Mach number of the lateral stability characteristics of the test model.

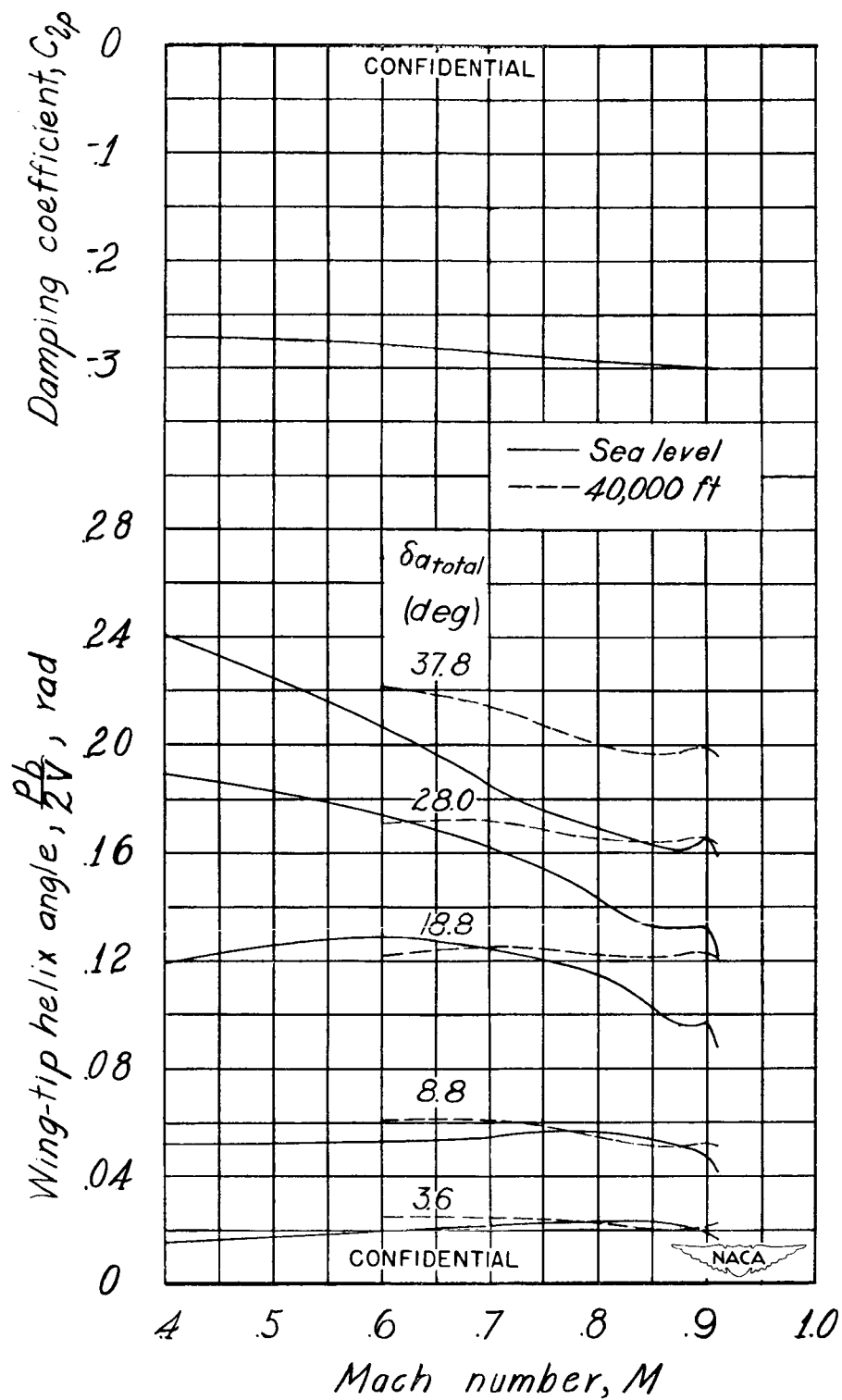


Figure 18.- Variation of the wing-tip helix angle with Mach number for various total aileron deflections.

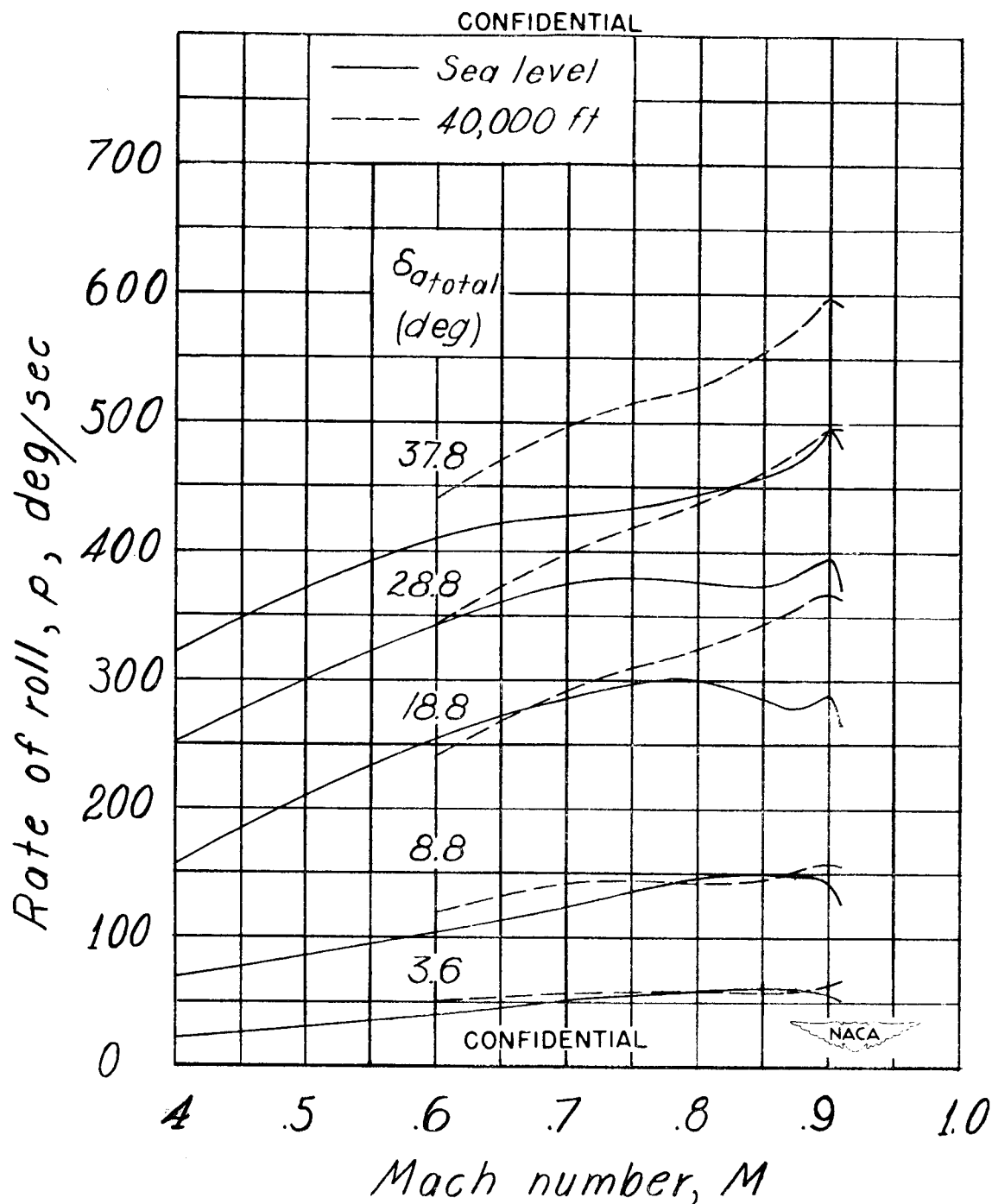


Figure 19.- Variation with Mach number of the rate of roll for various total aileron deflections.



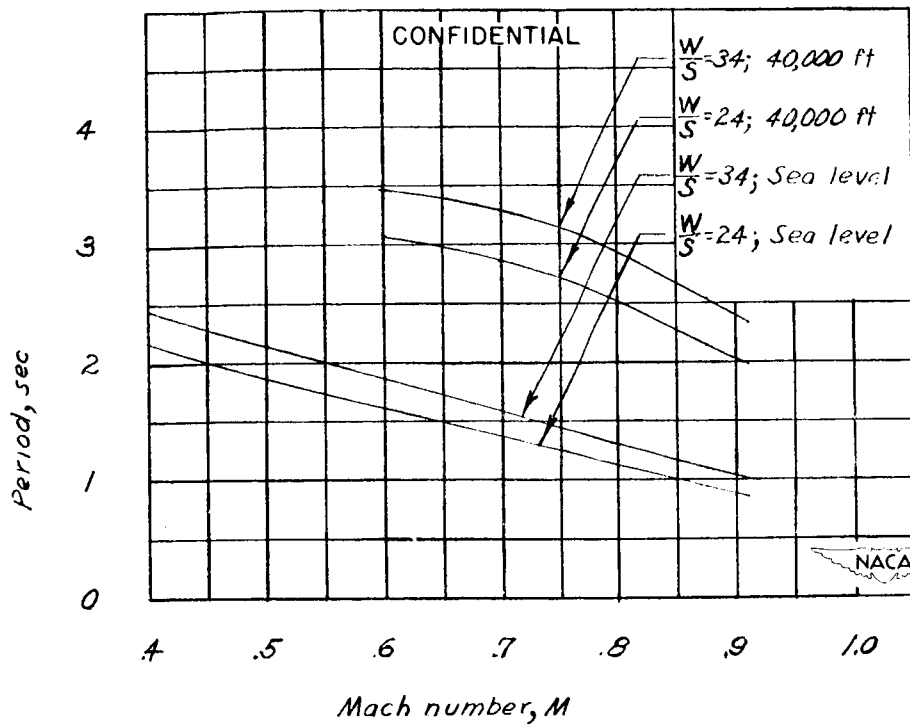


Figure 20.- Variation with Mach number of the period of the lateral oscillation.

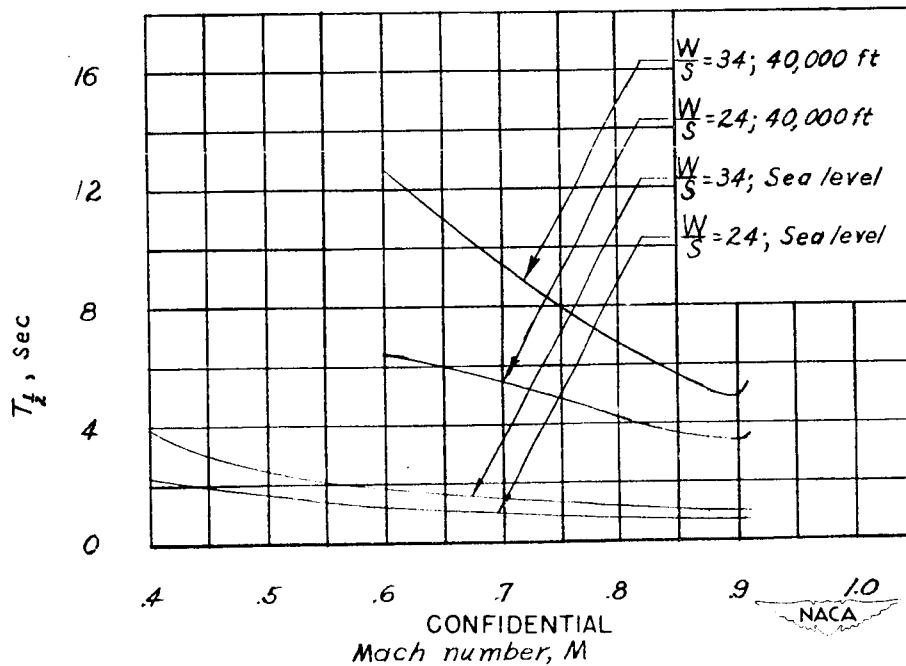


Figure 21.- Variation with Mach number of the time required for the lateral oscillation to damp to one-half amplitude.

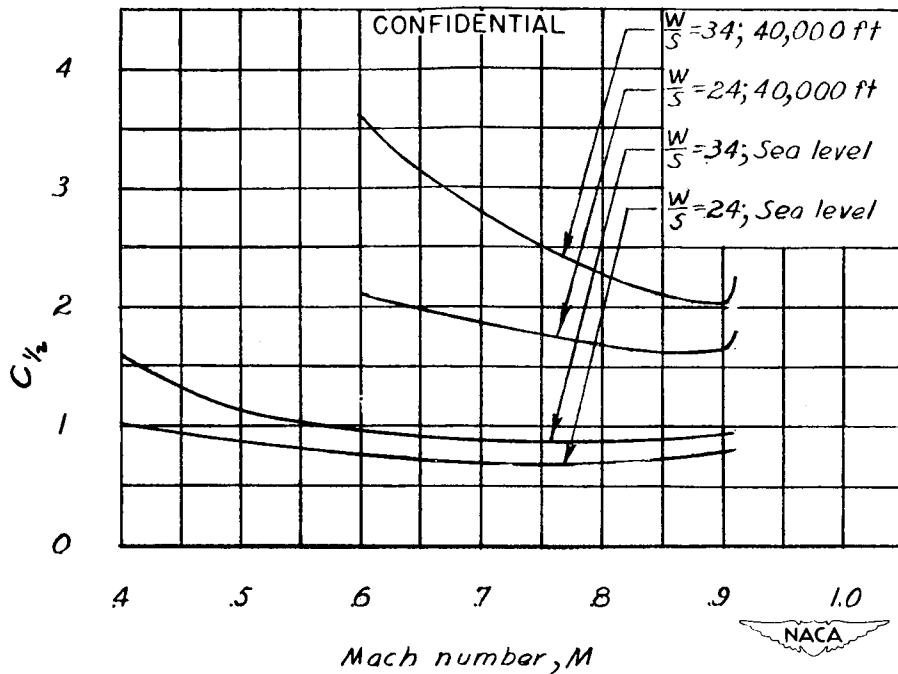


Figure 22.- Variation with Mach number of the number of cycles required for the lateral oscillation to damp to one-half amplitude.

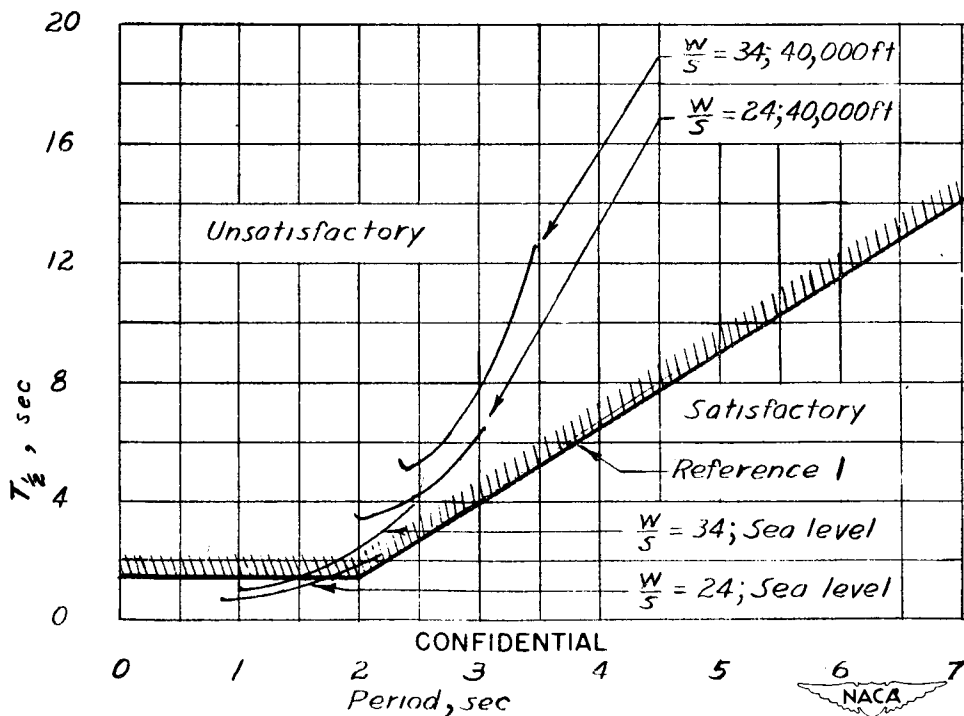


Figure 23.- Variation with Period of the time required for the lateral oscillation to damp to one-half amplitude.

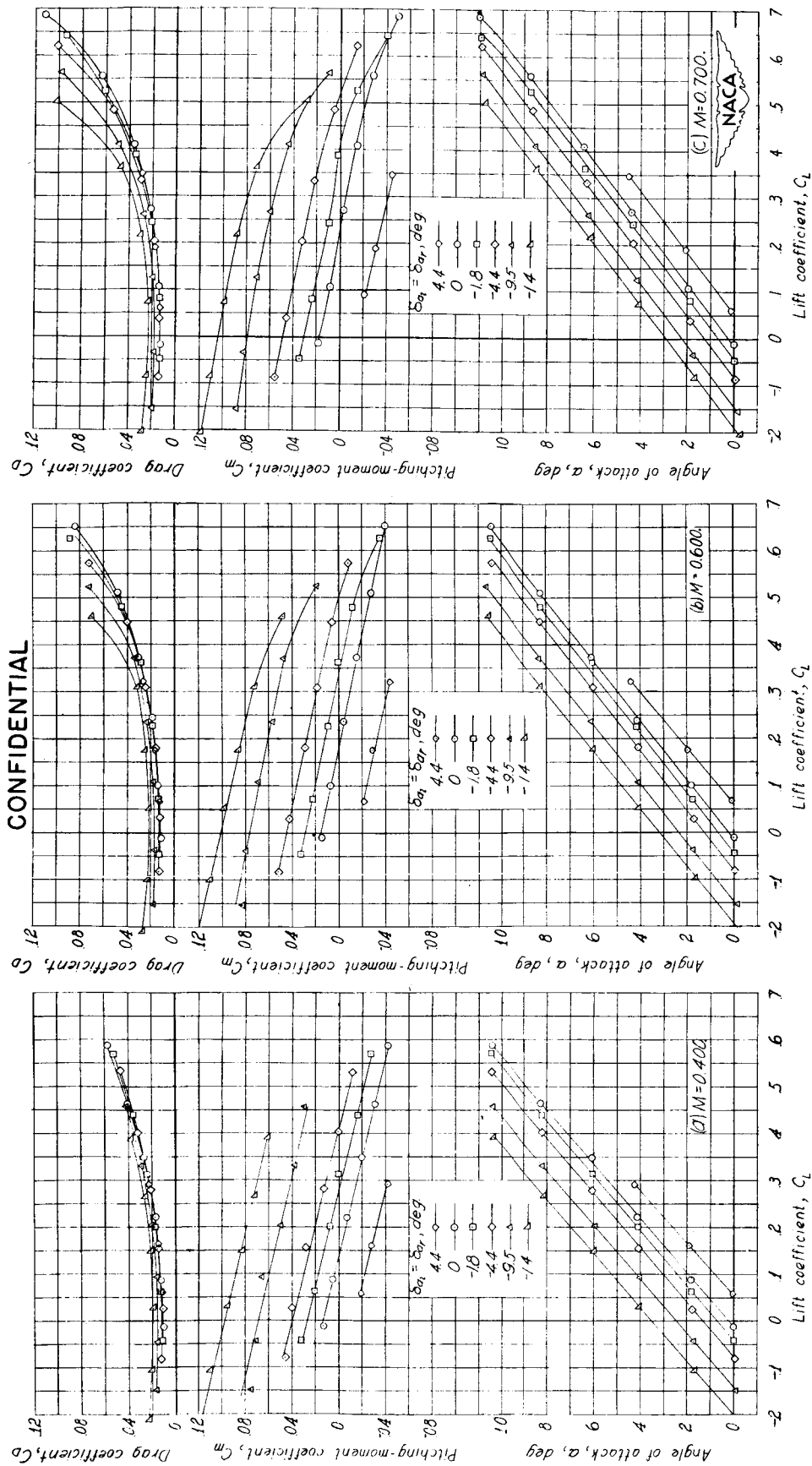


Figure 24. - Effect of control deflection on the aerodynamic characteristics in pitch of the test model.

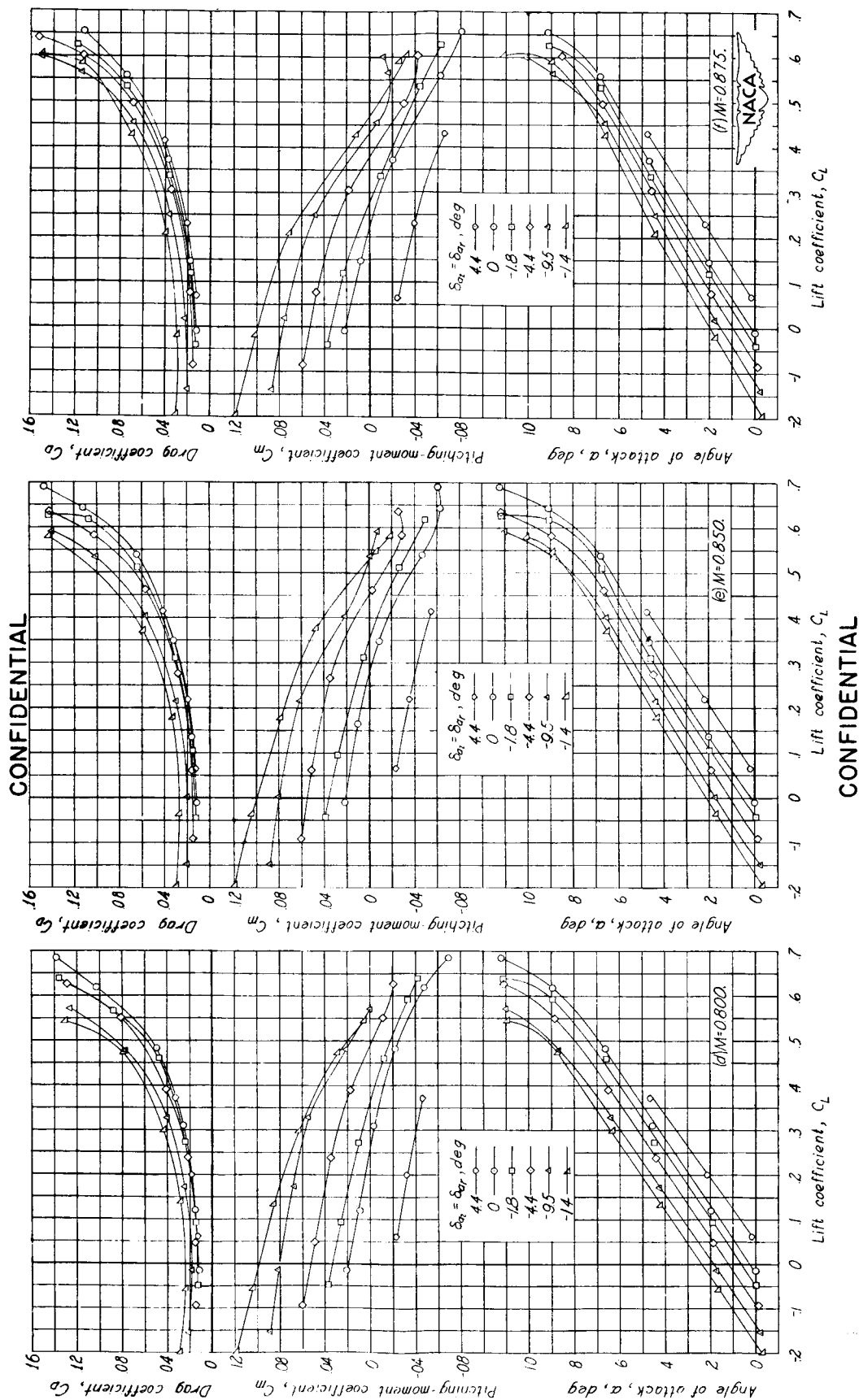


Figure 2.4.—Continued.

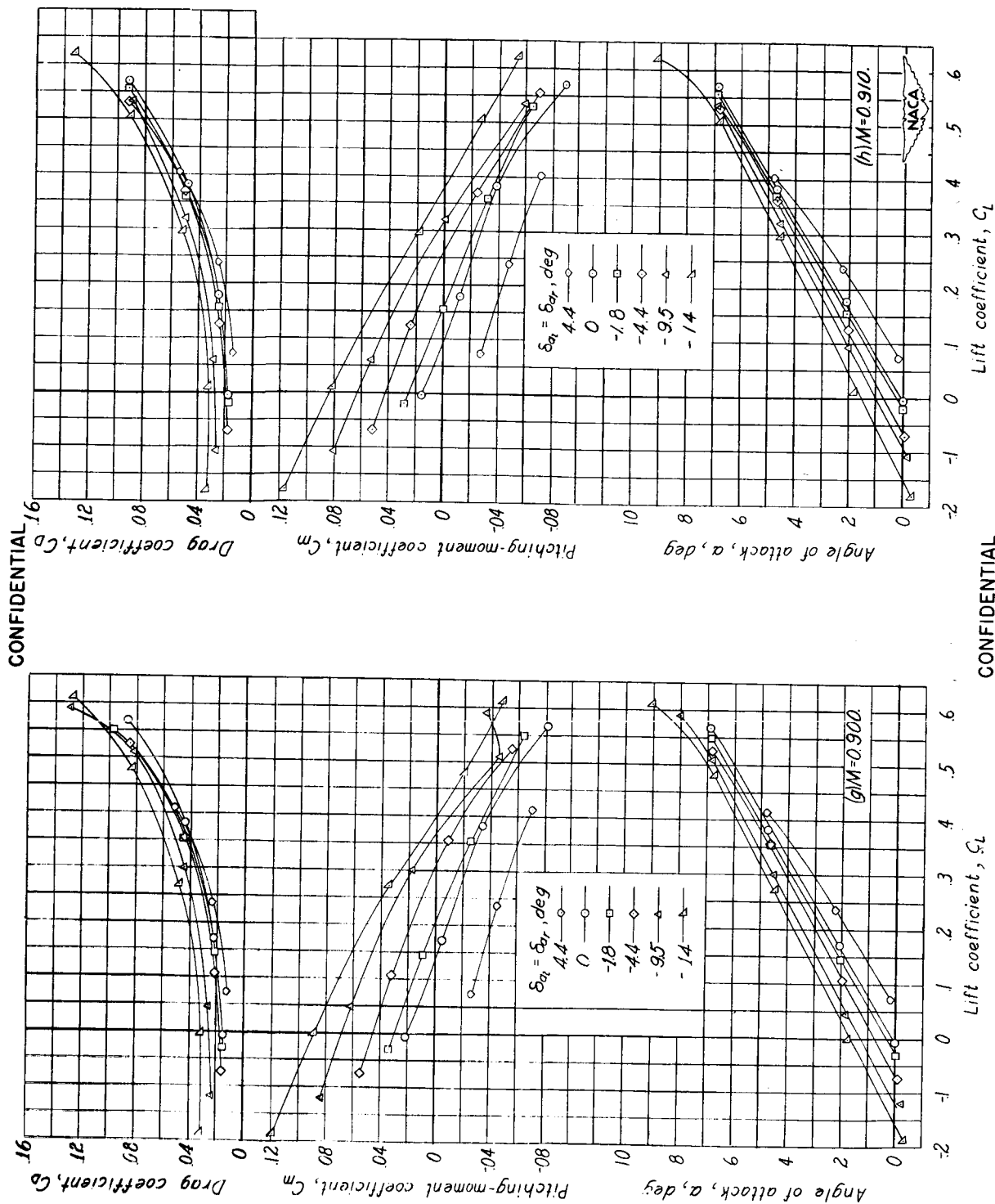


Figure 24- Concluded.

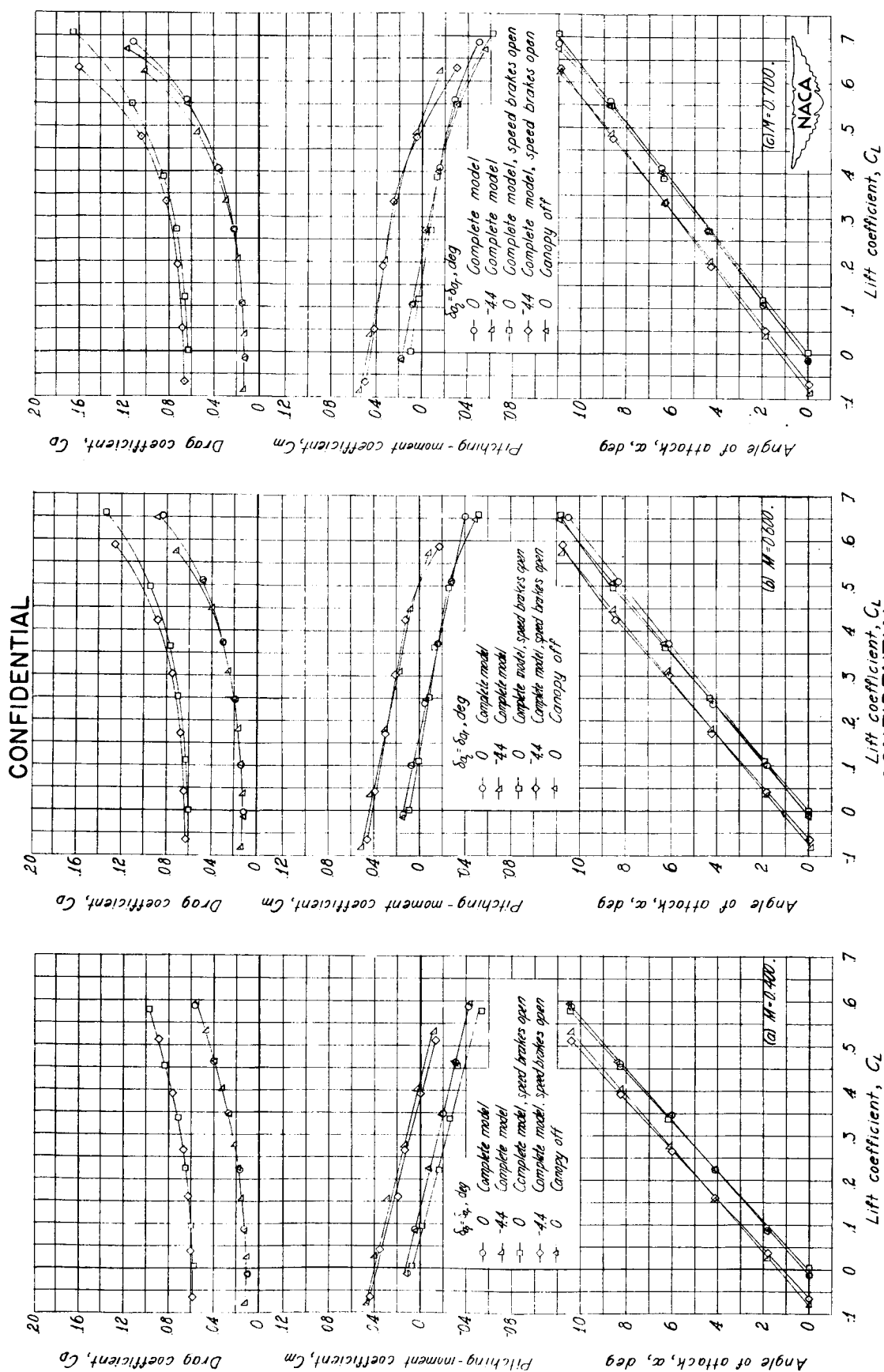


Figure 25. - Effect of speed brakes and canopy on the aerodynamic characteristics of the test model.

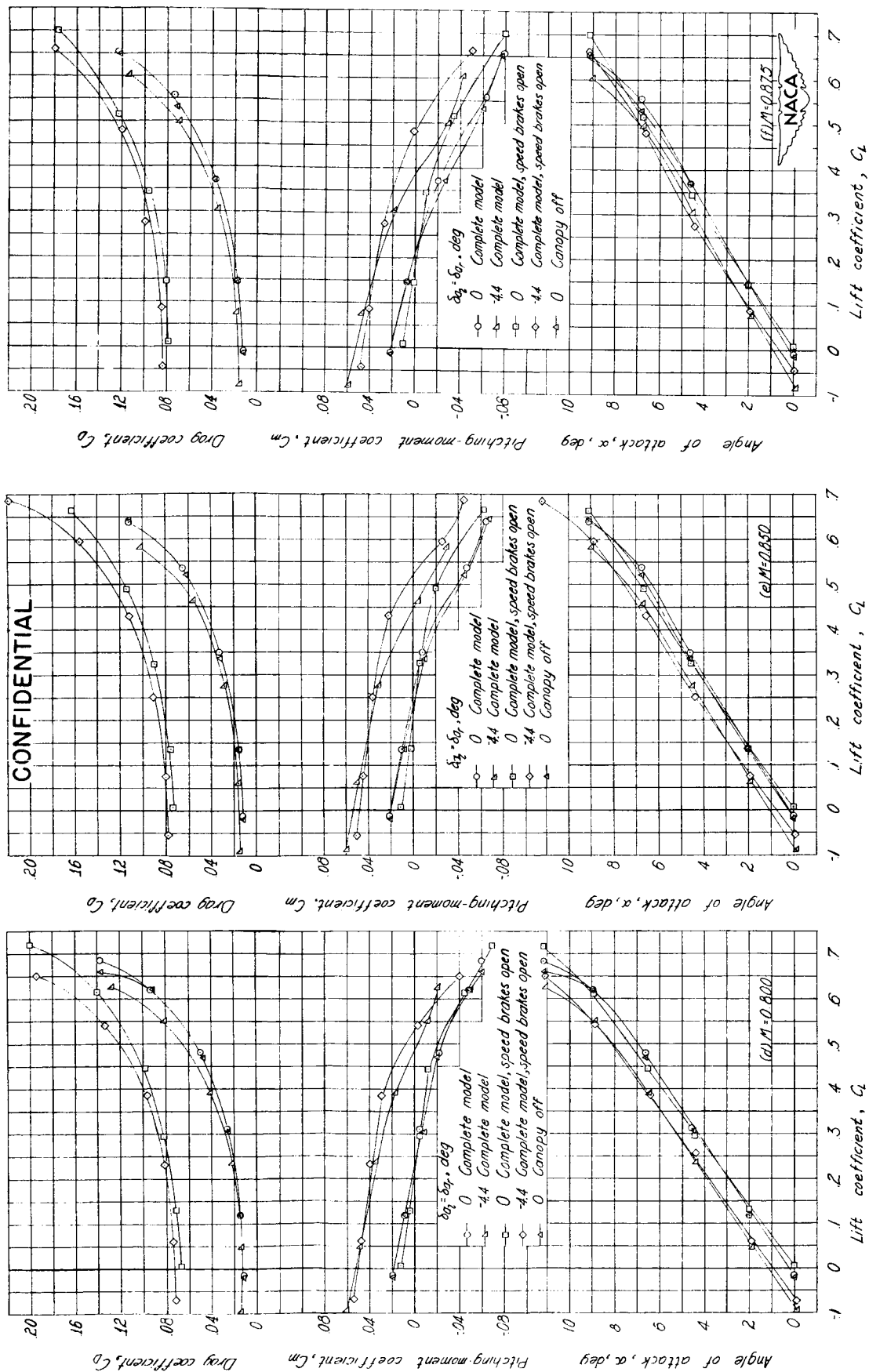
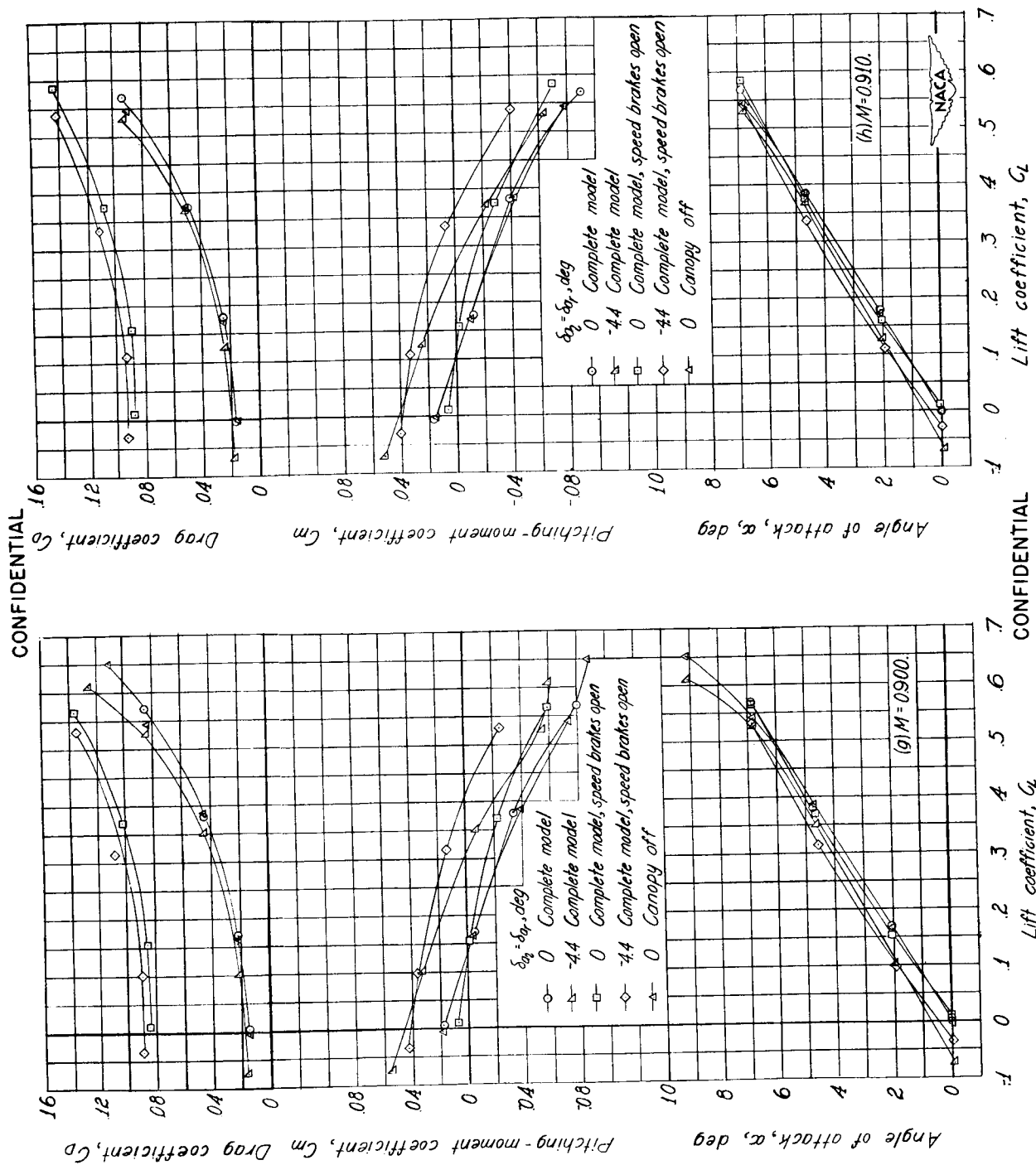


Figure 25.-Continued.





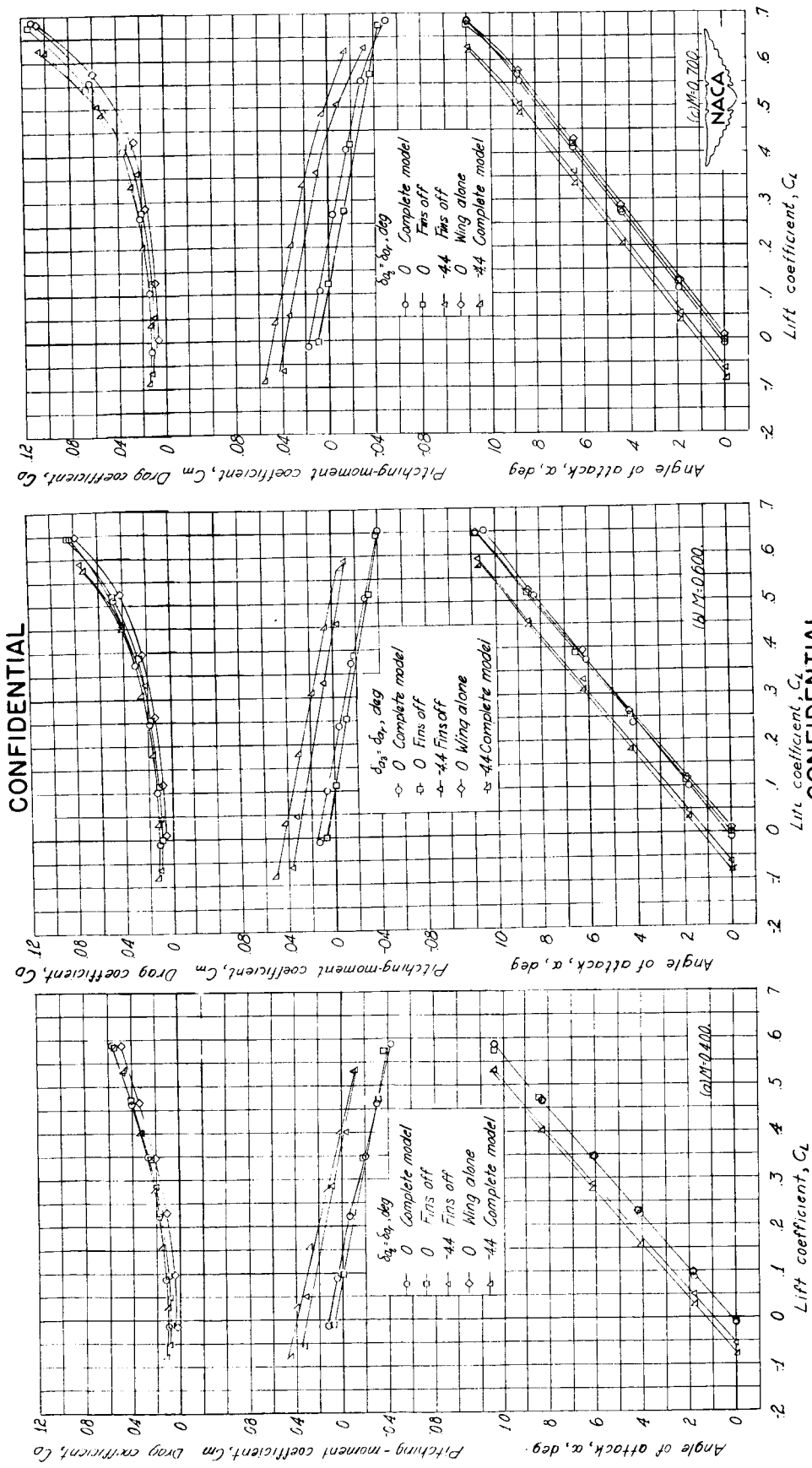
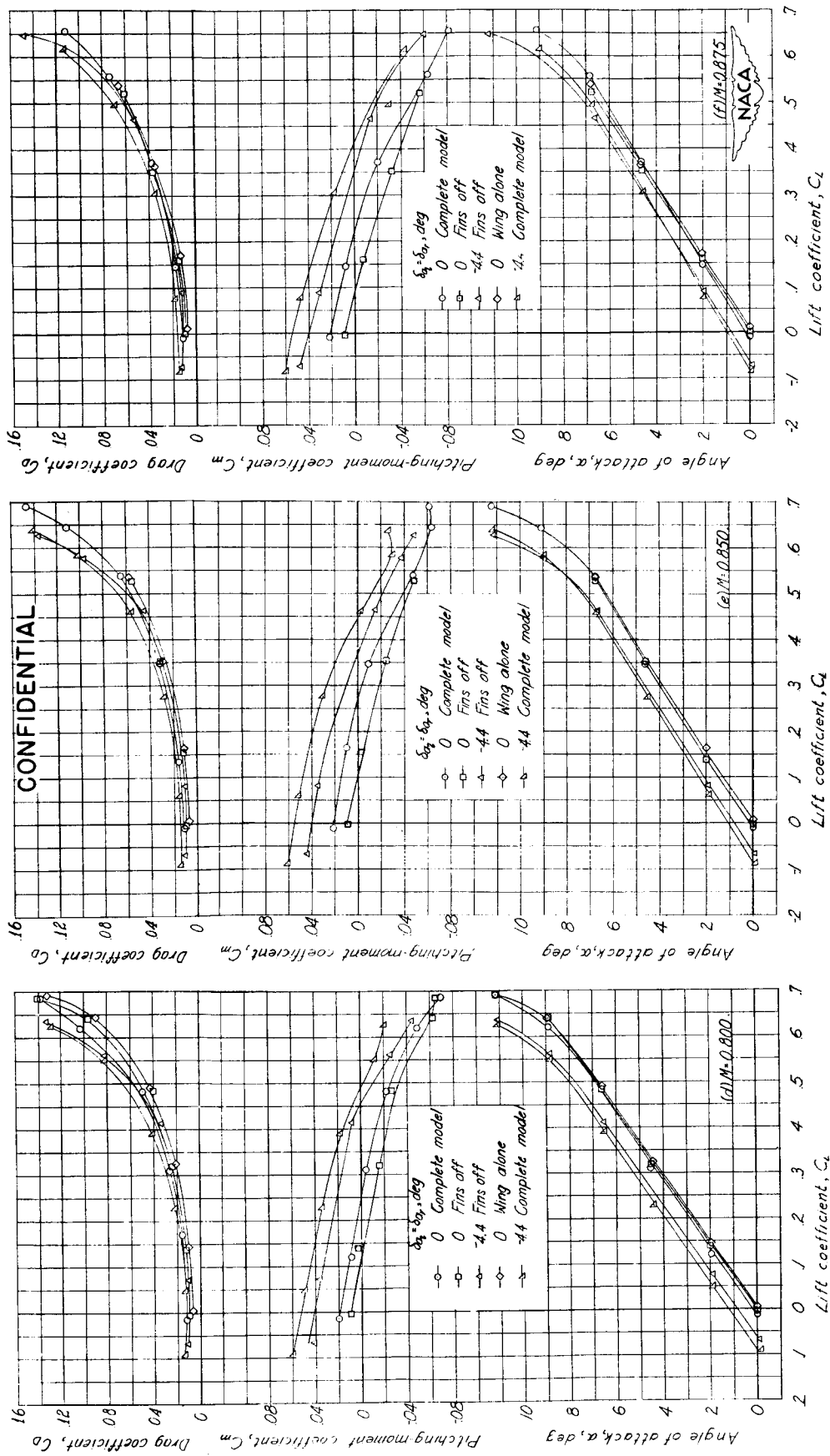


Figure 26. -fins off and wing alone characteristics of the test model.



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Figure 26.- Continued.

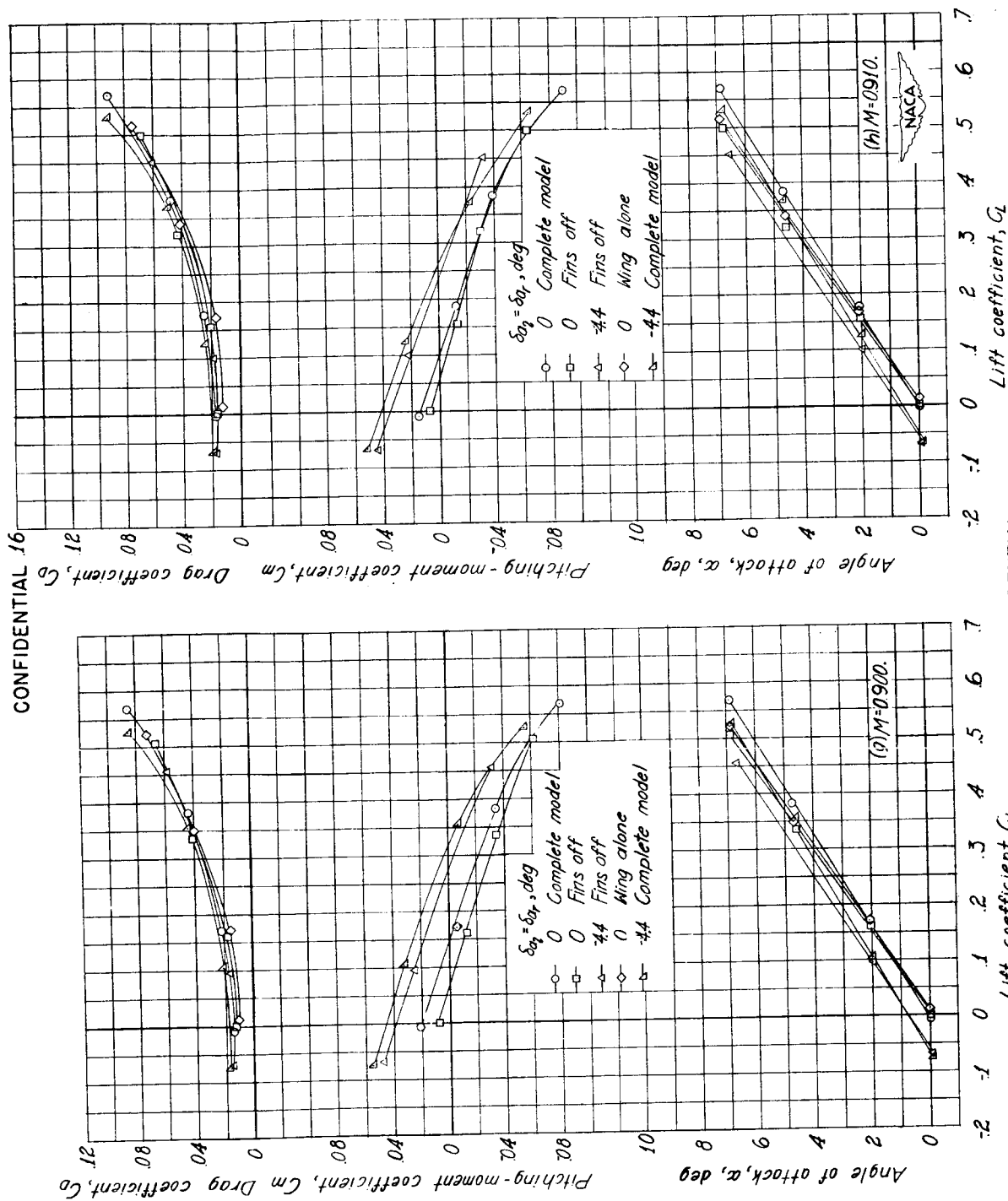


Figure 26. - Concluded.

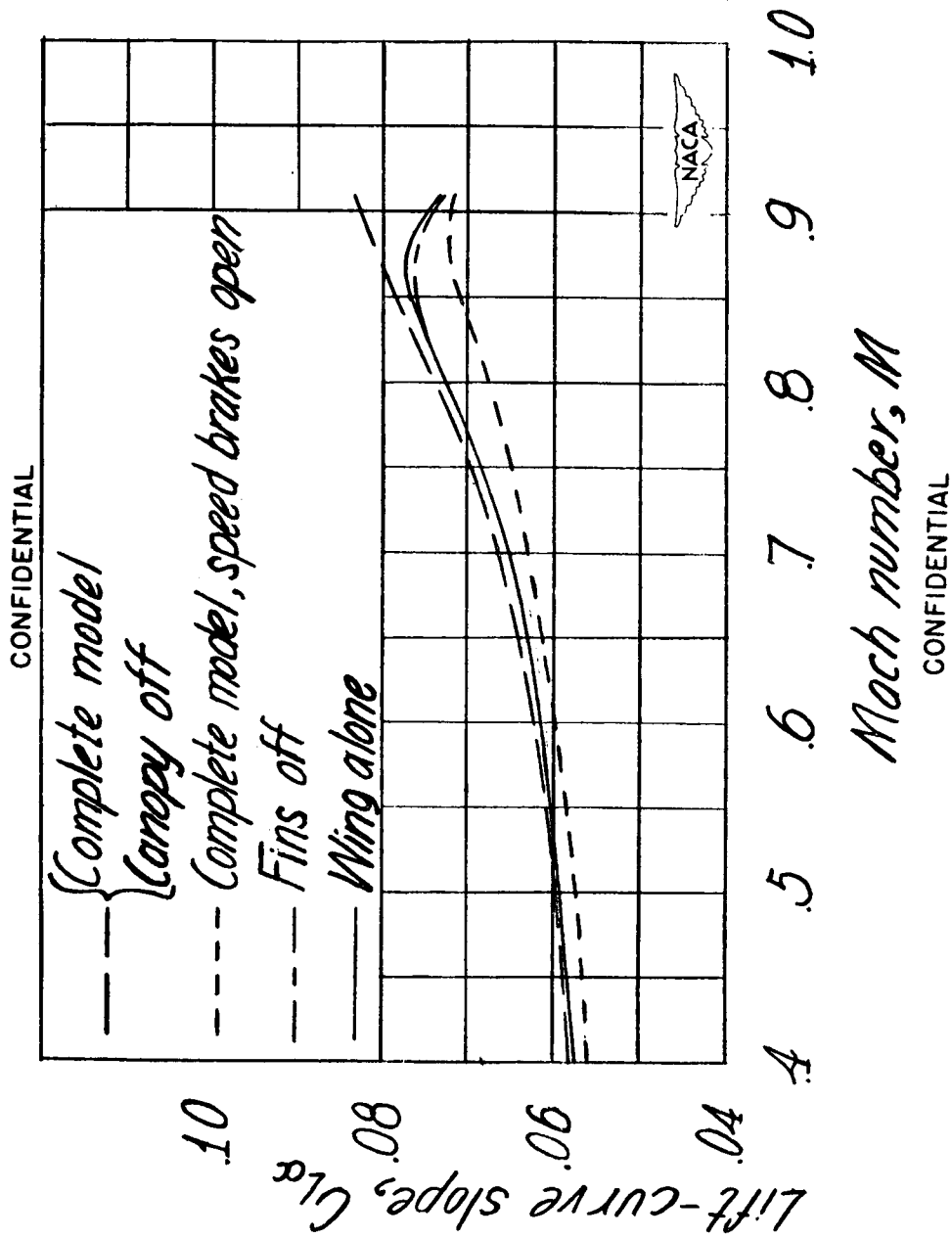


Figure 27.- Variation of lift-curve slope with Mach number for low lift coefficients for various configurations of the test model.

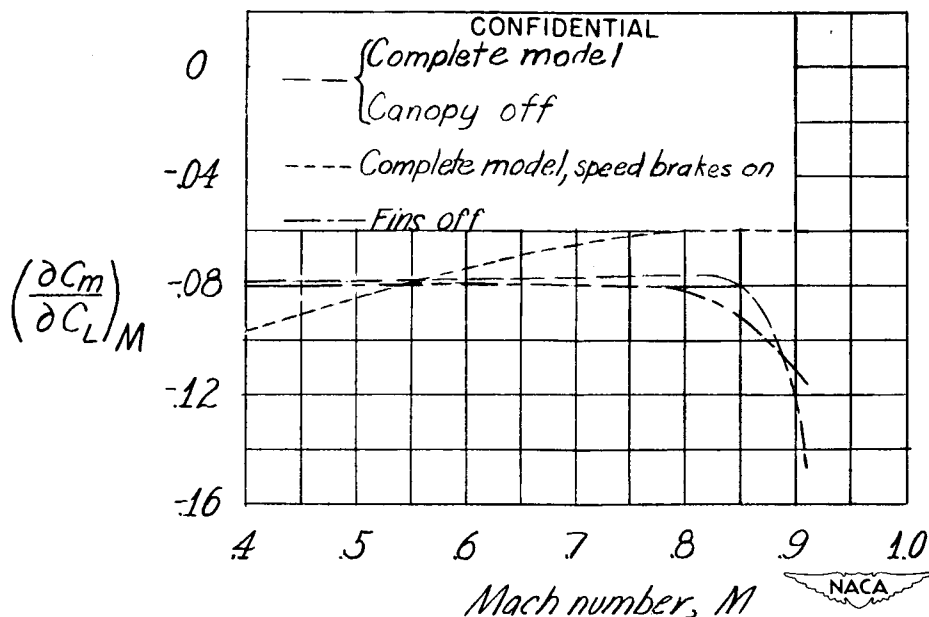


Figure 28.- Variation of  $\left(\frac{\partial C_m}{\partial C_L}\right)_M$  with Mach number for the low lift-coefficient range for various configurations of the test model.

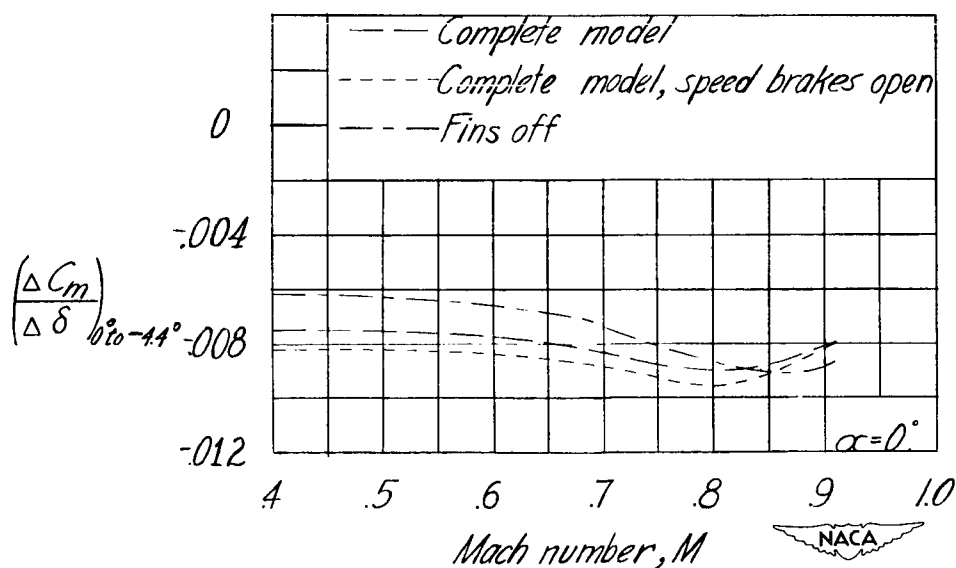
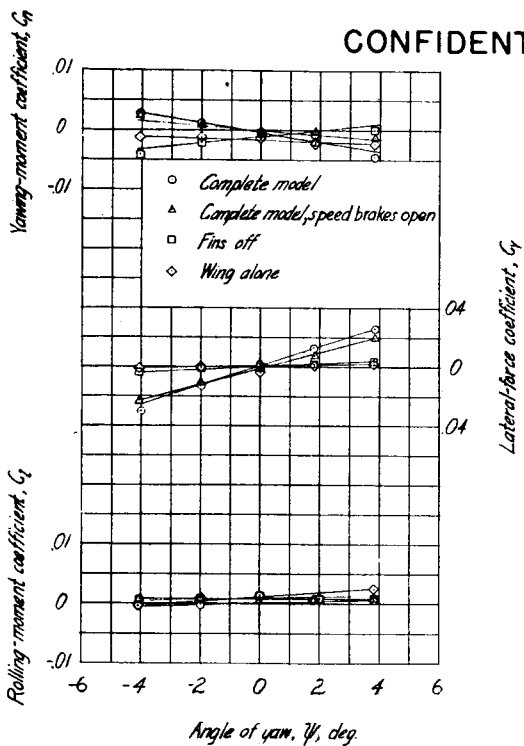
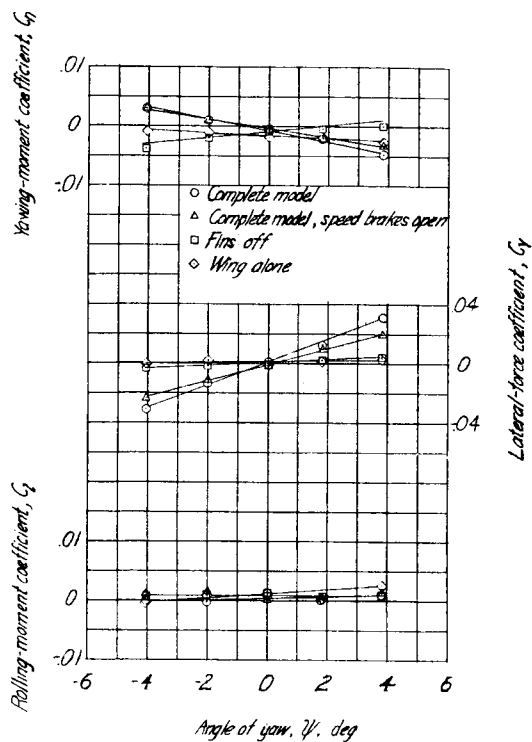
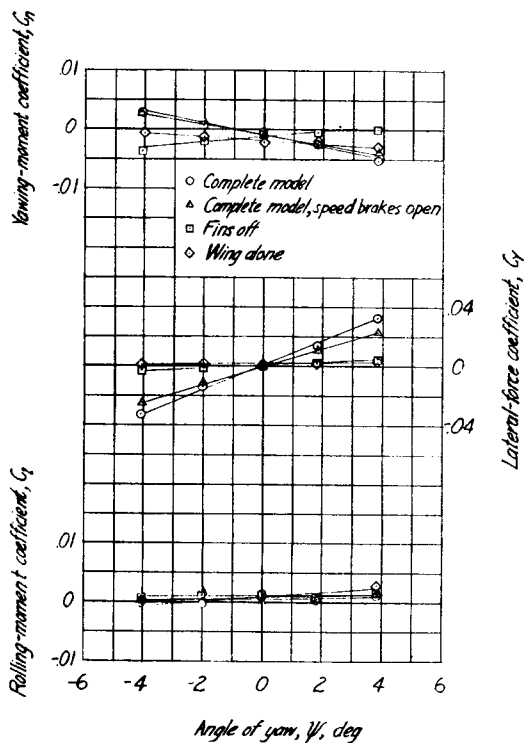
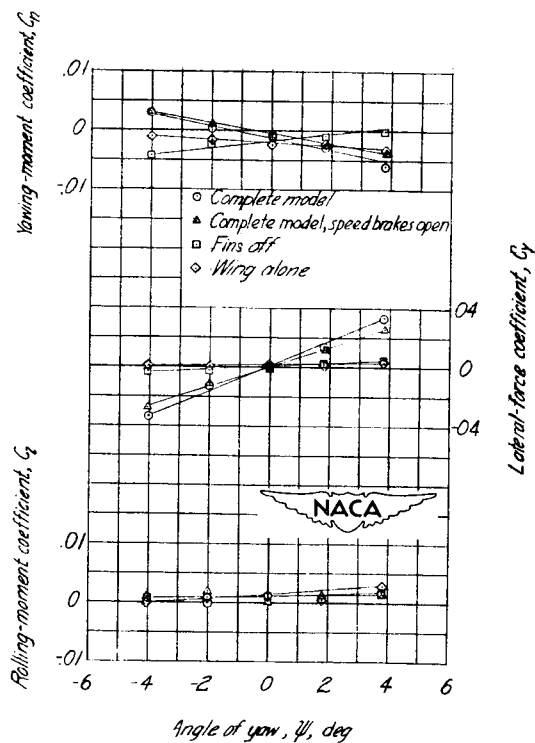


Figure 29.- Variation of the control effectiveness parameter  $\left(\frac{\Delta C_m}{\Delta \delta}\right)$  with Mach number for several configurations of the test model.

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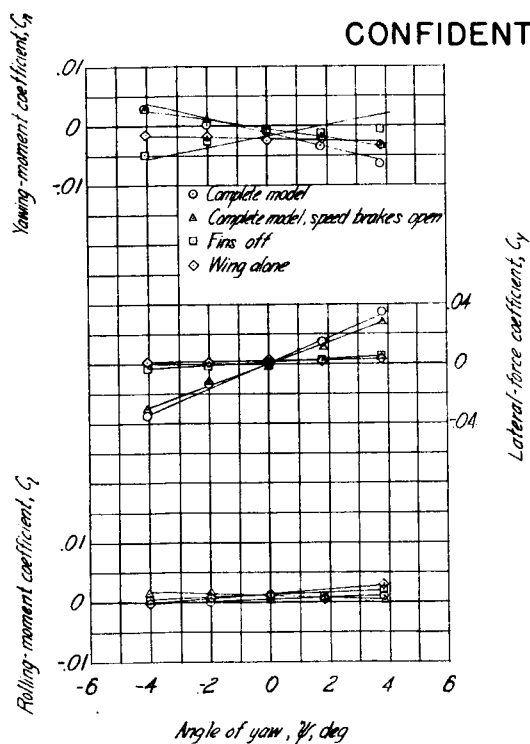
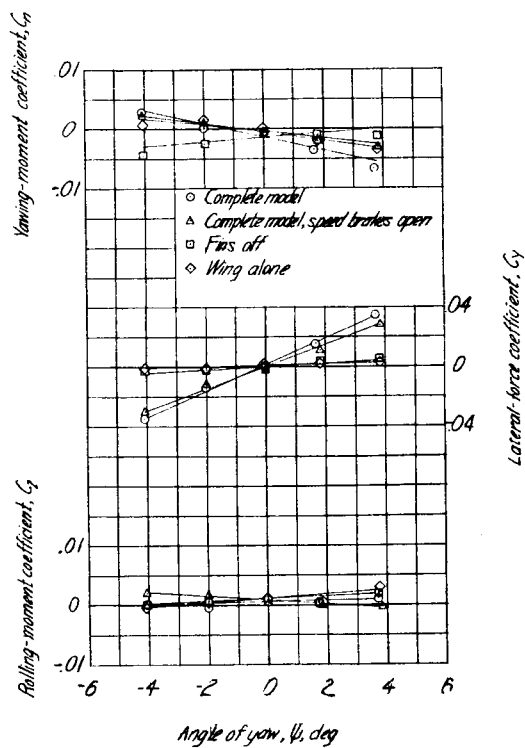
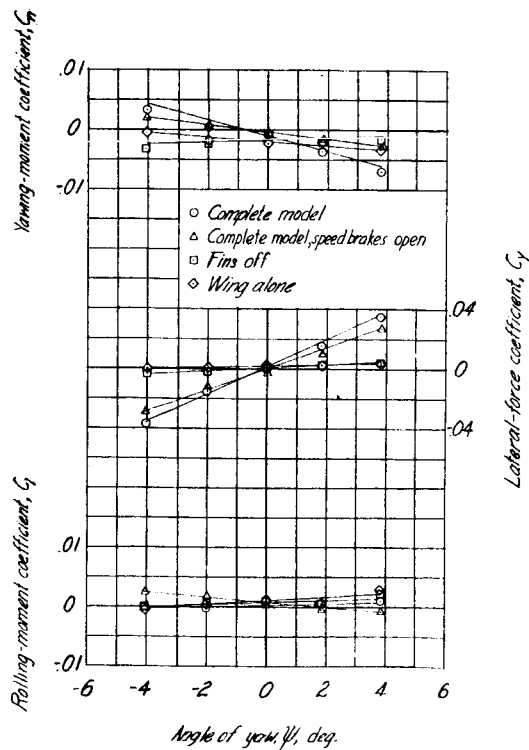
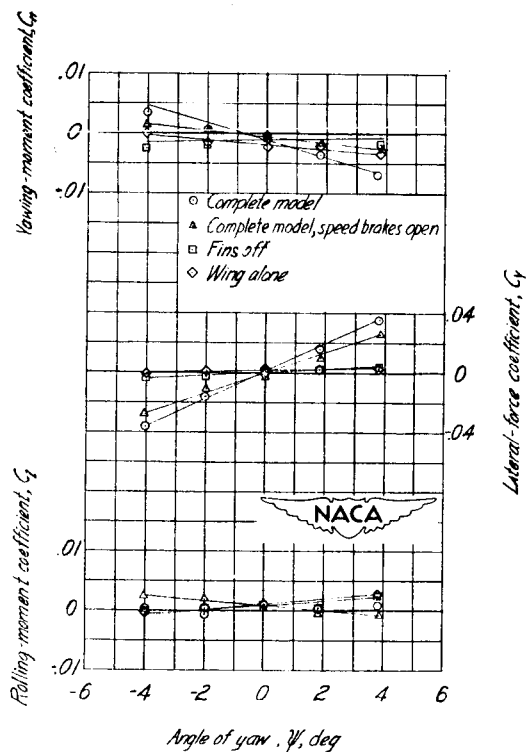
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(a)  $M=0.400$ .(b)  $M=0.600$ .(c)  $M=0.700$ .(d)  $M=0.800$ .

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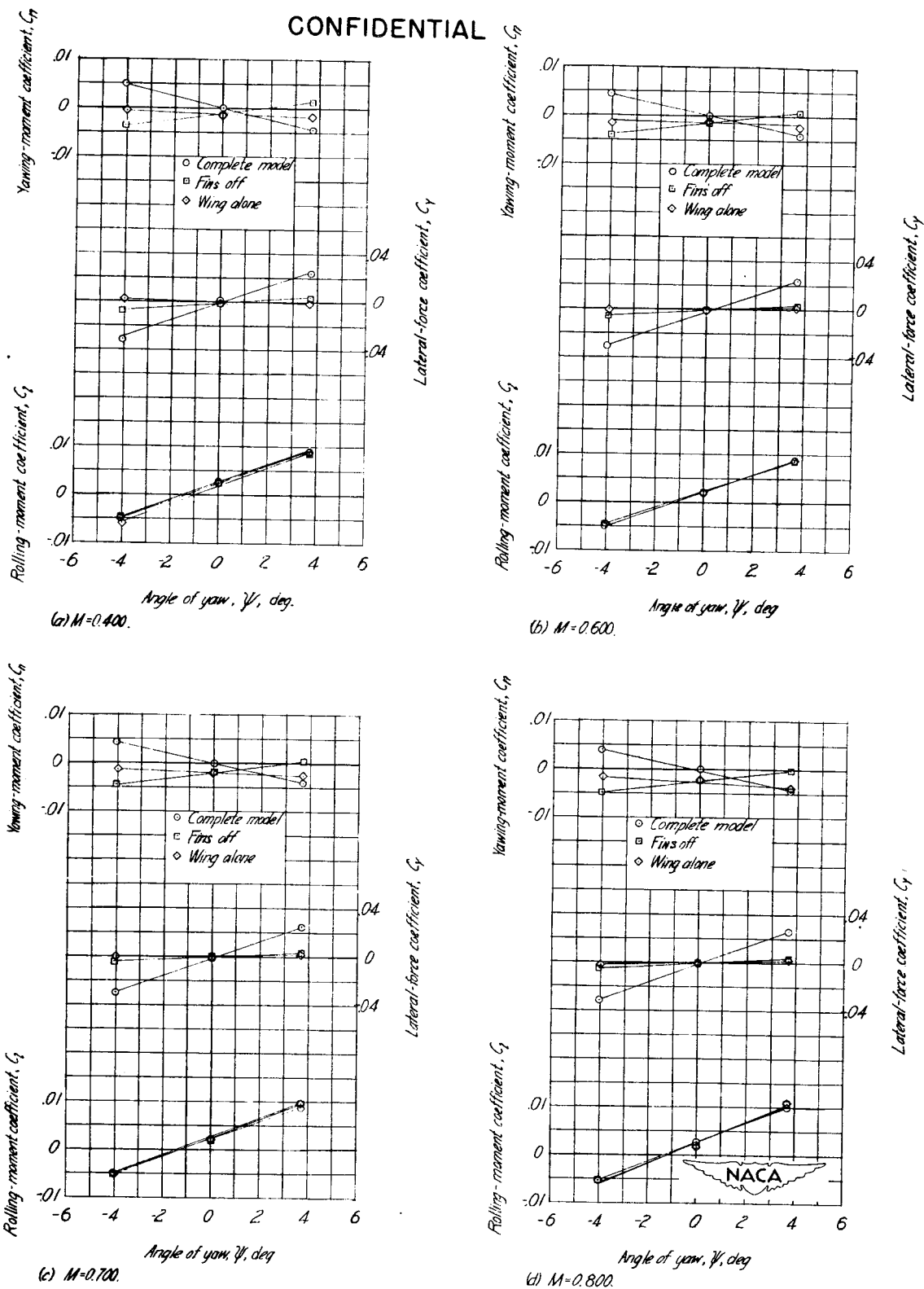
Figure 30.- Aerodynamic characteristics in yaw for several configurations of the test model,  $\alpha_{static} = 0$ .

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(e)  $M=0.850$ .(f)  $M=0.875$ .(g)  $M=0.900$ .(h)  $M=0.910$ .

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Figure 31 - Aerodynamic characteristics in yaw for several configurations of the test model;  $\alpha_{max} = 6^\circ$



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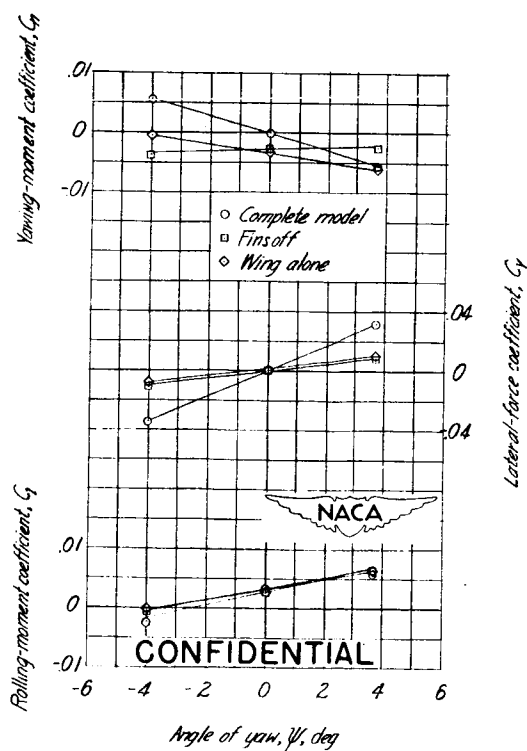
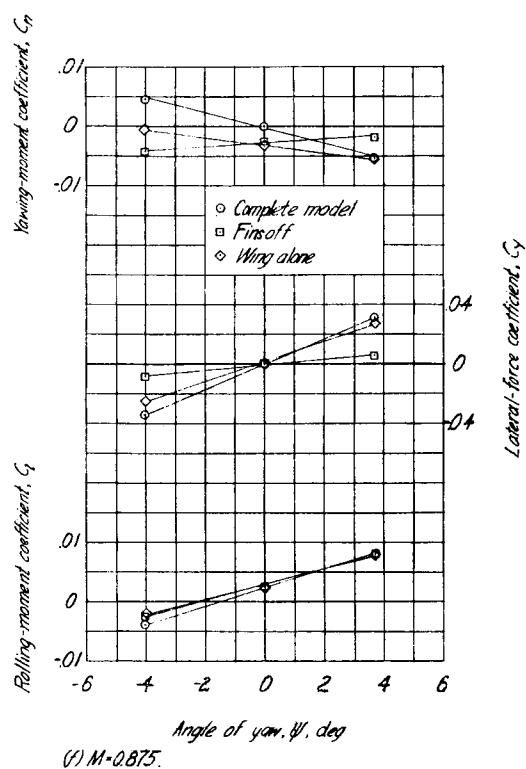
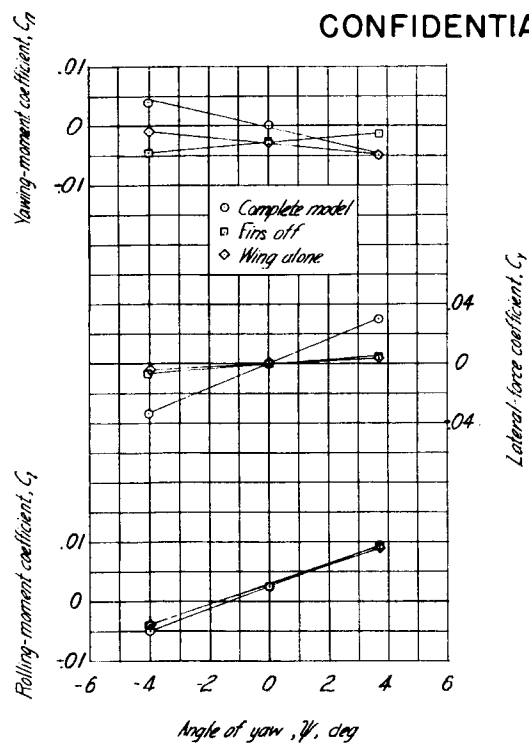


Figure 31 -Concluded

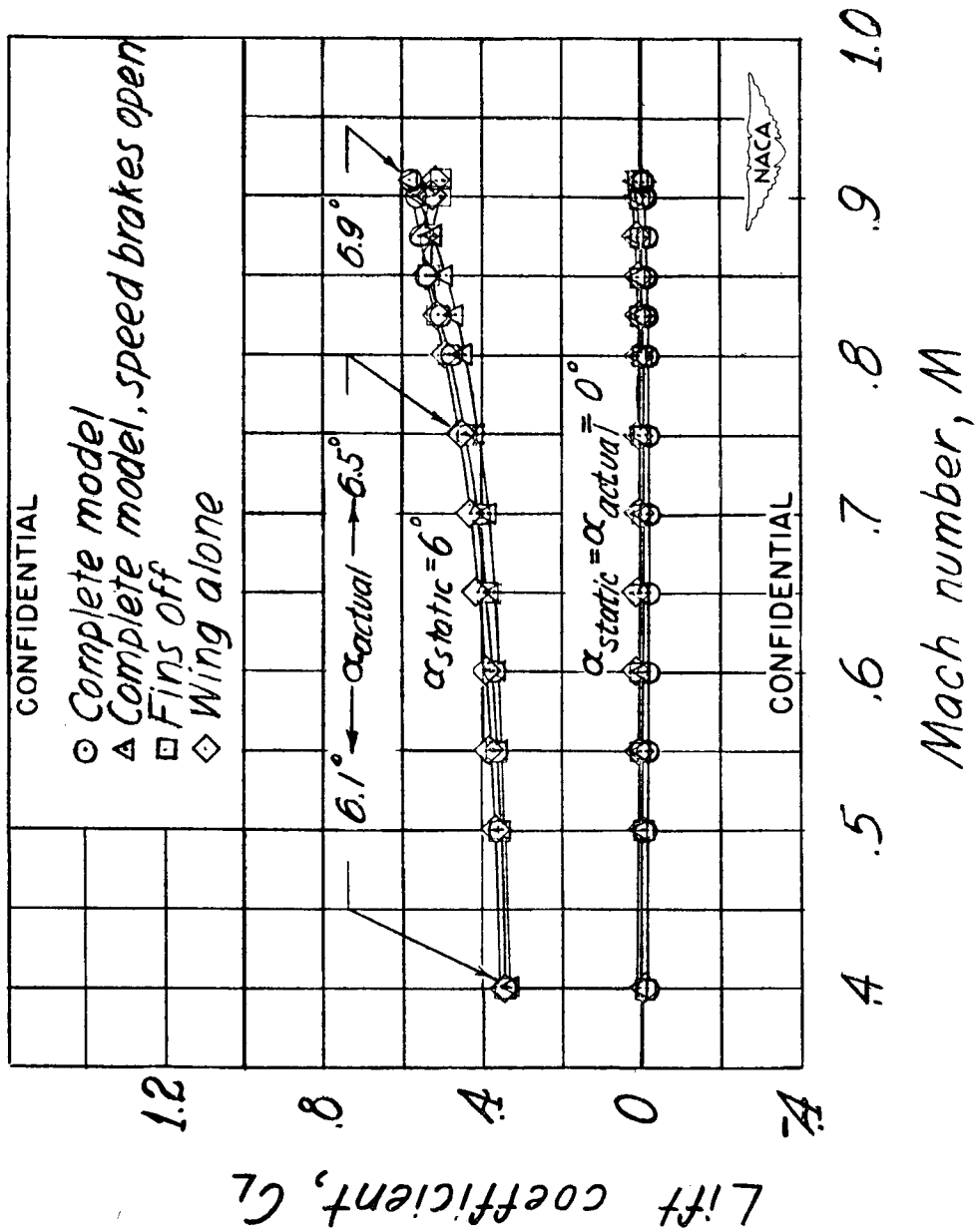


Figure 32-Variation of lift coefficient with Mach number at constant values of static angle of attack for several different model configurations of the test model.

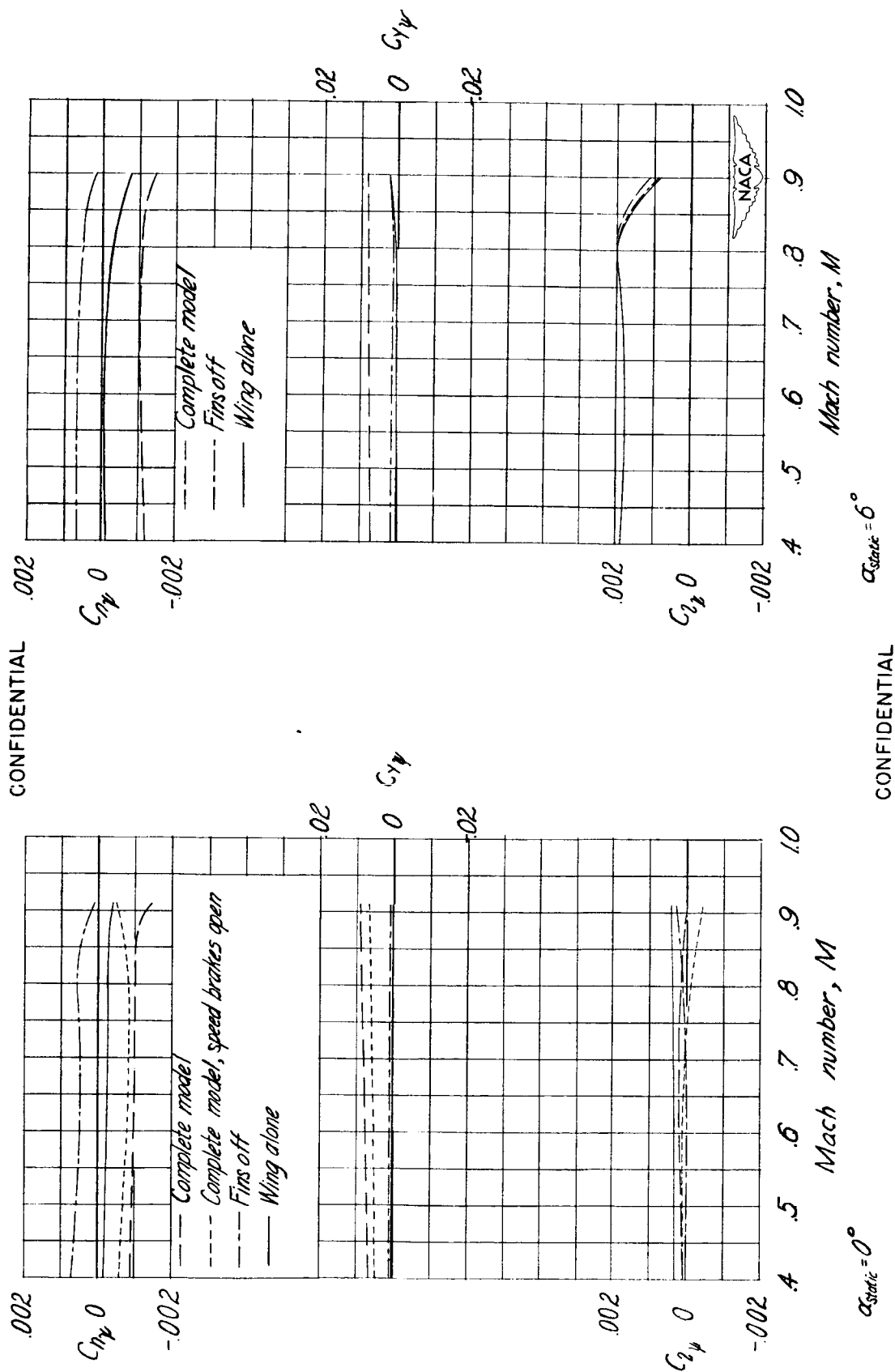
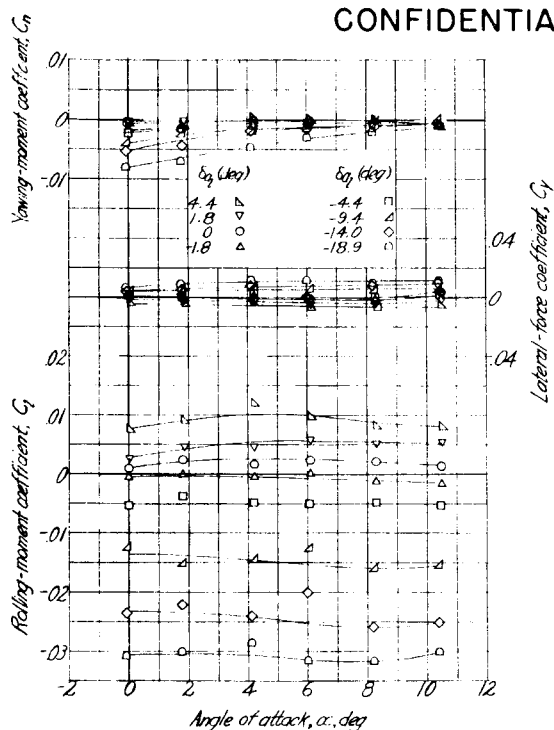
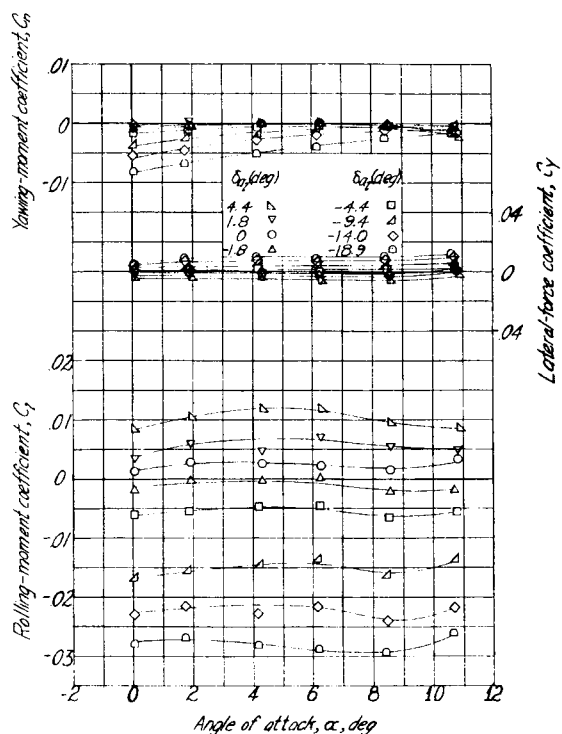
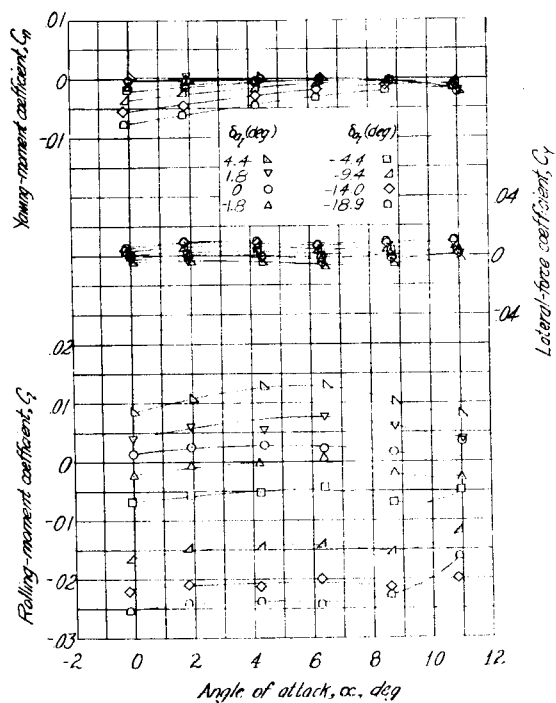
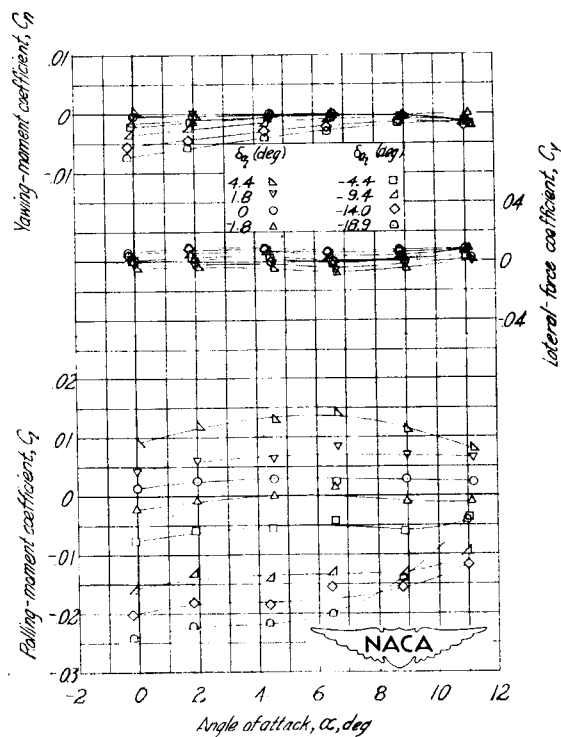


Figure 33- Variation with Mach number of the lateral stability characteristics for several configurations of the test model.

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(a)  $M=0.40$ .(b)  $M=0.60$ .(c)  $M=0.70$ .(d)  $M=0.80$ .

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Figure 34. Effect of airfoil deflection through an angle-of-attack range on the lateral characteristics of the test model.  $\delta_{a,0} = 0^\circ$

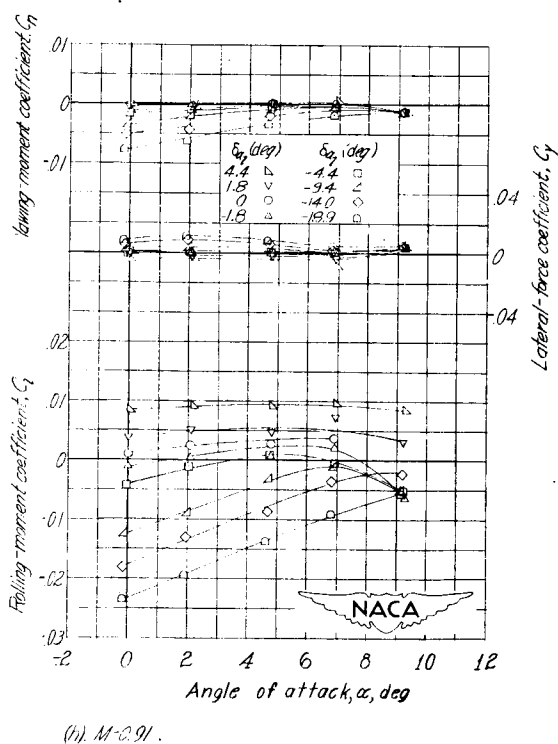
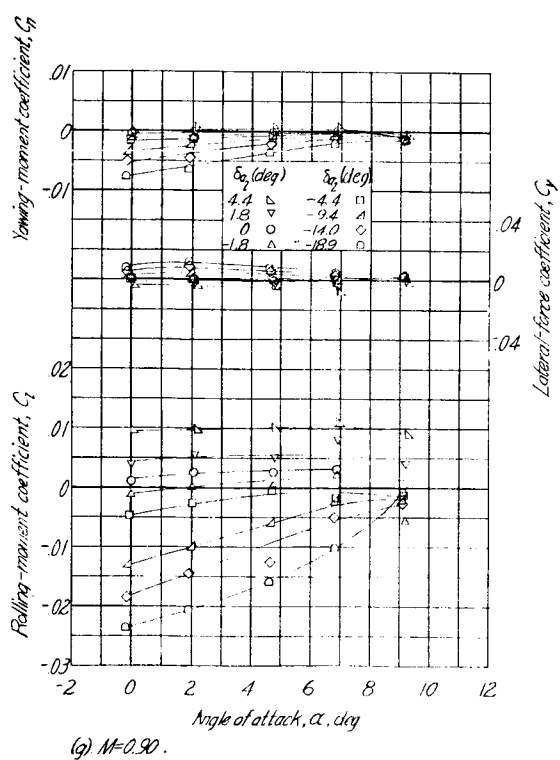
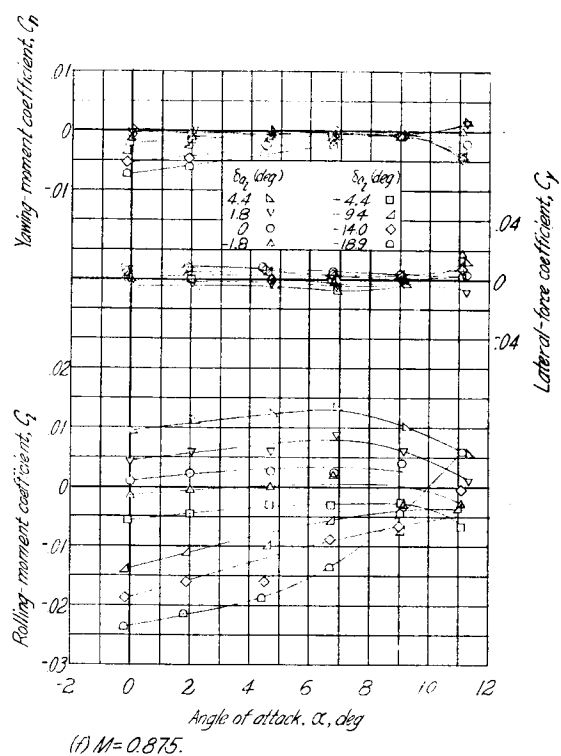
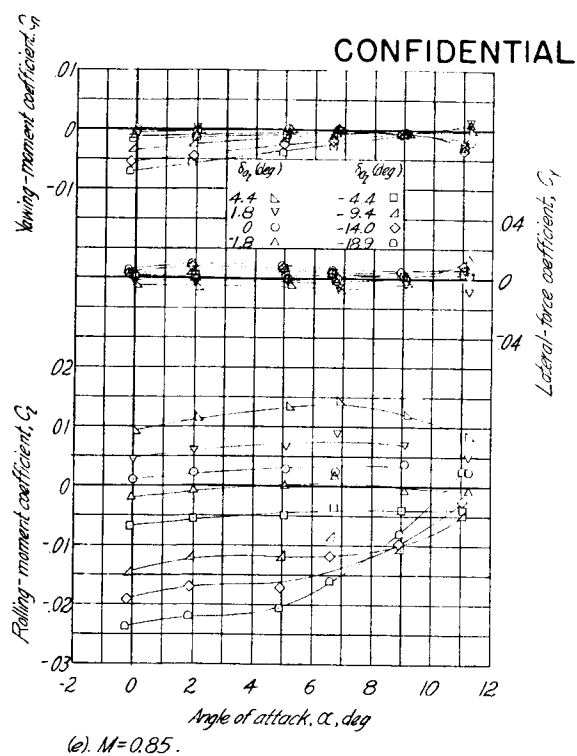
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Figure 3A-Continued

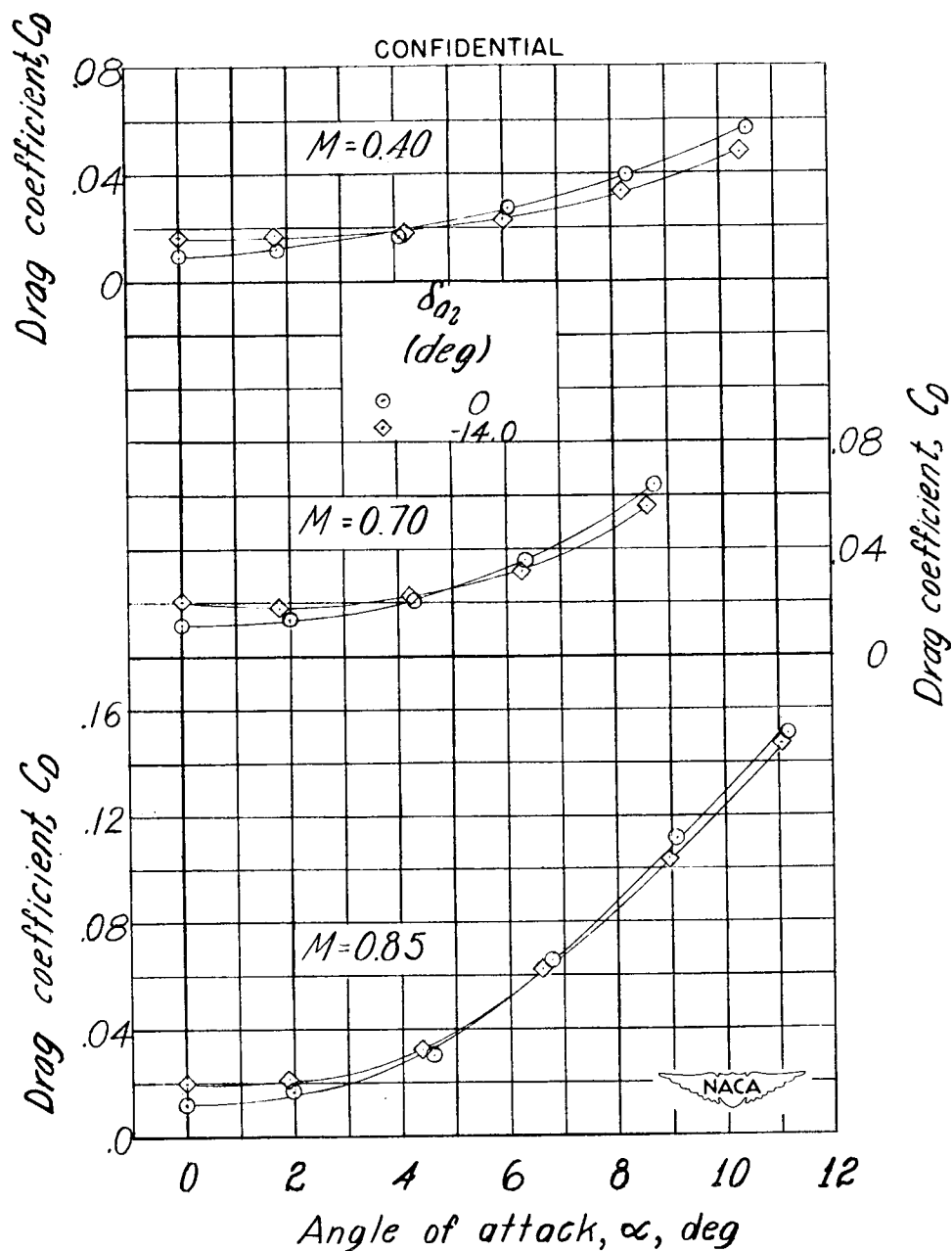


Figure 35.—Effect of aileron deflection through an angle-of-attack range on the drag characteristics of the test model.  $\delta_{ar} = 0^\circ$ .

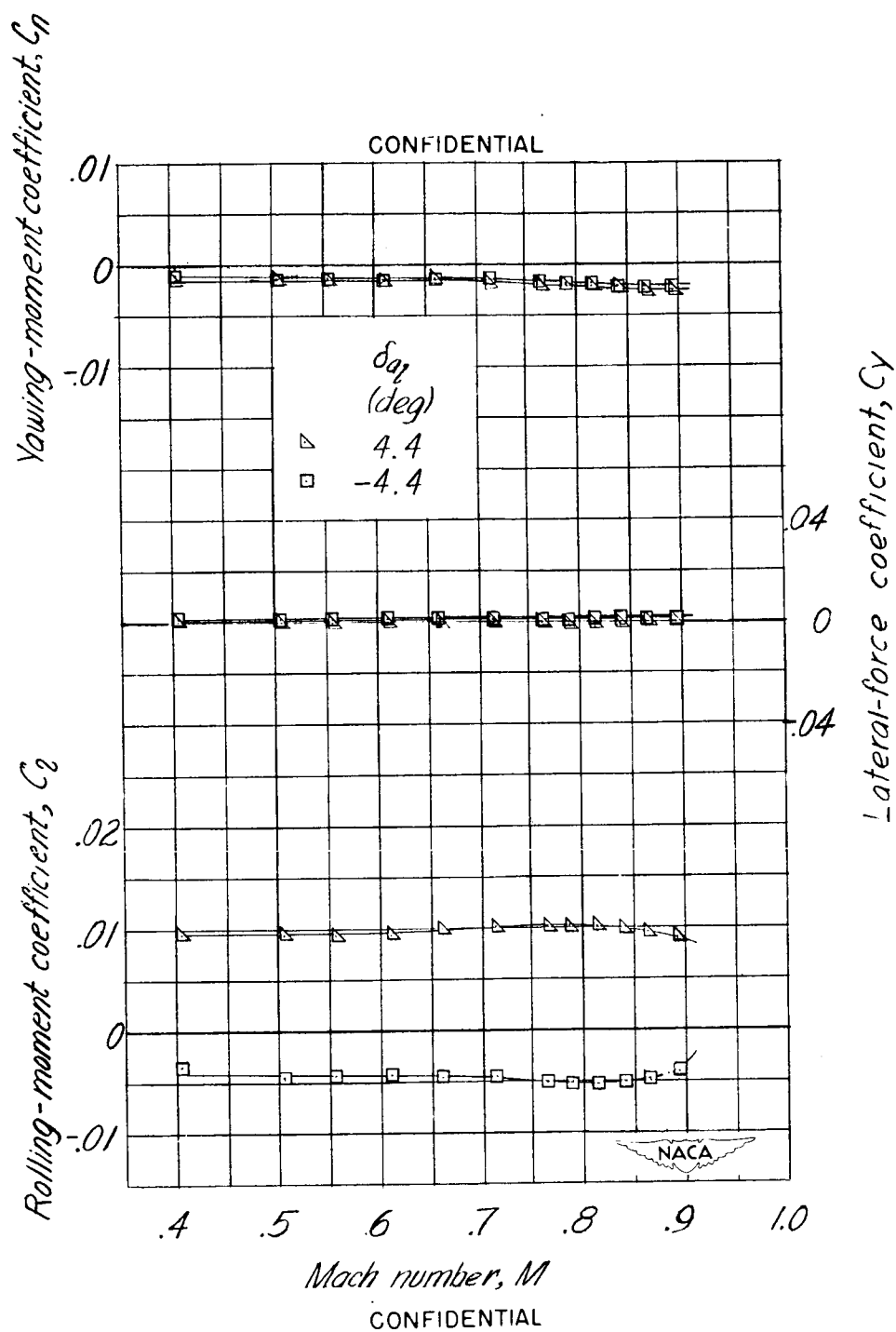


Figure 36.- Effect of aileron deflection through a Mach number range on the lateral characteristics of the test model.  $\delta a_r = 0^\circ$ ; vertical fins off;  $\alpha = 1.8^\circ$ .





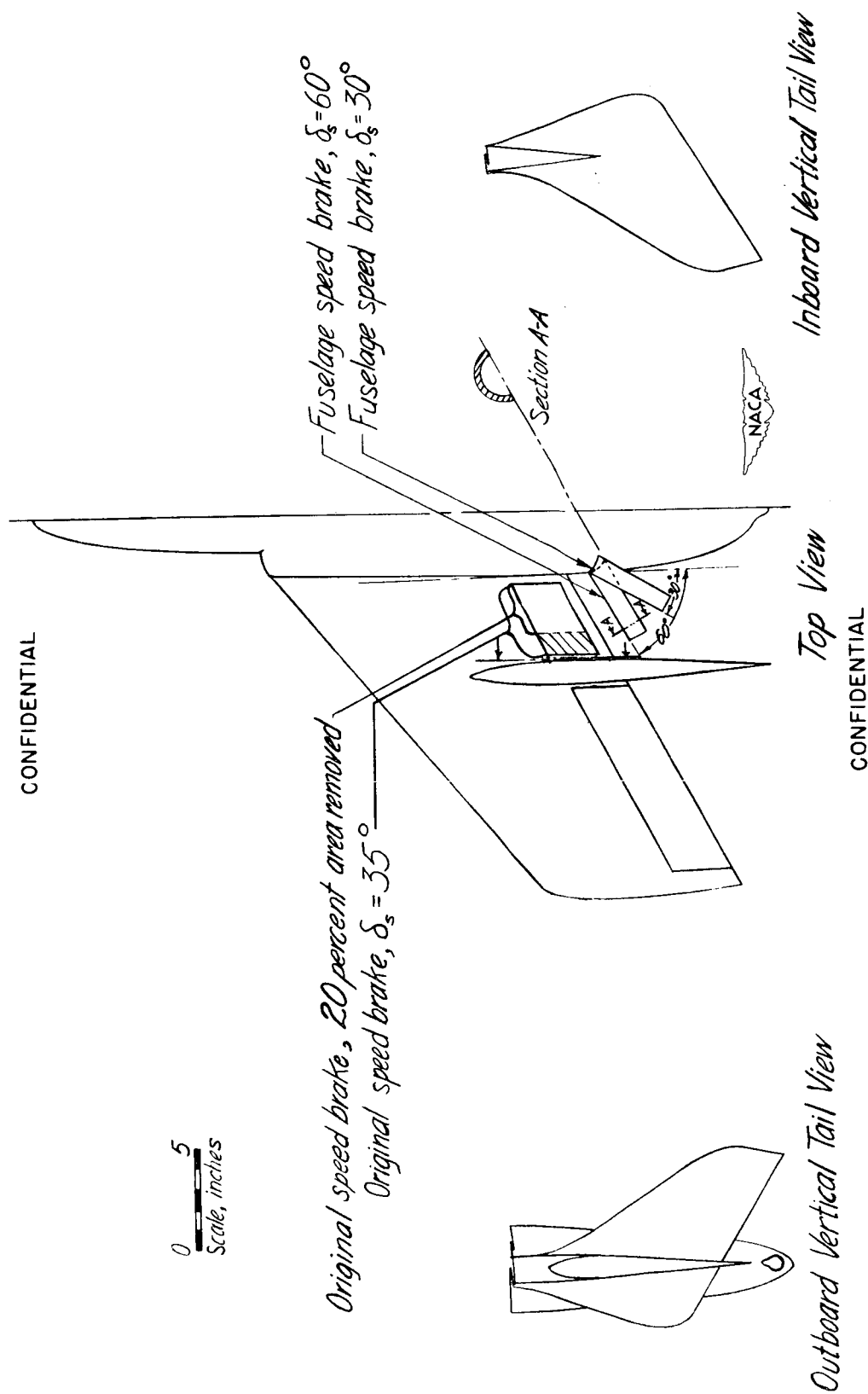
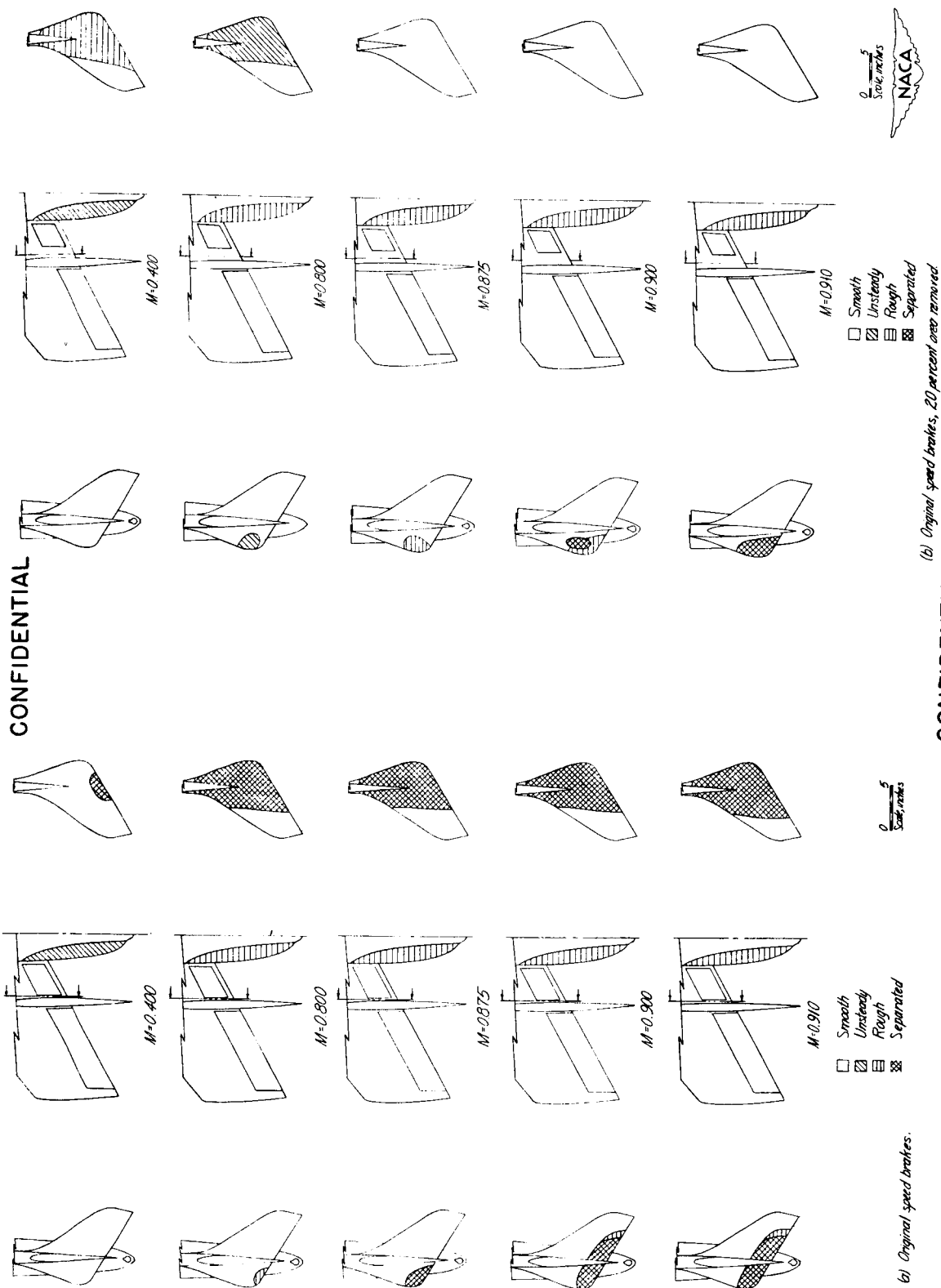
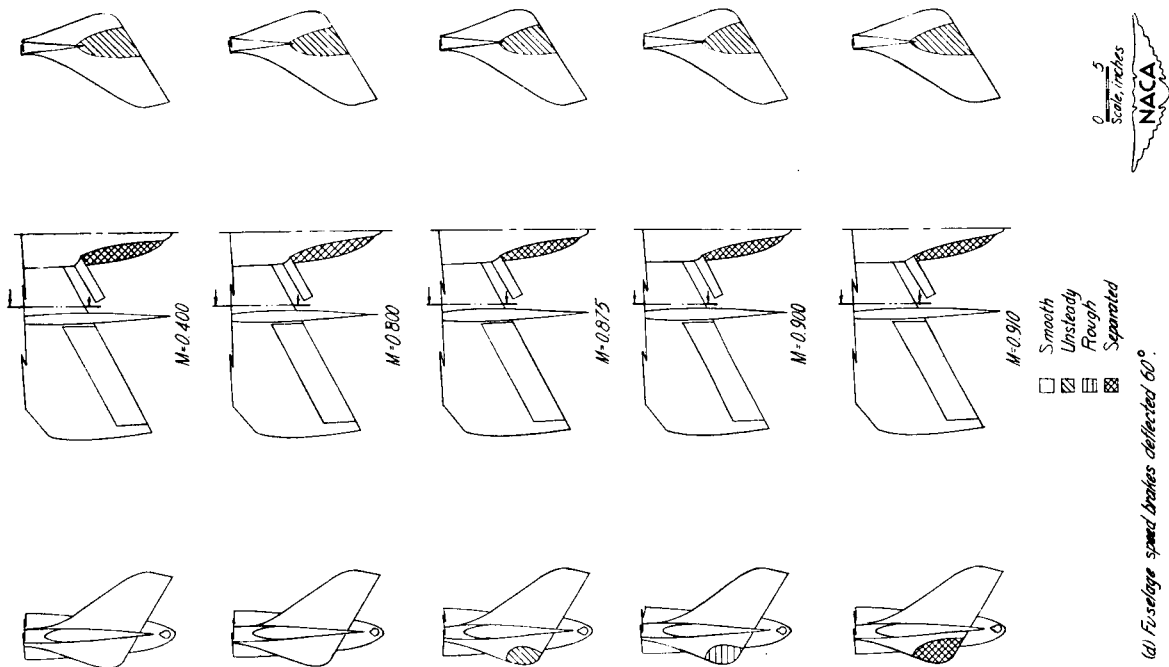


Figure 38.- General arrangement of views for tuft study presentation and the various speed-brake arrangements tested on the test model.

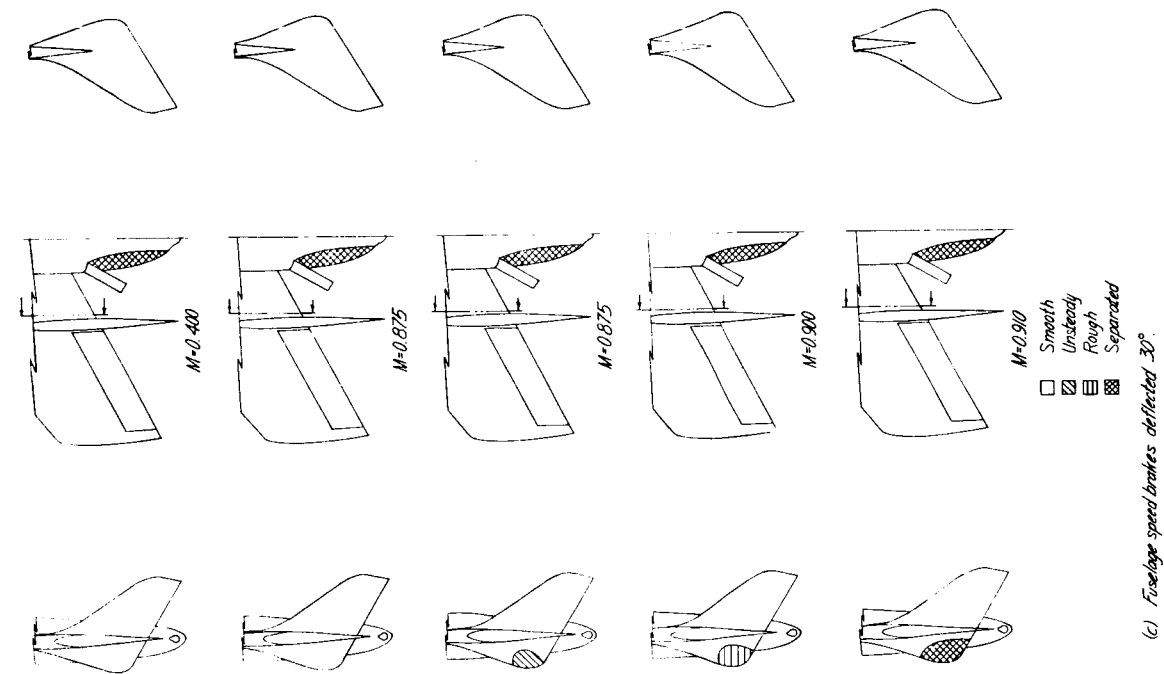




(c) FUSELAGE speed brakes deflected 60°

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(c) FUSELAGE speed brakes deflected 30°

Figure 39 - Continued

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- Original speed-brakes
- ◇—◇— Original brakes, area reduced 20 percent
- ▽—▽— Fuselage brakes, deflected 60°
- △—△— Fuselage brakes, deflected 30°
- All brakes closed

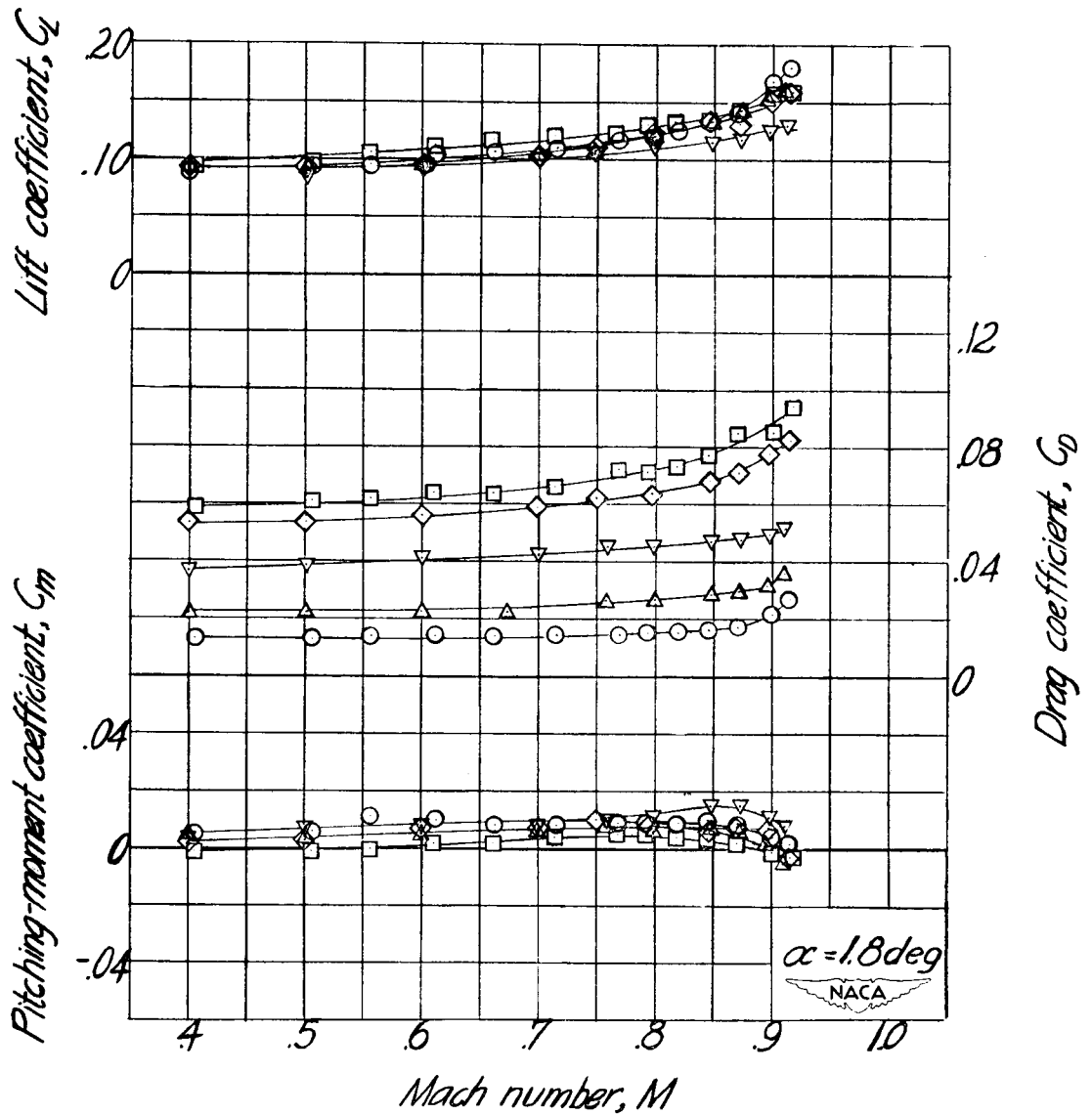


Figure 40.- Effect of various speed-brake configurations on the aerodynamic characteristics in pitch.

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SOME EFFECTS OF SWEEPBACK AND AIRFOIL THICKNESS ON LONGITUDINAL  
STABILITY AND CONTROL CHARACTERISTICS AT TRANSONIC SPEEDS

By Charles J. Donlan and Arvo A. Luoma

Langley Aeronautical Laboratory

INTRODUCTION

The accumulation of information on the longitudinal stability and control characteristics of complete transonic airplane configurations has proceeded less rapidly than other phases of aerodynamic research because of the difficulties involved in obtaining such data in the transonic speed range. Current information is largely based on the results of a few specific rocket model configurations (also the following paper by Clarence L. Gillis) and a few wind-tunnel investigations of complete models (references 1 to 3), plus whatever qualitative guidance can be provided from a number of wing-flow and transonic-bump investigations (references 4 to 10). In the present paper an attempt has been made to piece together some of this information in a form that might indicate whether or not a consistent pattern of behavior exists in regard to effects of airfoil thickness and sweepback on over-all stability and control characteristics at transonic speeds.

It is generally expected that airplane designs employing wings of low aspect ratio should encounter less severe stability and control difficulties in the transonic speed range and, consequently, most of the investigations that are useful for studying the effects of sweep and airfoil section on stability and control characteristics have been conducted on configurations employing wings of low aspect ratio.

STRAIGHT-WING CONFIGURATIONS

Scope.- The three straight-wing models shown in figure 1 represent similar but not identical configurations. All three models possessed somewhat different fuselage shapes but the plan forms of the wings and tail as well as the tail location were about the same. Model A was a complete configuration that was investigated at high subsonic Mach numbers in the Langley 8-foot high-speed tunnel and model B was a complete configuration investigated through the transonic range as a rocket model. The major difference between the models is found in the thickness ratios of airfoil sections employed. Model C was not actually a complete model at all but rather a synthetic configuration produced by combining wing-fuselage aerodynamic characteristics obtained from bump tests with the

measured downwash and wake characteristics appropriate to the particular tail arrangement. Models C and B had the same airfoil sections for the wing and tail.

Stability characteristics.- The variation of the stabilizer required for trim with Mach number for the three configurations is shown for an altitude flight condition in the upper part of figure 2. Although the wind-tunnel data are limited to  $M = 0.95$  it is evident that the configuration A, with the thicker airfoil section, experienced earlier, more rapid, and larger trim changes than the rocket model (model B), which behaved quite satisfactorily throughout the Mach range investigated. It is also interesting to find that the synthetic bump configuration (model C) employing essentially the same wing and tail as the rocket model also indicated small and gradual trim changes throughout the Mach number range.

The lower part of figure 2 illustrates one manner in which maneuvering stability may influence the amount of trim change that can be safely tolerated. The variation in control required for 1g and 2g flight for model A is almost identical. Only the curves for 1g and 2g are shown here but actually similar variations were found for higher accelerated flight conditions also. With a control characteristic of this kind - particularly if the stick-force variation follows a similar pattern - the pilot may easily experience abrupt accelerations as a result of slight inadvertent Mach number changes. The behavior exhibited by model C, on the other hand, is much more desirable inasmuch as accelerations must be produced by control movement and cannot arise from slight inadvertent changes in Mach number.

The reasons for the superior characteristics exhibited by the models with the thinner wings and tail surfaces are found in the behavior of the various stability components. Unfortunately, breakdown information of this kind is lacking for model B, but such a breakdown study has been made for models A and C and is presented in figure 3.

Stability analysis.- Figure 3 summarizes all the information essential to the analysis of configurations having nonlinear stability characteristics. It illustrates how the basic stability components vary with Mach number for the particular flight plan selected and the manner in which these components combine to produce the final result. The three factors on the left -  $C_{m_{WF}}$ ,  $\alpha_{WF}$ , and  $\epsilon$  - may be thought of as the primary components and the three factors on the right -  $\alpha_T$ ,  $(\alpha - \epsilon)$ , and  $i_t$  - as the derived components. The factor  $\alpha_T$  is the angle of attack of the tail (relative, of course, to the local flow direction) required to balance the wing-fuselage pitching-moment coefficient  $C_{m_{WF}}$  at each value of Mach number. Nonlinearities in the lift characteristics of the tail itself are manifested directly in this quantity also. The

factor  $(\alpha - \epsilon)$  is the local flow angularity existing at the tail and  $i_t$ , of course, is merely  $[\alpha_T - (\alpha - \epsilon)]$  and represents the amount the stabilizer must be adjusted to produce the angle of attack  $\alpha_T$  relative to the local flow direction  $(\alpha - \epsilon)$ . If the models possessed linear variations of  $C_{m_{WF}}$ ,  $\alpha$ , and  $\epsilon$  with  $C_L$  all the factors shown should vary in a gradual hyperbolic manner with  $M$  in the absence of compressibility effects. It is evident that model A exhibited more marked irregularities in all components than model C. Model A is a particularly useful one for illustrating the complexity of the problem. Inasmuch as  $C_{m_{WF}}$ ,  $\alpha_{WF}$ ,  $\epsilon$ , and  $\alpha_T$  all exhibit specific irregularities of their own, the values of  $i_t$  at each Mach number represent only one of several values that might have resulted from combinations of these variables. For example, the rapid rise in the effective downwash angle  $\epsilon$  at  $M = 0.90$  has been traced to local interference effects between the vertical fin, the horizontal tail, and the fuselage rather than to the wing itself (reference 2). Yet, because of compensating effects in other components, the over-all variation of  $i_t$  in the Mach number range considered is less with this interference effect than it would have been without it. Thus, a configuration can be conceived of that has excellent stability and trim characteristics as a whole despite the fact that individual components may indicate rather erratic behavior. Such fortunate compensating circumstances are obviously difficult to anticipate, however, and satisfactory characteristics are more likely to be obtained if the basic stability components vary less violently with Mach number as, for example, occurred for the model with the thinner wing and tail.

### SWEPT-WING CONFIGURATIONS

A similar study of the effect of wing thickness on the stability characteristics of a swept-wing configuration for which Langley 8-foot high-speed-tunnel results and rocket-model results are available is shown in figure 4.

Models. - The wind-tunnel and rocket models are designated as models A and B. These models were of identical geometry except for the sweepback of the horizontal tail. For the sake of comparison a bump model has been added. The bump model had a somewhat different fuselage but had essentially the same plan form of the wing and tail and approximately the same tail height and tail length as models A and B. The main difference between the models is again in the thickness of the airfoils employed. The airfoil for models A and B is designated as normal to the 0.30-chord line. The tip section, however, corresponds to a streamwise thickness ratio of about 10 percent as compared with the 6-percent wing of model C.

Stability characteristics.- The results for level flight for the assumed flight plan indicated that the rocket model (model B) underwent rapid, irregular, and large trim changes in the transonic range. The wind-tunnel model (model A) indicated an initial trim change similar to model B but occurring at a slightly higher Mach number. Unfortunately, no data were available in the critical transonic range but the point at  $M = 1.2$  is consistent with the trends indicated by the rocket model. The absolute differences in trim between models A and B may be partially due to the difference in tail plan form and Reynolds numbers although much of it is believed to be caused by the greater flexibility of the rocket model. This may account also for the earlier Mach number at which trim changes occurred on the rocket model. Model C, with the thinner profile, appeared, on the other hand, to be free of rapid and large trim changes in the transonic range. An analysis of the component data available for models A and C indicated that for this case the superior behavior of model C was associated with less irregular downwash changes at the tail and less irregular lift characteristics of the thinner horizontal tail.

#### EFFECT OF SWEEP

A comparison of the wind-tunnel results for model A with those for the straight-wing model of similar thickness as shown in figure 2 would have indicated some improvement due to sweep, particularly in regard to the delay in initial trim changes. On the basis of the limited results given in figure 4, however, it appears that airfoil thickness is so important that it may completely mask any effects of sweep. It is important, therefore, in evaluating the effects of sweep to compare configurations having airfoil sections of comparable thickness. At the present time the only extensive systematic information of this kind that is available was obtained from bump tests of the configurations shown in figure 5.

Methods.- The models were tested as wings alone and as wing-fuselage combinations and some of the basic force and moment characteristics, as well as the downwash and wake characteristics existing at the tail, are discussed in the paper entitled "The Lift, Drag, and Pitching-Moment Characteristics of Wings and Wing-Body Combinations in the Transonic Speed Range" by Edward C. Polhamus and in a subsequent paper entitled "Downwash and Wake Characteristics at Transonic Speeds" by Joseph Weil and Ralph P. Bielat. As for the present analysis, an attempt has been made to synthesize the stability and control characteristics of the four configurations shown by combining the basic wing-fuselage data with the isolated force characteristics of a horizontal tail operating in the particular flow field existing at the tail. Inasmuch as wake and downwash data were available for a number of vertical positions the study



was also extended to include the effect of tail height. It is realized that this procedure of adding together component data does not take account of any interference effects at the tail but may serve, nevertheless, to indicate qualitative trends, particularly for thin wings where these interference effects are not so pronounced. The calculations, which are summarized in figure 6, were made for level flight for one flight condition and, except for the  $60^\circ$  case, the center of gravity was selected to provide a static margin of about 0.10 at zero lift. For the  $60^\circ$  configuration it was necessary to adopt a larger static margin - about 0.30 - because the unstable break in the wing pitching-moment data occurred at a smaller lift coefficient.

Stability characteristics.- The most interesting result of these calculations is that for this series of wings of aspect ratio 4 and 6-percent thickness ratio there was no particular sweep angle or tail height that was outstandingly superior to the others. The least trim changes encountered did seem to occur for the  $35^\circ$  configuration, particularly for the highest tail position, but all the configurations appear to be quite satisfactory. It will be necessary, of course, to extend such studies to include accelerated flight conditions to establish definitely the superiority of any one configuration. Such studies, however, must await the acquisition of data at higher lifts, and particularly at higher Reynolds numbers and these investigations are now underway.

The discussions of stability and control thus far have considered the use of an all-moving tail as the longitudinal control. If an elevator control is used the accompanying trim changes will, in general, depend on the particular stabilizer setting employed. This effect is illustrated in figure 7.

#### EFFECT OF ELEVATOR CONTROL

The example selected to illustrate the effect of elevator control is the zero-sweep configuration of figure 6, although the  $i_t$  variation for the configuration shown in figure 7 is for a tail height of  $0.20 \frac{b}{2}$ .

The elevator angle required for trim for two stabilizer settings also has been computed by using flap-effectiveness data obtained by the transonic-bump method on a 0.30c full-span flap (reference 11). The reason for the different trim characteristics exhibited by the elevator control is traceable to the effectiveness characteristics shown on the right side of the figure. This figure shows the manner in which the tail lift due to angle of attack  $C_{L_\alpha}$  and the tail lift due to flap deflection  $C_{L_\delta}$  vary with Mach number, expressed as a fraction of their values at  $M = 0.80$ .

If the  $C_{L\delta}$  variation with  $M$  were identical to the  $C_{L\alpha}$  variation - that is, if the ratio  $C_{L\delta}/C_{L\alpha}$  were constant - the elevator trim curves would be found to have identical variations with Mach number and would be merely displaced from one another by an amount dependent on the value of  $\alpha/\delta$ . Inasmuch as the ratio  $\alpha/\delta$  is unlikely to be a constant with Mach number it appears that the trim changes obtained with elevator control can always be expected to depend on the particular value of stabilizer used and, hence, also on the particular flight plan employed, and, perhaps, on the piloting technique itself.

#### SUMMARY

In recapitulation, the limited results obtained thus far suggest that a thin wing is of paramount importance in securing minimum stability and control changes at transonic speeds and that if the wing and horizontal tail are thin enough the effects of sweep on the over-all stability and control characteristics at transonic speeds may be of secondary importance. The beneficial effects of sweep do increase, however, as the wing thickness increases. It appears also from aerodynamic considerations alone that an all-moving tail is to be preferred as the longitudinal control instead of an elevator control if stability and control characteristics are to be less dependent on particular flight conditions and piloting techniques.

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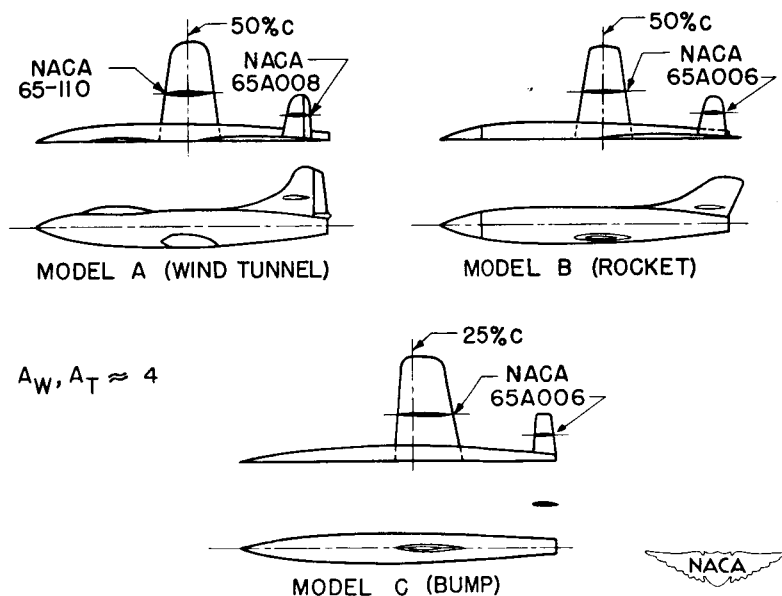


Figure 1.- Straight-wing models.

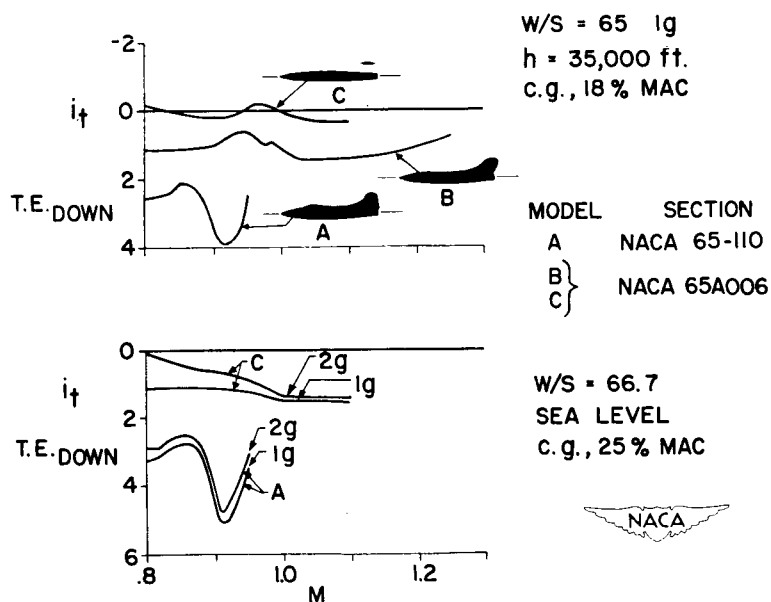


Figure 2.- Longitudinal stability characteristics of straight-wing configurations.

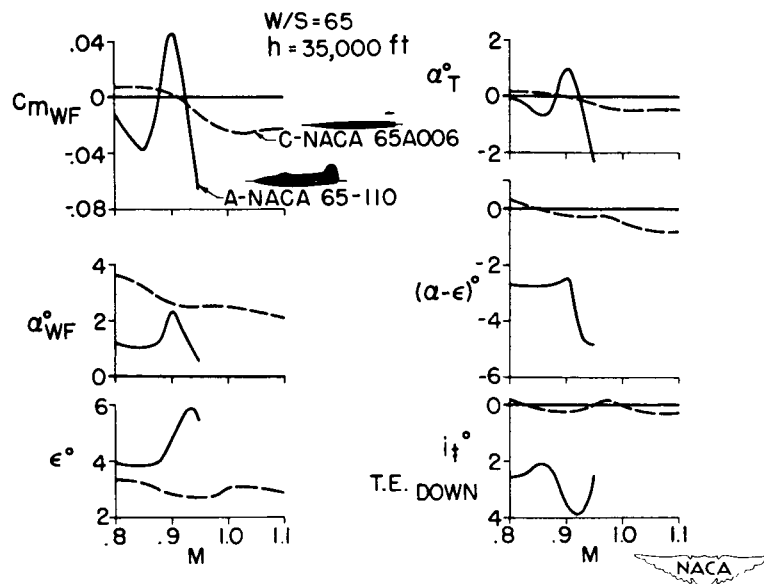


Figure 3.- Stability analysis of straight-wing models A and C.

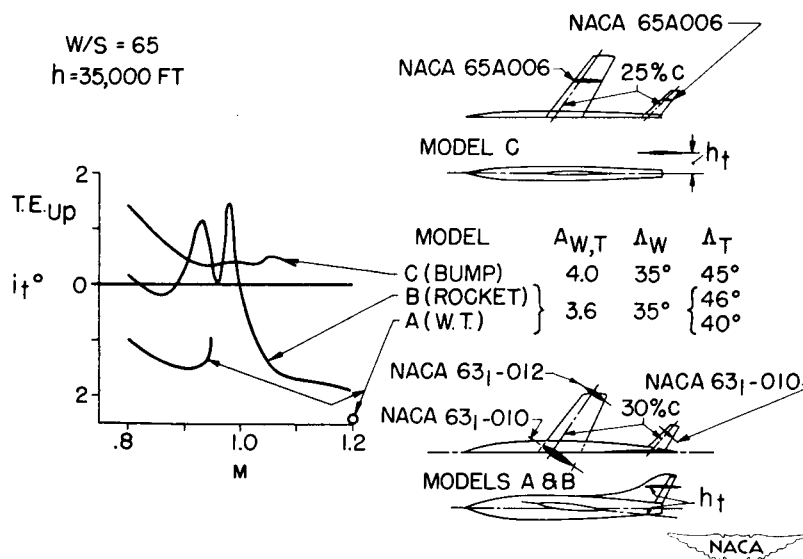


Figure 4.- Longitudinal stability characteristics of swept-wing configurations.

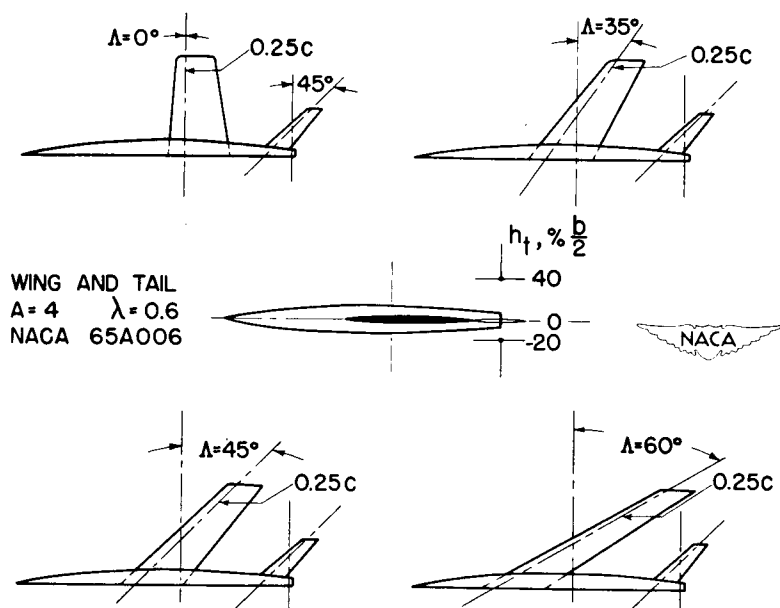


Figure 5.- Systematic-sweep series.

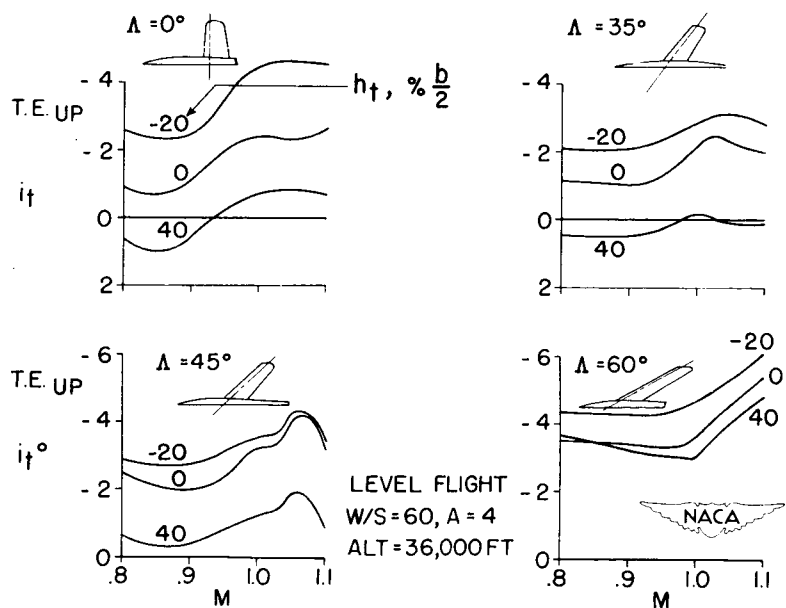


Figure 6.- Stability characteristics of systematic-sweep series.

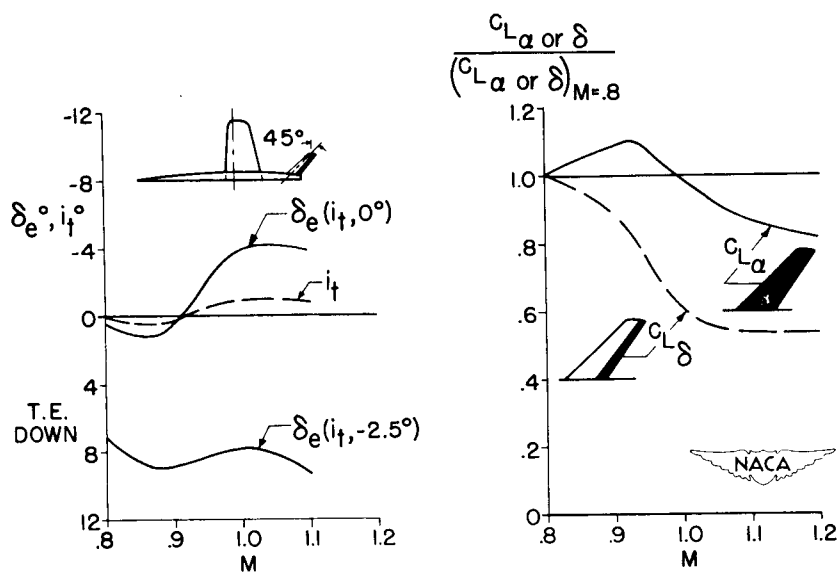
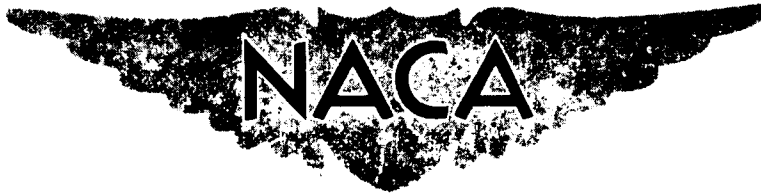


Figure 7.- Effect of elevator control on stability characteristics.



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# RESEARCH MEMORANDUM

A COMPARISON OF THE AERODYNAMIC CHARACTERISTICS AT  
TRANSONIC SPEEDS OF FOUR WING-FUSELAGE CONFIGURATIONS  
AS DETERMINED FROM DIFFERENT TEST TECHNIQUES

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## RESEARCH MEMORANDUM

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## SUMMARY

A comparison is made of the high-speed aerodynamic characteristics of a family of four wing-fuselage configurations of  $0^\circ$ ,  $35^\circ$ ,  $45^\circ$ , and  $60^\circ$  sweepback as determined from transonic-bump model tests in the Langley high-speed 7- by 10-foot tunnel, sting-supported model tests in the Langley 8-foot high-speed tunnel and in the Langley high-speed 7- by 10-foot tunnel, and rocket model tests conducted by the Langley Pilotless Aircraft Research Division. A complementary study of the effect of Mach number gradients and streamline curvature on bump results is also included.

It was found that qualitatively the data obtained from the various test facilities for the wing-fuselage configurations were in essential agreement as regards the relative effects of sweepback and Mach number except for drag at zero lift. Quantitatively, important differences were present. Lift-curve slopes as determined from bump tests tended to be somewhat higher than sting-model data indicated, with consequent differences occurring in drag due to lift. Fuselage-alone drag and wing-fuselage drag as obtained by the bump method were found to be unreliable particularly at Mach numbers above 1.0, but wing-alone drag data were found to be in surprisingly good agreement with available rocket model data throughout the transonic speed range. Aerodynamic-center position as determined from bump data was generally more rearward than sting data indicated, especially for the  $60^\circ$  configuration. Some of this effect has been attributed to the effects of Mach number gradients and flow curvature over the bump. It was evident, however, that for configurations for which aeroelastic effects were important the relative flexibility of the models used in the various facilities accounted for part of the differences in results.

## I N T R O D U C T I O N

As part of a transonic research program recommended by an NACA Special Subcommittee on Research Problems of Transonic Aircraft Design, a series of wing-fuselage configurations has been investigated in the Langley high-speed 7- by 10-foot tunnel using the transonic-bump method. Publication of these data was expedited despite uncertainties concerning the technique, because it was believed that such data would at least afford qualitative guidance to the aircraft designer. While direct comparison of these data with data obtained by other methods is still limited, recent investigations of several geometrically similar sting-supported models of this series in the Langley 8-foot high-speed tunnel and in the Langley high-speed 7- by 10-foot tunnel permit a comparison of the results at subsonic Mach numbers up to approximately 0.95 and at a supersonic Mach number of 1.2.

The purpose of this paper is to present a comparison of bump data and sting-supported model data for four wing-fuselage configurations of the primary transonic series, corresponding to the  $0^\circ$ ,  $35^\circ$ ,  $45^\circ$ , and  $60^\circ$  configurations described, respectively, in references 1, 2, 3, and 4. Included in the data for the  $45^\circ$  configuration are some drag comparisons at zero lift obtained from a rocket model investigation conducted by the Langley Pilotless Aircraft Research Division. Some effects of aeroelasticity are also discussed inasmuch as such effects are important in comparing data obtained from different test facilities.

The paper also presents the results of a complementary experimental investigation conducted in the Langley high-speed 7- by 10-foot tunnel to determine the extent to which bump model results are affected by such factors as flow curvature and Mach number gradients.

## S Y M B O L S

|              |                                                                                                                                                   |
|--------------|---------------------------------------------------------------------------------------------------------------------------------------------------|
| $C_L$        | lift coefficient (Twice semispan lift/ $qS$ or Total lift/ $qS$ )                                                                                 |
| $C_D$        | drag coefficient (Twice semispan drag/ $qS$ or Total drag/ $qS$ )                                                                                 |
| $\Delta C_D$ | total drag coefficient minus drag coefficient at zero lift                                                                                        |
| $C_m$        | pitching-moment coefficient, referred to $0.25\bar{c}$<br>(Twice semispan pitching moment/ $qS\bar{c}$ or<br>Total pitching moment/ $qS\bar{c}$ ) |
| $q$          | effective dynamic pressure over span of model, pounds per<br>square foot $\left(\frac{1}{2}\rho V^2\right)$                                       |

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|             |                                                                                                                                         |
|-------------|-----------------------------------------------------------------------------------------------------------------------------------------|
| S           | twice wing area of semispan model or wing area of full-span model                                                                       |
| $\bar{c}$   | mean aerodynamic chord (M.A.C.) of wing based on relationship<br>$\frac{2}{S} \int_0^{b/2} c^2 dy \quad (\text{using theoretical tip})$ |
| c           | local wing chord                                                                                                                        |
| b           | twice span of semispan model or span of full-span model                                                                                 |
| y           | spanwise distance from plane of symmetry                                                                                                |
| $\rho$      | air density, slugs per cubic foot                                                                                                       |
| V           | airspeed, feet per second                                                                                                               |
| M           | effective Mach number over span of model                                                                                                |
| $M_l$       | local Mach number                                                                                                                       |
| $M_a$       | average chordwise local Mach number                                                                                                     |
| R           | Reynolds number of wing based on $\bar{c}$                                                                                              |
| $\alpha$    | angle of attack, degrees                                                                                                                |
| L           | total lift load on wing, pounds                                                                                                         |
| $\Lambda$   | angle of sweepback, relative to $c/4$ line                                                                                              |
| $\Lambda_l$ | effective sweep angle at any spanwise station, referred to $c/4$                                                                        |

## Subscripts:

|       |                |
|-------|----------------|
| L = 0 | at zero lift   |
| m     | measured value |

## PART I - BASIC DATA COMPARISON

## SCOPE

A comparison is made at several subsonic Mach numbers (0.70, 0.80, 0.85, 0.90, and 0.93) and one supersonic Mach number (approx. 1.2) of the lift, drag, and pitching-moment characteristics of four wing-fuselage combinations of  $0^\circ$ ,  $35^\circ$ ,  $45^\circ$ , and  $60^\circ$  sweepback as determined from transonic-bump tests of semispan models and sting-supported wind-tunnel models. Most of the transonic-bump data used in the comparison have been taken from published papers (references 1 to 6), although additional bump data for the  $0^\circ$  and  $45^\circ$  configurations obtained subsequent to the original investigations are presented herein for the first time. The wind-tunnel data for the  $45^\circ$  configuration as determined in the Langley 8-foot high-speed tunnel were taken from reference 7. More complete data than that included herein for the other sting-supported wind-tunnel models are being obtained.

For the  $45^\circ$  configuration, a comparison of the drag at zero lift is also made with the results of a rocket model investigation of the same configuration (reference 8).

For the  $45^\circ$  and  $60^\circ$  configurations, static deflection measurements were made for simulated loading conditions and estimated aeroelastic corrections were evaluated for the lift-curve slope and aerodynamic-center position for the bump models and sting-supported models.

## REVIEW OF TEST TECHNIQUES

The test techniques employed in obtaining most of the experimental data included in this comparison involve two basic methods: (1) the transonic-bump method employing semispan models and (2) the conventional wind-tunnel method employing full-span sting-supported models. Representative Reynolds numbers for the various testing facilities are given in figure 1. For the most part, only the essential points of the techniques will be reviewed in this section, inasmuch as appropriate references will afford sources for details.

## Bump Tests

Method.- The transonic-bump method as used in the Langley high-speed 7- by 10-foot tunnel is described in reference 1. This method of testing involves the placement of a small model in the high-velocity-flow

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field generated over a curved surface and is an outgrowth of the NACA wing-flow method (reference 9). Typical contours of local Mach numbers in the vicinity of the model location on the bump are plotted in figure 2(a). Outlines of the  $\Lambda = 0^\circ$  and  $\Lambda = 60^\circ$  wings have been superimposed on this diagram in order to illustrate the extent of the spanwise and chordwise gradients in Mach number. In practice, no attempt is made to account for the effects of these gradients apart from evaluating the effective test Mach number on the basis of the relationship

$$M = \frac{2}{5} \int_0^{b/2} cM_a dy$$

The results of a complementary investigation conducted primarily to study additional effects that Mach number gradients may have on the aerodynamic characteristics as determined from bump tests are discussed in part II of this paper.

The model is attached to an electrical strain-gage balance mounted inside the bump and is so arranged that the gap between the bottom of the fuselage end plate and the bump surface is about 3/64 inch. The chamber containing the balance is sealed and vented to an orifice on the bump such that the static pressure in the chamber is roughly the same as that existing over the bump. A small opening of a size sufficient to allow an angle-of-attack range of about  $16^\circ$  is cut in the vicinity of the wing butt of the model. This opening is covered by the fuselage and its end plate. A photograph of a typical wing-fuselage model configuration mounted on the bump is shown in figure 3(c).

A unique arrangement (fig. 4) is used for measurements of the drag at zero lift in order to minimize leakage and to avoid the use of an end plate. In this arrangement, a foam-rubber wiper seal is attached to the model and rests against the underside of the cover plate and effectively blocks off the small gap (about 1/16 in.) around the root of the wing. The seal pressure must be carefully adjusted to avoid friction effects, but this method, in general, has been found to be more satisfactory for drag determinations than any unsealed arrangement.

Models.- A drawing of the four bump models used in this comparison is given in figure 5. Actually, data are being presented for six bump models, two sets each for the  $\Lambda = 0^\circ$  and  $45^\circ$  configurations and one set each for  $\Lambda = 35^\circ$  and  $60^\circ$  configurations. The additional models for the  $0^\circ$  and  $45^\circ$  configurations were constructed for the complementary investigation discussed in part II of this paper. These models were made of steel and hence were stiffer than the original  $\Lambda = 0^\circ$  and  $45^\circ$  models which were made of beryllium copper. The bump data for these additional tests have been included in the basic comparison, however, because of certain discrepancies in results.

The fuselage used was made of brass and its ordinates are given in table I. It was bent to conform to the bump surface as indicated on figure 5.

The various bump models for which data are presented, as well as the references from which the data were taken, are summarized in the following table:

| $\Lambda_c/4$<br>(deg) | Designation        | Construction<br>material  | Source                                         |
|------------------------|--------------------|---------------------------|------------------------------------------------|
| 0                      | Model 1<br>Model 2 | Beryllium copper<br>Steel | Reference 1<br>Unpublished                     |
| 35                     | Model 1            | Beryllium copper          | References 2 and <sup>a</sup> 5                |
| 45                     | Model 1<br>Model 2 | Beryllium copper<br>Steel | References 3 and <sup>a</sup> 6<br>Unpublished |
| 60                     | Model 1            | Steel                     | Reference 4                                    |

<sup>a</sup>Since the publication of references 2 and 3, drag at zero lift with the foam-rubber seal was obtained for this model and subsequently published in noted reference.

Corrections.- No jet-boundary corrections have been applied to any of the bump data. Corrections for fuselage base pressure were determined and found to be small. They have not been applied to the data.

Except for the measurements of the drag at zero lift, no tare corrections for the effect of the fuselage end plate or leakage have been applied to other components. Special tests, in which a few models were investigated at several angles of attack while mounted without end plates and with the foam-rubber seal attached, indicated small but inconsistent tare effects on lift, pitching moment, and drag due to lift.

#### Sting Tests

The sting methods used in the Langley 8-foot high-speed tunnel and the Langley high-speed 7- by 10-foot tunnel are basically similar and will be considered together in the following discussion. Additional details regarding the 8-foot sting method of testing may be found in reference 7..

Method.- The tests were conducted in closed-throat, single-return-type tunnels. In order to minimize interference effects between model and support the models were mounted on a sting which is attached to a supporting structure downstream of the test sections. A view of the models mounted in the tunnels is shown in figure 3. The Langley high-speed 7- by 10-foot-tunnel tests were run at various angles of attack through the Mach number range by employing angle-of-attack couplings, whereas the 8-foot high-speed-tunnel tests were conducted by changing angle of attack at a constant Mach number. The models are attached to internal strain-gage balances.

Models.- A drawing of the sting models together with pertinent dimensions is given in figure 6. The wings were constructed of aluminum alloy except the  $\Lambda = 35^\circ$  wing in the Langley 8-foot high-speed tunnel and the  $\Lambda = 45^\circ$  wing in the Langley high-speed 7- by 10-foot tunnel which were made with a steel insert and bismuth-tin filler. The fuselages were hollow and constructed of aluminum for the Langley high-speed 7- by 10-foot-tunnel configuration and of steel for the Langley 8-foot high-speed-tunnel configuration. They were designed by cutting off the rear portion of a body of revolution with a fineness ratio of 12 to form one with a fineness ratio of 10. (See table I.)

Corrections.- The method of reference 10 was used to correct for the effects of the model and its wake on Mach number and dynamic pressure. No corrections due to sting interference, which are believed to be small, have been determined or applied to the data. A representative account of interference effects as well as base-pressure measurements are presented in reference 7 for the 8-foot-tunnel models. Base-pressure corrections to drag were determined at zero angle of attack and have been applied to both the 7- by 10-foot-tunnel data and the 8-foot tunnel data. The drag at zero lift, therefore, corresponds to free-stream static pressure at the base of the fuselage. The correction to the drag coefficient was of the order of 0.002.

#### Effect of Flexibility

When the angle of sweepback is appreciably increased, the influence of wing flexibility on the aerodynamic characteristics becomes increasingly important. It follows, therefore, that comparisons of aerodynamic data for swept wings from different test facilities become exceedingly difficult because of differences in model construction, methods of mounting, and testing techniques. In order to obtain some idea of the effect of wing model flexibility on the aerodynamic results, two of the 7- by 10-foot-tunnel bump and sting wings ( $\Lambda = 45^\circ$  and  $60^\circ$ ) were loaded with a simulated elliptical-type loading along the quarter-chord line. The angle of attack at several spanwise stations was measured while the models were statically loaded. As an example of the results



obtained, a deflection diagram for the  $\Lambda = 60^\circ$  sting model is shown in figure 7. Correction factors for the lift-curve slope ( $C_{L\alpha}$ ) and the aerodynamic center ( $\partial C_m / \partial C_L$ ) have been evaluated from such diagrams for these wings and are presented in figure 8. The reason the generally weaker sting wings ( $\Lambda = 45^\circ$ , steel insert;  $\Lambda = 60^\circ$ , aluminum) have a smaller correction to  $C_{L\alpha}$  than the steel bump wings is attributed directly to the method of support. The 7- by 10-foot-tunnel sting wings are mounted about a point 10 percent semispan outboard of the fuselage center line, whereas the bump wings are mounted about a point inside the bump 25 percent semispan from the fuselage center line.

Inasmuch as the 8-foot-tunnel models are similar in construction and mounting to the 7- by 10-foot-tunnel models, the same correction factors are applicable to the 8-foot-tunnel results.

## RESULTS AND DISCUSSION

The results of the basic comparison are summarized in the following figures:

|                                                                                   | <u>Figure</u> |
|-----------------------------------------------------------------------------------|---------------|
| Basic wing-fuselage force and moment data:                                        |               |
| $\Lambda = 0^\circ$ configuration . . . . .                                       | 9             |
| $\Lambda = 35^\circ$ configuration . . . . .                                      | 10            |
| $\Lambda = 45^\circ$ configuration . . . . .                                      | 11            |
| $\Lambda = 60^\circ$ configuration . . . . .                                      | 12            |
| Zero-lift drag variation with Mach number:                                        |               |
| Bump and sting results for -                                                      |               |
| Wing-fuselage, all configurations . . . . .                                       | 13            |
| Fuselage alone . . . . .                                                          | 14            |
| Bump, sting, and rocket results for -                                             |               |
| $\Lambda = 45^\circ$ configuration . . . . .                                      | 15            |
| Basic aerodynamic parameter comparisons:                                          |               |
| $C_{L\alpha}$ and $\partial C_m / \partial C_L$ against M . . . . .               | 16            |
| Aerodynamic parameter comparisons with estimated aeroelastic corrections applied: |               |
| $C_{L\alpha}$ and $\partial C_m / \partial C_L$ against M . . . . .               | 17            |

The existence of nonlinear variations of  $C_L$  and  $C_m$  with  $\alpha$  decreases the significance of the comparison of the parameters  $C_{L\alpha}$

and  $\partial C_m / \partial C_L$  because of the difficulty in ascertaining slopes. For the comparisons presented in this paper the slopes were averaged through zero lift over an angle-of-attack range up to an angle at which obvious departures from linearity occurred. Accordingly, the slopes presented in this comparison paper may not be in exact quantitative agreement with those presented in the basic reference reports.

### $0^\circ$ Sweep Configuration

Lift.- The variation of lift with angle of attack is approximately the same for all models (fig. 9(a)) although quantitative differences are evident especially at the higher angles of attack. In the lower lift range the sting models exhibit a somewhat more rapid variation of lift-curve slope with Mach number than the bump model, particularly in the vicinity of the force break (fig. 16(a)). Also variations in lift coefficient, especially in the vicinity of  $C_{L_{max}}$ , are more pronounced for the sting models although the actual values of  $C_{L_{max}}$  are in fair agreement. It is perhaps to be expected that the agreement in lift is generally poorest wherever the lift changes are most rapid. Bump model 2 agrees a little better with the sting-mounted data than the original model 1 (reference 1). Some of the factors that are partly responsible for the lack of duplication in bump results are discussed in part II of this paper. In this regard, however, it should be noted that small differences are also evident in the two sets of sting data, particularly in regard to the variation of  $C_{L_\alpha}$  with  $M$  (fig. 16).

Pitching moment.- For the most part the variations of  $C_m$  with  $C_L$  (fig. 9(b)), particularly the rapid variation at high values of  $C_L$ , are in practical agreement for all models up to Mach numbers at which rapid rearward movement of aerodynamic center occurs (about  $M \approx 0.85$  in fig. 16(b)). As regards aerodynamic-center position, the bump models are in good agreement with the 7- by 10-foot-tunnel sting data, whereas the 8-foot-tunnel sting results indicate a more forward aerodynamic-center location at Mach numbers below 0.85. However, above  $M = 0.85$  the bump results and the 8-foot-tunnel sting results are in good agreement throughout the subsonic Mach number range as well as in the upper transonic speed range near  $M = 1.2$ .

Drag.- Drag due to lift (fig. 9(c)) is in good agreement for all models except in the maximum lift range. The lower drags indicated by the 8-foot-tunnel sting model at the higher lift coefficients are a result of the lower angle of attack required to sustain these lift coefficients.

The variation of  $C_{D_{L=0}}$  with Mach number for the  $\Lambda = 0^\circ$  wing-fuselage configurations (fig. 13) appears to be in good agreement, but

the bump models exhibit a considerably higher value of drag throughout the Mach number range than do the sting models. It was the high drag obtained with the bump model in the original investigation (reference 3), even when a sponge wiper seal was used to avoid leakage and end-plate effects, that prompted the construction and investigation of bump model 2. The results of the two bump tests, however, seem to be in excellent agreement. Roughness applied to the bump model wings in an effort to simulate the type of boundary-layer transition likely to occur at a high Reynolds number was of little value for this model and the application of roughness resulted in even higher values of drag.

The high absolute drag obtained with the bump model is attributable, in part, to the high drag obtained with the fuselage alone. Fuselage-alone drag data as obtained by the bump method are compared with fuselage-alone drag data from other sources in figure 14. Not only does the fuselage-alone bump drag appear to be about twice as great as that obtained from wind-tunnel and drop tests (reference 11) but the variation with Mach number above  $M = 1.0$  appears to be unreliable. For this reason, the variation of drag with Mach number at Mach numbers in excess of 1.0 for any of the bump wing-fuselage configurations should be used with caution. It is believed that the high fuselage drag is largely a result of the gap between the fuselage and the bump surface inasmuch as the drag caused by the base pressure was found to be small.

### 35° Sweep Configuration

Lift.- In regard to the variation of lift with angle of attack, the bump results and the 7- by 10-foot-tunnel sting results are in good agreement (fig. 10(a)) and both exhibit higher lift-curve slopes than the 8-foot-tunnel sting model (fig. 16(a)). The differences in lift behavior exhibited by the two sting models in the lower angle-of-attack range are somewhat surprising even when the slight changes in wing sweep angle ( $2.4^\circ$ ) are considered (fig. 6). The reasons for these discrepancies in lift behavior are not known at present. It is evident also that while the lift variations at the higher lift coefficients are similar for the bump model and the 8-foot-tunnel sting model, the rapid changes in lift for the 8-foot-tunnel sting model are delayed to a higher angle of attack, perhaps because of the higher Reynolds number of the 8-foot-tunnel data.

Pitching moment.- The bump model and the two sting models exhibit similar trends in pitching-moment behavior in the lower lift range but the lift coefficients at which rapid changes in pitching-moment behavior occur are higher for the 8-foot-tunnel sting model (fig. 10(b)). The bump data give an aerodynamic-center position about 5 percent more rearward than the sting models at the lower Mach numbers (fig. 16(b)) and indicate a less rapid rearward movement with Mach number than was

obtained with the 8-foot-tunnel sting model. At supersonic Mach numbers the bump and sting results are in fair quantitative agreement.

Drag.- The drag due to lift (fig. 10(c)) indicates similar trends for all models and reflects the differences in lift behavior previously noted. The variation of  $C_{DL=0}$  with  $M$  (fig. 13) for the bump model is more rapid below the Mach number for drag rise and less rapid in the vicinity of drag rise than that for the sting models. It is of interest to note, however, that the drag variation obtained with the bump wing-fuselage configuration is similar to that obtained for the bump wing alone (reference 5).

#### 45° Sweep Configuration

Lift.- The lift characteristics of the two sting models are in very good agreement (fig. 11(a)), particularly in regard to the variation of the mean lift-curve slope with Mach number (fig. 16(a)). The 45° configuration is flexible enough to require consideration of aeroelastic effects on lift-curve slope. These corrections have been estimated for the 7- by 10-foot-tunnel sting model and for bump model 2 (fig. 8) and have been applied to the data in figure 17(a). Inasmuch as the 7- by 10-foot-tunnel sting data and the 8-foot-tunnel sting data are in good basic agreement, the corrected values given in figure 17(a) can be assumed to apply to the 8-foot-tunnel data also.

The bump models appear to be in general qualitative agreement with the sting results. It will be noted, however, that the bump models give somewhat higher lift-curve slopes (fig. 16(a)) and indicate a more linear variation of lift with angle of attack in the higher angle-of-attack range (fig. 11(a)).

Pitching moment.- The pitching-moment behavior exhibited by the various models is perplexing (fig. 11(b)). Even the sting models indicate some differences, especially in regard to aerodynamic-center position (fig. 16(b)). The development of an unstable pitching-moment variation at a very low value of  $C_L$  for bump model 1 was regarded with suspicion when originally obtained (reference 3), but check tests made at that time produced similar results. Subsequently, bump model 2 was constructed for the bump-wall comparison discussed later in this paper and gave the type of pitching-moment variation shown in figure 11(b). It is evident that for bump model 2, the onset of the unstable pitching-moment variation has been delayed to higher lift coefficients and is in better accord with 8-foot-tunnel sting data in this respect. The aerodynamic center is, however, somewhat more rearward than was obtained for other models (fig. 16(b)). The pitching-moment data in reference 12 indicate that, in the Reynolds number range

of the bump tests, surface conditions can result in a forward movement of the aerodynamic center of several percent; whereas, at the higher Reynolds numbers (greater than  $1 \times 10^6$ ), roughness has little effect. It is interesting to note that the application of roughness to the 8-foot-tunnel sting model (reference 7) and to the 7- by 10-foot-tunnel sting model (unpublished) also had little effect on aerodynamic-center position.

The corrections to the aerodynamic-center location attributable to aeroelastic effects (figs. 8 and 17(b)) appear to be rather small for this configuration and are not an important factor in explaining the discrepancies obtained.

Drag.- Drag due to lift (fig. 11(c)) is in good agreement for the two sting models. Bump model 2 is in fair agreement with the sting results, but the originally published drag data for bump model 1 indicate definitely lower drags in certain portions of the lift range, particularly at the lower Mach numbers.

The differences observed in the zero-lift drag variation with Mach number for the two bump models (fig. 13) are probably a result of changes in fuselage drag inasmuch as the wing-alone drag for the two models is in excellent agreement (compare fig. 15 of reference 6 with fig. 21 of this paper). The inconsistencies in these wing-fuselage drag data again emphasize the difficulty encountered in measuring the drag of half-bodies of revolution by the bump method.

A comparison of the drag at zero lift utilizing the results of rocket model tests of the  $45^\circ$  configuration, reference 8, is presented in figure 15. The abnormally high fuselage-alone drag obtained on the bump (fig. 14) makes a comparison in the transonic range difficult even when compared on the basis of wing-fuselage minus fuselage drag. It appears that the bump model does not reflect the peak drag obtained with the rocket configuration in the vicinity of  $M = 0.98$ . This peak drag has been traced to interference effects between the wing and fuselage (reference 8). Inasmuch as the fuselage-alone drag is considered questionable from bump tests, it is not surprising that such interference effects are not observed in the bump data.

### $60^\circ$ Sweep Configuration

Lift.- The lift characteristics are in essential agreement (fig. 12(a)) although the bump model fails to reveal the nonlinearities exhibited by the sting models at the low angles of attack. This fact is perhaps responsible for the slightly higher values of lift-curve slope obtained for the bump model (fig. 16(a)). The corrections for flexibility

do not appreciably affect the comparison in regard to lift-curve slope (figs. 8 and 17(a)).

Pitching moment.- It is the pitching-moment behavior of the  $60^\circ$  configuration that exhibits the largest discrepancies encountered in this comparison. The sting models are in fair agreement at the higher angles of attack (fig. 12(b)) but exhibit different pitching-moment slopes in the lower lift range (fig. 16(b)). These differences are small, however, compared to the extremely rearward aerodynamic-center position indicated from bump data. The bump data also indicate unstable pitching moments occurring at considerably lower lift coefficients than the sting models. This result does not appear to be a sole consequence of the low Reynolds number of the bump tests inasmuch as unpublished tests of the 7- by 10-foot sting model in the Langley 300 MPH 7- by 10-foot tunnel at a Reynolds number of about 500,000 indicated similar characteristics to those shown for the 7- by 10-foot sting model at  $M = 0.7$ .

It is believed that the indicated differences for the sting models at the low lift coefficients may not really be present. It was necessary to fair the 8-foot sting data with considerably fewer points than were available for the 7- by 10-foot sting data, and wherever nonlinear variations occur a great many points are necessary to define the character of the nonlinearity. Essentially, therefore, the sting data are in good agreement and the important feature of the comparison is found in the considerably more rearward aerodynamic-center position obtained with the bump model. A considerable portion of the discrepancy in aerodynamic-center location is attributable to the larger aeroelastic effects experienced by the sting model (fig. 8). The estimated effect of the wing flexibility on the sting model results was to move the aerodynamic-center appreciably rearward as shown in figures 16(b) and 17(b). It will be noted, however, that the flexibility for the bump model is rather small and that the aerodynamic center indicated for the bump model is still considerably more rearward than for the sting model even when the flexibility of both models is considered.

It is believed that the more rearward aerodynamic-center location obtained on the bump model for highly swept wings is also closely related to the effect of the bump curvature. For the  $60^\circ$  bump model the spanwise variation of sweep angle due to the curvature of flow is shown in figure 18. Thus, the root sections are operated in excess of  $60^\circ$  whereas the tip sections are operated at sweep angles of less than  $60^\circ$ . The effect of this sweep variation on the span loading has been estimated and it has been determined that, although the effect of this sweep variation on lift-curve slope was small, the aerodynamic-center position was moved about 5 percent of the mean aerodynamic chord rearward. This correction has been noted on figure 17(b) for  $M = 0.7$  and it is seen that much of the discrepancy in the pitching-moment

slope existing between the sting and bump results at this Mach number appears to be accounted for. Although it is realized that these factors may not be entirely responsible for all the discrepancies attributed to them, the indications are that: (1) highly swept models tested on the bump used in this comparison are apt to result in considerably further rearward movements of the aerodynamic center than would be anticipated and that (2) large differences in aeroelastic effects can appreciably modify comparisons of data obtained in different test facilities.

Drag. - Drag due to lift is in fair agreement (fig. 12(c)) but the drag at zero lift (fig. 13) is considerably higher in absolute magnitude for the bump model than that obtained on either of the sting models, although the results do exhibit the very small variation with Mach number that would be expected for this wing.

#### General Remarks on Data Comparison

Despite the differences that have been noted in the comparison of the separate configurations, a cross-comparison of the data for the  $0^\circ$ ,  $35^\circ$ ,  $45^\circ$ , and  $60^\circ$  configurations indicates that the bump model results exhibit about the same qualitative effects of sweepback and Mach number on the aerodynamic characteristics of the wing family except for drag at zero lift. Important quantitative difference in the results are evident, however. In general, wherever sudden changes in lift, drag, and pitching moment occurred, the bump model results indicated less rapid changes with Mach number and angle of attack than the sting model results. (See, for example, figs. 9, 13, 16, and 17.) The bump data generally resulted in higher lift-curve slopes than were obtained from sting data, and the variation of lift-curve slope with Mach number was less rapid than sting data indicated. Drag at zero lift as obtained from the bump data for the wing-fuselage combinations and for the fuselage alone does not appear to be reliable as regards either the absolute value of drag or the rate of drag increase with Mach number in the neighborhood of the drag rise Mach number. It will be shown subsequently in this paper that this result is largely attributable to fuselage drag results. On the other hand, drag due to lift was generally in fair agreement for the bump and sting models except where discrepancies existed in the angle of attack required to support the same lift. The position of the aerodynamic center as determined from bump tests appears to be more rearward than sting model data indicate, especially at the higher sweep angles, but differences in the flexibility of the models used make comparisons of aerodynamic center position difficult because of aeroelastic effects.

Although distinguishing trends are evident in the data, the results of a comparison of only four models do not permit detailed conclusions to be drawn regarding the reliability of bump data in general. It appears almost essential to examine each model individually because of the many factors involved in comparing the results obtained from one technique with those of another.

## General Remarks on Test Methods

In comparing the results obtained with similar models in different test facilities, it is usually not possible to control test conditions closely enough so that differences in results can be attributed to a unique factor. A list of some of the factors that must be considered in evaluating the results obtained by various test methods would include the following items:

1. Mach number gradients
    - (a) Spanwise
    - (b) Chordwise
  2. Flow curvature
  3. Boundary layer at model
  4. Flow leakage about model
  5. End-plate conditions
  6. Flow steadiness
  7. Humidity conditions
  8. Reynolds number of test
  9. Accuracy of model construction
  10. Flexibility of model
- } Reflection-plane technique

All of these items are perhaps not of equal importance for all test methods, but each test method must be examined for those factors most likely to influence the results obtained by that method. Thus, it is evident from the Mach number gradients shown in figure 2(a) that items 1 and 2 constitute important defects in the bump method of testing, at least for the particular bump referred to in this paper. Items 3, 4, and 5 are important considerations for any method utilizing the reflection-plane technique and perhaps assumed more critical roles in bump testing because of the presence of items 1 and 2. Items 6, 7, 8, 9, and 10 are important considerations in any test method. Little is known concerning the effects of flow steadiness, item 6. Humidity conditions, item 7, are not believed to be an important factor in the comparison presented in this paper because of the elevated temperatures at which the wind tunnels were operated (reference 7). The low Reynolds number of bump tests, item 8, has always been considered one of the major deficiencies of this method of testing (fig. 1). Accuracy of model construction, item 9, becomes relatively more important in bump



investigations because the small size of the models requires precision workmanship not only in regard to constructing models but in positioning them for tests.

It has not been possible to control test conditions on the bump closely enough to permit isolation of the influence of the many factors involved. It has been possible, however, to examine the importance of Mach number gradients and flow curvature (items 1 and 2) by investigating the bump models on a reflection plane in the Langley high-speed 7- by 10-foot tunnel. The test conditions for this arrangement were practically the same as the bump test conditions except that there was no flow curvature and relatively small Mach number gradients compared to the bump. It was not possible, however, to obtain as high Mach numbers with this arrangement as was possible with the bump method.

An investigation to examine the effects of Mach number gradients and flow curvature on the bump results is described in the following section.

## PART II - EFFECTS OF MACH NUMBER GRADIENTS AND FLOW CURVATURE ON BUMP RESULTS

### DESCRIPTION OF WALL REFLECTION-PLANE TECHNIQUE

Models and method.- The two models that were used in the bump-wall investigation were the  $0^\circ$  and  $45^\circ$  models shown in figure 5. The wings of these models were of steel and were especially constructed for this investigation. The models were tested on the bump and on the wall plate as wings alone and in combination with the fuselage. Figure 3(d) shows one of the models mounted on the reflection plate. The plate was fastened to the wall of the Langley high-speed 7- by 10-foot tunnel and was located so as to bypass the tunnel boundary layer. The length of the plate was such that the boundary layer at the model position was approximately the same as that existing at the model location on the bump. Every effort was made to make the wall and bump installation similar by duplicating details such as mounting brackets, end-plate, and gap conditions. For the wall tests, the fuselage was not curved as shown in figure 5.

Test conditions.- The velocity field in the vicinity of the models is shown in figure 2(b). For the Mach numbers indicated, it is evident that the velocity gradients are very much less than those occurring on

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the bump at similar Mach numbers and are principally chordwise gradients, whereas the bump gradients are predominantly spanwise. For Mach numbers below  $M = 0.95$ , the flow field was essentially free of any velocity gradients. The wall reflection-plane method is essentially the same as the bump method. The main tunnel flow remains subsonic for all test Mach numbers below  $M = 1.08$  at which value a Mach number of 1.0 is obtained on the opposite wall from the plate. By testing these models on the wall plate, it was hoped that most of the itemized factors would be duplicated in the wall and bump tests except items 1 and 2. Actually, it was not possible to achieve this end completely. Nevertheless, it was believed that by having the same Reynolds number for both tests (see fig. 1), one of the principal uncertainties in the data comparison would be eliminated.

### RESULTS AND DISCUSSION

A comparison of the results obtained by the wall reflection-plane method and the bump method is contained in the following figures:

|                                                             | Figure |
|-------------------------------------------------------------|--------|
| Wing-alone and wing-fuselage characteristics:               |        |
| $\Lambda = 0^\circ$ configuration . . . . .                 | 19     |
| $\Lambda = 45^\circ$ configuration . . . . .                | 20     |
| Variation of drag at zero lift with Mach number . . . . .   | 21     |
| $CL_\alpha$ and $\partial C_m / \partial C_L$ against $M$ : |        |
| $\Lambda = 0^\circ$ . . . . .                               | 22(a)  |
| $\Lambda = 45^\circ$ . . . . .                              | 22(b)  |
| Comparison with 8-foot sting data . . . . .                 | 23     |

The bump data presented in the preceding figures are the same as that presented for bump model 2 in figures 9 and 11.

#### $0^\circ$ Sweep Configuration

Lift.- Similar trends in the over-all variation of lift with angle of attack are evident (fig. 19(a)) although bump data consistently indicate a somewhat less rapid variation of lift at the angles of attack near  $CL_{max}$ . It has been noted previously (fig. 9(a)) that this wing appears to be particularly sensitive at the high angles of attack, so that differences in results between the bump and wall tests in this angle-of-attack range are not too surprising.

For the wing alone, the lift-curve slopes as determined from wall data (fig. 22(a)) are in excellent agreement with theoretical values determined from reference 13. It will be noted, however, that the lift-curve slopes for both the wing and wing-fuselage configurations as determined from bump data are somewhat higher than the values determined from the wall data.

Pitching-moment.- The pitching-moment characteristics are in good qualitative agreement, particularly for the wing-fuselage configuration and especially as regards the lift coefficient at which rapid changes in lift coefficients occur (fig. 19(b)). Bump data appear to give a slightly more rearward aerodynamic-center location (fig. 22(a)), although estimated theoretical values of aerodynamic center overlap both sets of wing-alone data. The theoretical aerodynamic-center locations were approximated by applying a correction factor for the effect of chordwise loading, as estimated from reference 14, to the values as determined from reference 13.

The variation of  $\partial C_m / \partial C_L$  with Mach number is in good agreement for both the wing-alone and the wing-fuselage configurations. It is of interest that both test methods indicated about the same change in aerodynamic-center position attributable to the fuselage despite the fact that the fuselage was curved for the bump tests.

Drag.- The drag due to lift (fig. 19(c)) is in good agreement except at the higher values of  $C_L$  where the higher angles of attack required for the wall models resulted in greater drag increments.

It was in the drag at zero lift that the effects of the Mach number gradients and curvature have been expected to be most evident (fig. 21). The combined effect of these factors resulted in somewhat higher drags for the bump results for both wing alone and wing-fuselage configurations but the rate of drag rise in the transonic range was only slightly less rapid than that obtained on the wall. The wing-alone wall results are in good agreement with the drag data determined from rocket model tests made by the Langley Pilotless Aircraft Research Division of a wing of zero sweep, taper ratio 1, and aspect ratio 3.7 mounted on a cylindrical fuselage, for which interference effects are small (unpublished).

It will be noted also that the effect of the fuselage on the drag at zero lift is essentially the same for the bump and wall tests. The fuselage-alone drag as measured on the wall, is in agreement with the bump fuselage data (fig. 14), except above  $M = 1$  where the wall fuselage drag increases more rapidly because of the increased longitudinal velocity gradient at these Mach numbers (fig. 2(b)). In any event, neither the fuselage-alone nor the wing-fuselage drag is very reliable. The wing-fuselage drag indicated for the sting models (fig. 13) is more nearly obtained if the wing-alone drag (fig. 21) is added to the sting or drop-test fuselage drag shown in figure 14.

### 45° Sweep Configuration

Lift.- The agreement in lift characteristics for the wing-alone data for the bump and wall model is very good both with respect to lift-curve slope (fig. 22(b)) and the lift behavior at the higher angles of attack (fig. 20(a)), although both sets of data give somewhat greater values of lift-curve slope than would be determined from theory (reference 13). The results for the wing-fuselage combination also show excellent agreement as regards lift-curve slope (fig. 22(b)) although in this case differences in lift behavior are evident at the higher angles of attack, perhaps indicating that wing-fuselage interference effects are critical.

Pitching moment.- The pitching-moment behavior for both wing-alone and wing-fuselage configurations is generally in very good agreement as regards the character of variation of  $C_m$  with  $C_L$ , particularly at the higher values of  $C_L$  (fig. 20(b)); however, the bump results do indicate a more rearward aerodynamic-center position (fig. 22(b)) of almost a constant amount throughout the Mach number range as compared with either wall data or theory. As for the  $\Lambda = 0^\circ$  wing, the theoretical values have been approximated by applying chordwise correction factors estimated from reference 14 to the aerodynamic-center positions determined from reference 13.

The fact that the difference in aerodynamic-center location is almost a constant value at all Mach numbers points to the possibility of an error in positioning of the models relative to the axes of moments. It would take, however, a relative error of about 0.10 inch to account for this difference and the models are believed to be located correctly to within at least 0.01 inch. Some of the differences, therefore, might be attributed to the effect of Mach number gradients. The effect of fuselage curvature would not appear to be so important in this case inasmuch as the same result was obtained for the wing-alone tests.

Drag.- The drag due to lift agrees well in the low lift range but differences are evident at the higher value of  $C_L$  (fig. 20(c)). However, it is especially difficult to measure accurate values of drag at the higher values of  $C_L$  because of flow unsteadiness. Therefore, some of the drag differences shown may well be within experimental accuracy.

The drag at zero lift for the 45° configuration (fig. 21) shows similar trends to those observed for the 0° configuration and the notable effect of sweepback in diminishing the rate of drag rise is reflected in both sets of data. The drag determined for the wall tests is slightly lower at the lower Mach numbers but the rate of drag rise for the 45° configuration is in excellent agreement for both models, indicating little effect of Mach number gradient or curvature. The agreement between the wing-alone results and the rocket data is noteworthy,

particularly in view of the differences in Reynolds number. The rocket drag data used in this comparison are unpublished and differ from those presented in reference 8 (and used in fig. 15), in that a fuselage having a cylindrical section at the wing root was used instead of the fuselage described by table I.

#### Comparison with Sting Data

A comparison at several subsonic Mach numbers of the bump, wall, and 8-foot sting data is presented in figure 23 for the  $0^\circ$  and  $45^\circ$  wing-fuselage configurations.

For the  $0^\circ$  configuration, the lift characteristics, particularly at high angles of attack, are somewhat different for all three methods. For the  $45^\circ$  configuration, however, the agreement in lift between 8-foot sting results and wall results is very good. The pitching-moment characteristics exhibited by the wall model appear to agree with the 8-foot sting results for the  $45^\circ$  configuration, particularly as regards (1) the lift coefficient at which the moment curve breaks unstable and (2) the aerodynamic-center position and its change with Mach number. The wall data for the  $0^\circ$  configuration, on the other hand, are in no better agreement with the 8-foot sting results than the bump data.

The drag due to lift does not show any extreme differences (fig. 23(c)) and, as far as the drag at zero lift is concerned, neither the bump nor wall data for wing-fuselage drag can be considered reliable because of the extremely high drag obtained with the fuselage alone (fig. 14).

#### C O N C L U S I O N S

Based on a study of the aerodynamic characteristics of a family of four wing-fuselage configurations of  $0^\circ$ ,  $35^\circ$ ,  $45^\circ$ , and  $60^\circ$  sweepback as determined from bump model tests, sting-supported wind-tunnel model tests, and a few rocket model tests, the following conclusions are indicated:

1. Qualitatively, the bump model results and the sting model results indicated about the same relative effects of sweepback and Mach number on the aerodynamic characteristics of the wing-fuselage family except for drag at zero lift. Quantitatively, significant differences in results were evident. In general, wherever sudden changes in lift, drag, and pitching moment occurred, the bump model results indicated less rapid changes with Mach number and angle of attack than the sting model results.

2. Lift-curve slopes as determined from bump model tests were generally a little higher, and the variation with Mach number somewhat less pronounced, than were obtained from sting-model tests.

3. Drag due to lift was generally in fair agreement for the bump and sting models, but discrepancies were evident whenever differences occurred in the angle of attack required to support the same lift.

4. Drag at zero lift as determined by bump tests for either the fuselage alone or for the wing-fuselage combinations, is considered to be unreliable because of exhibited discrepancies with the results of sting model tests and rocket model tests, particularly at Mach numbers above 1. However, wing-alone drag as determined from bump models appeared to agree well with available rocket model data throughout the transonic range.

5. Aerodynamic-center position as determined from bump data was generally more rearward than was found from sting model results, particularly for the  $60^\circ$  sweep configuration.

6. A study of the effect of Mach number gradient and bump curvature on the bump results indicated that the principal effect of these factors on the wings investigated was to move the aerodynamic-center position somewhat more rearward. No consistent effect of these variables was noticed on other aerodynamic parameters.

7. It was important in comparing the results obtained in the different test facilities to consider the relative flexibility of the model installations because the aeroelastic effects exhibited were sufficiently different in some cases to affect the comparison, particularly in regard to aerodynamic-center position.

Langley Aeronautical Laboratory  
National Advisory Committee for Aeronautics  
Langley Air Force Base, Va.

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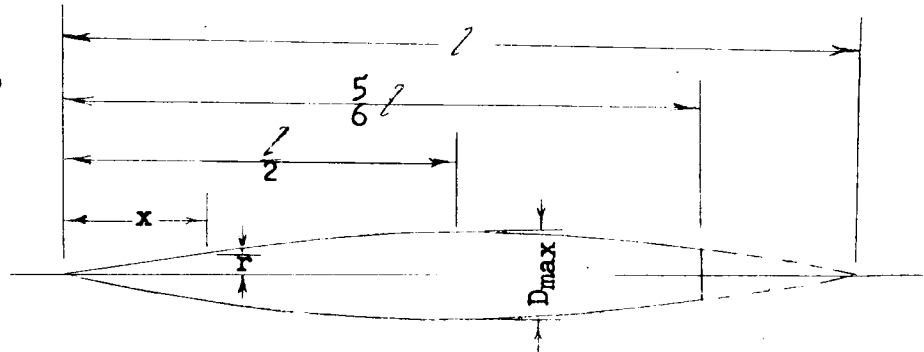
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TABLE I.- TRANSONIC FUSELAGE ORDINATES

[Basic fineness ratio 12; actual fineness ratio 10 achieved by cutting off the rear one-sixth of the body]



| Ordinates               |        |        |        |
|-------------------------|--------|--------|--------|
| $x/l$                   | $r/l$  | $x/l$  | $r/l$  |
| 0                       | 0      | 0      | 0      |
| .005                    | .00231 | .4500  | .04143 |
| .0075                   | .00298 | .5000  | .04167 |
| .0125                   | .00428 | .5500  | .04130 |
| .0250                   | .00722 | .6000  | .04024 |
| .0500                   | .01205 | .6500  | .03842 |
| .0750                   | .01613 | .7000  | .03562 |
| .1000                   | .01971 | .7500  | .03128 |
| .1500                   | .02593 | .8000  | .02526 |
| .2000                   | .03090 | .8338  | .02000 |
| .2500                   | .03465 | .8500  | .01852 |
| .3000                   | .03741 | .9000  | .01125 |
| .3500                   | .03933 | .9500  | .00439 |
| .4000                   | .04063 | 1.0000 | 0      |
| L. E. radius = 0.0005 l |        |        |        |

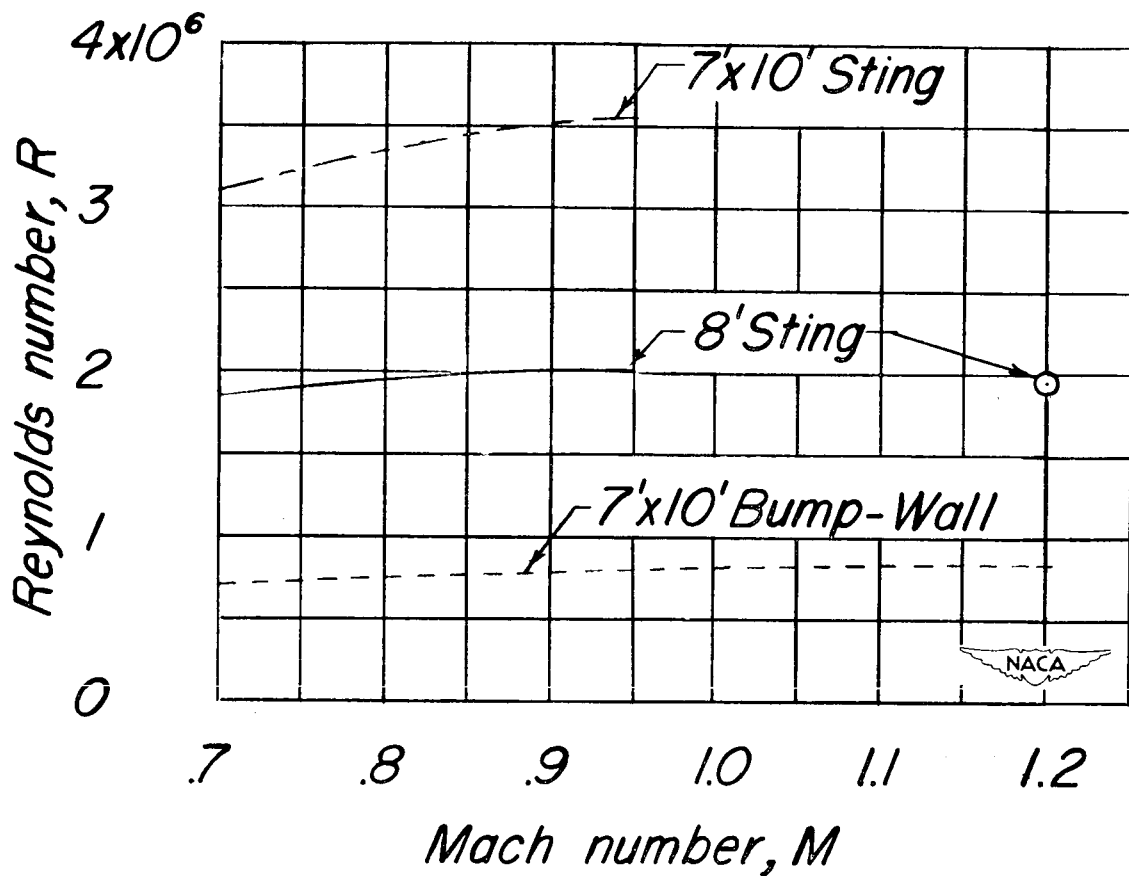
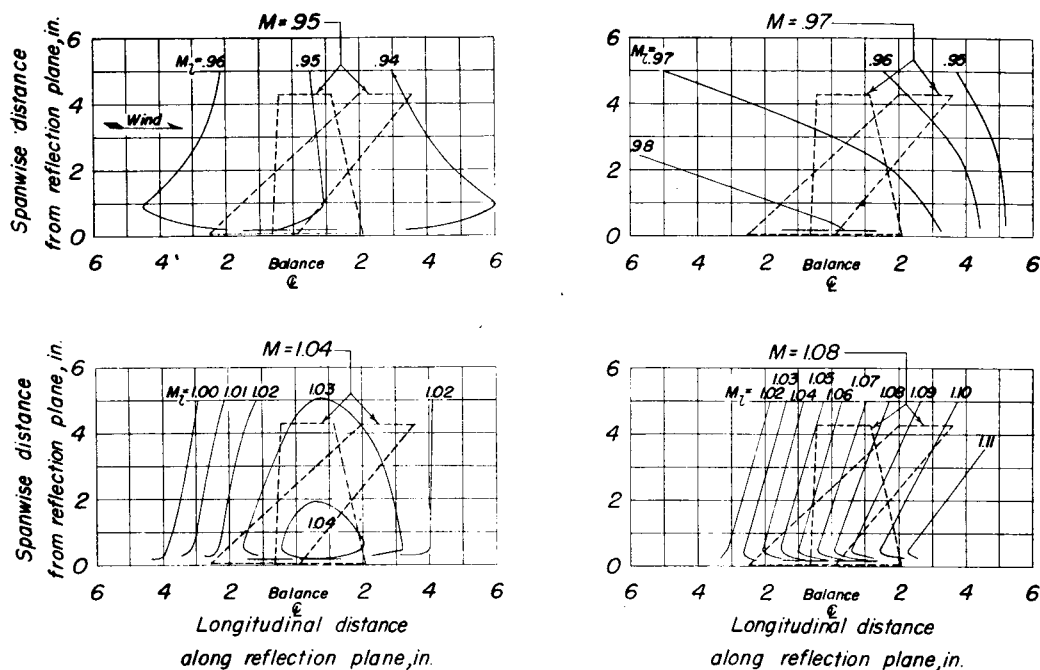
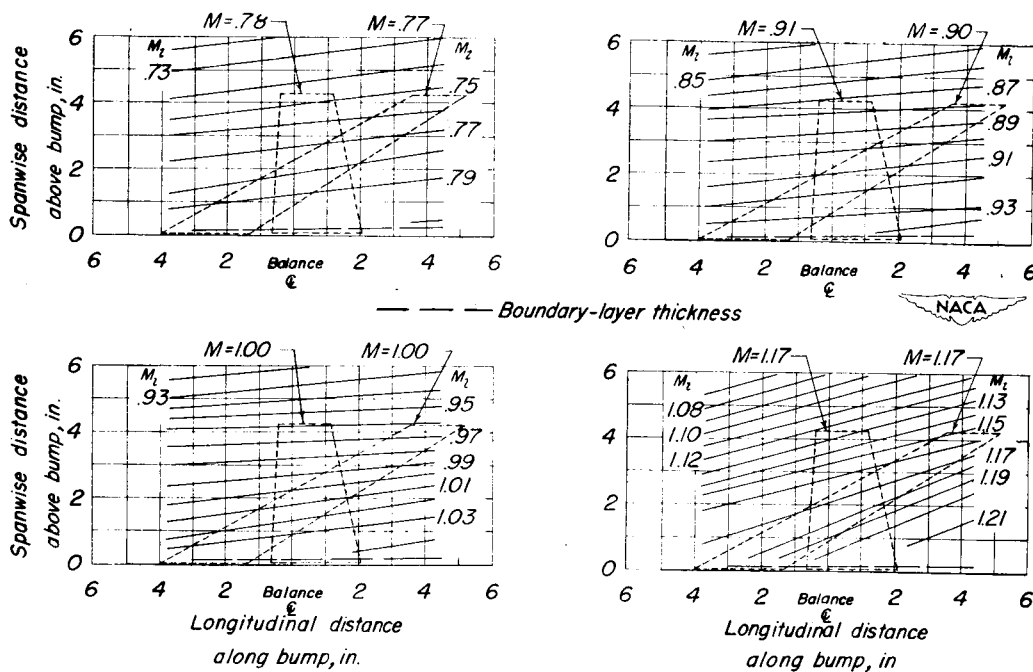


Figure 1.- Variation of Reynolds number with Mach number for various test techniques.



(b) Sidewall reflection plane.



(a) Transonic Bump

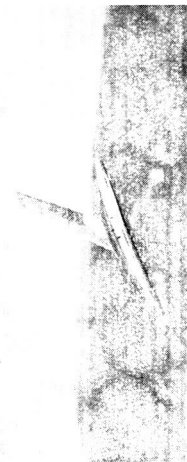
Figure 2.- Typical Mach number contours in the region of the model location for two transonic test techniques used in the Langley high-speed 7- by 10-foot tunnel.



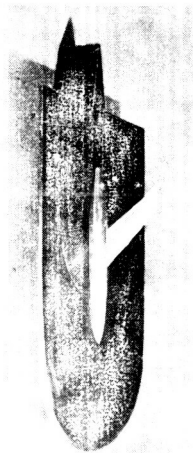
(a) 7'x10' Sting.



(b) 8' Sting.



(c) 7'x10' Bump.



(d) 7'x10' Wall reflection plane.



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Figure 3.- Photographs of a wing-fuselage model mounted in four different test facilities.

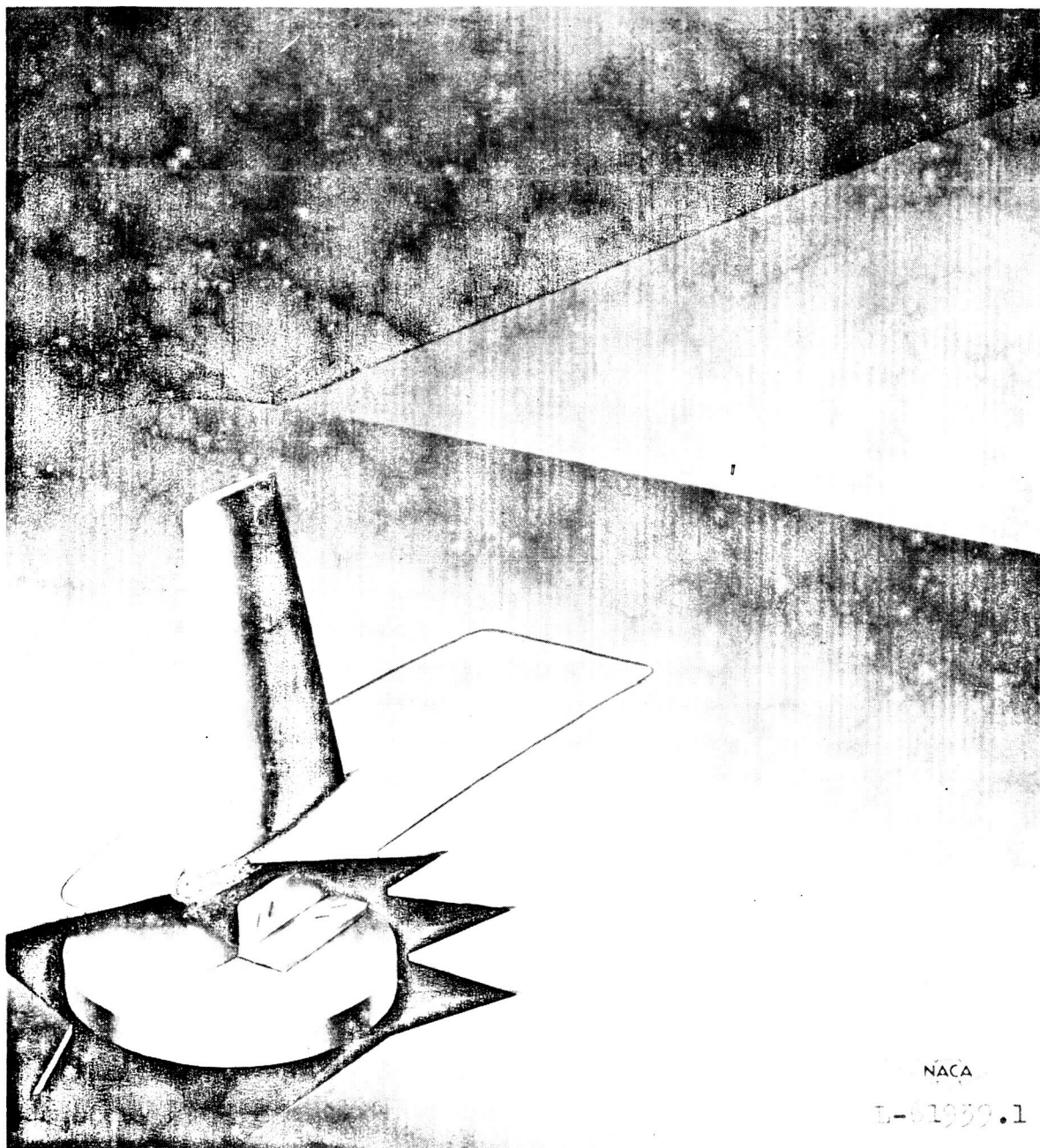
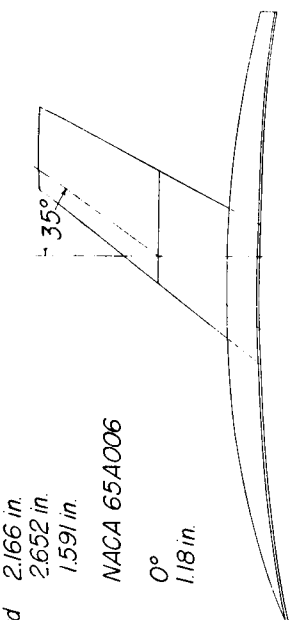
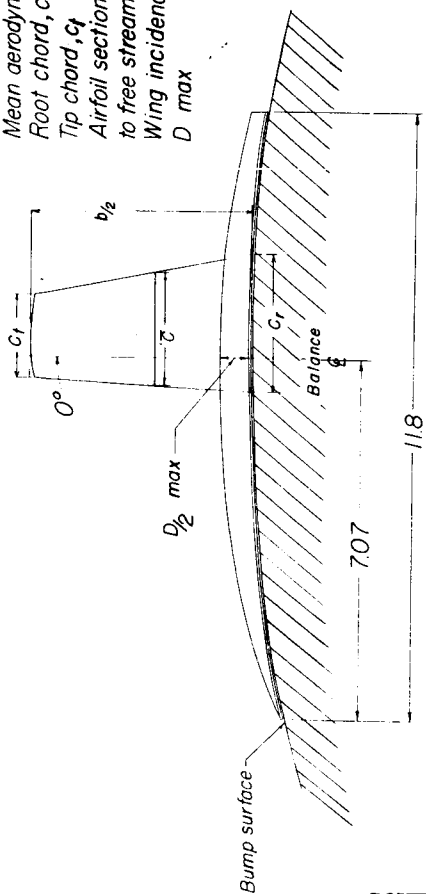


Figure 4.- A view of a wing model mounted on the transonic bump showing the foam-rubber wiper seal.

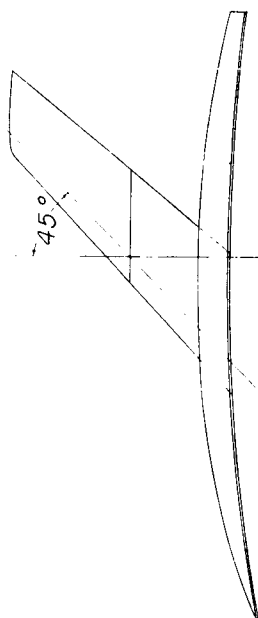
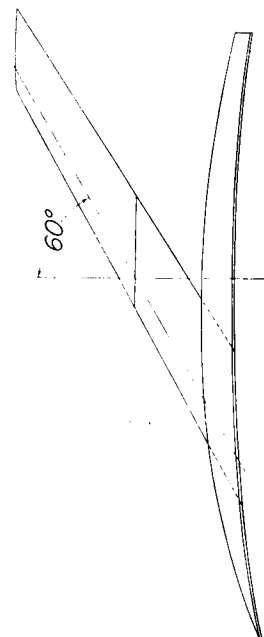
Aspect ratio 4.0  
 Taper ratio 0.6  
 Wing area 0.125 sqft  
 Wing span 8.486 in.  
 Mean aerodynamic chord 2.166 in.  
 Root chord,  $c_r$  2.652 in.  
 Tip chord,  $c_t$  1.591 in.  
 NACA 65A006  
 Airfoil section parallel to free stream.  
 Wing incidence  $0^\circ$   
 D max 1.18 in.



Note:  
 Some model wings with unbent fuselages  
 used in wall reflection plane technique.



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7'x10' BUMP MODELS

Figure 5.- Drawing of the wing-fuselage configurations used on the transonic bump.

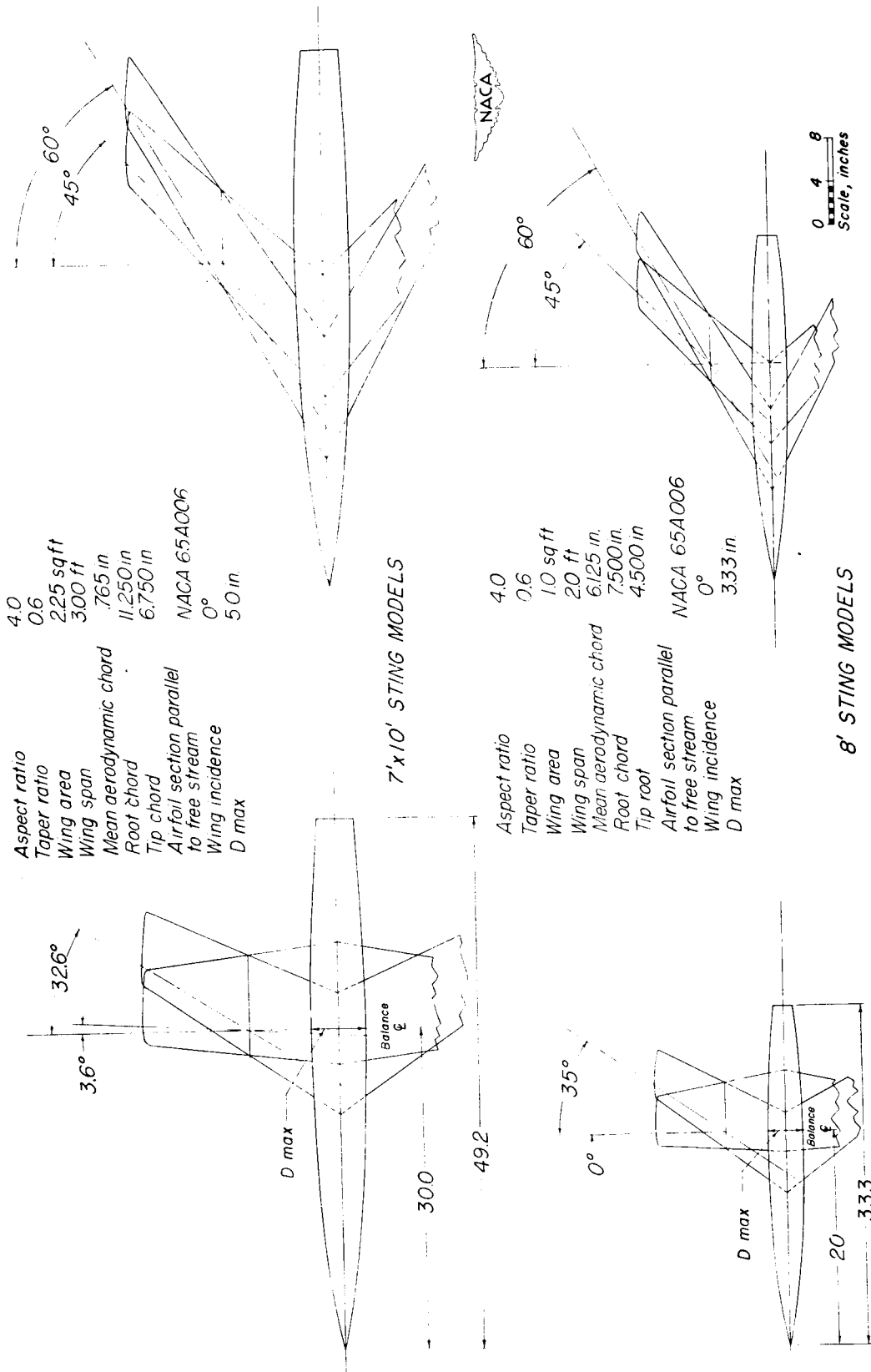


Figure 6.- Drawing of sting models used in two different test facilities.

*7' x 10' Sting Wing*  
 *$\Lambda = 60^\circ$  (75ST Aluminum)*

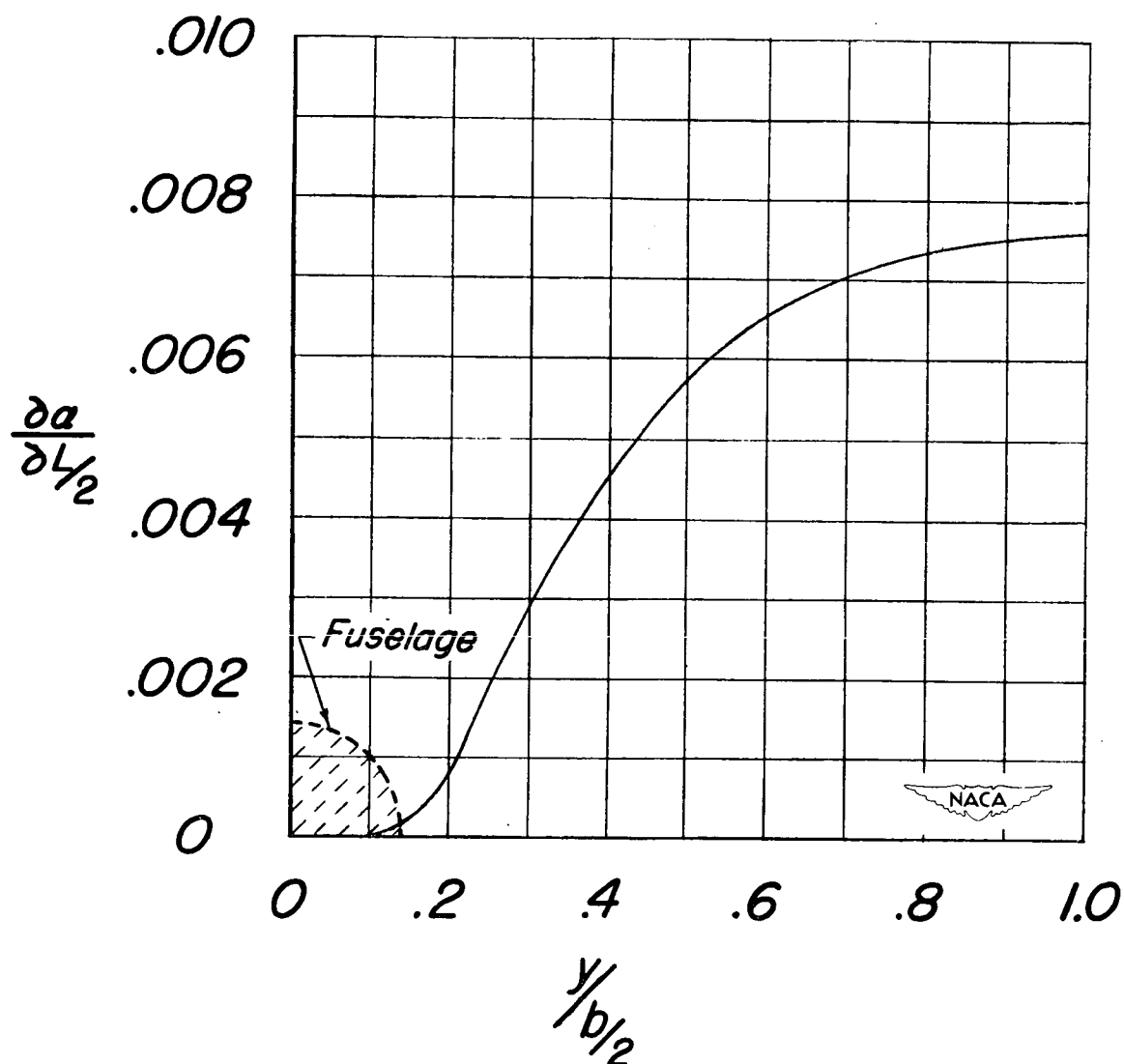


Figure 7.- Measured twist per unit of panel lift, degrees per pound, along the span of a  $60^\circ$  sweptback wing model.



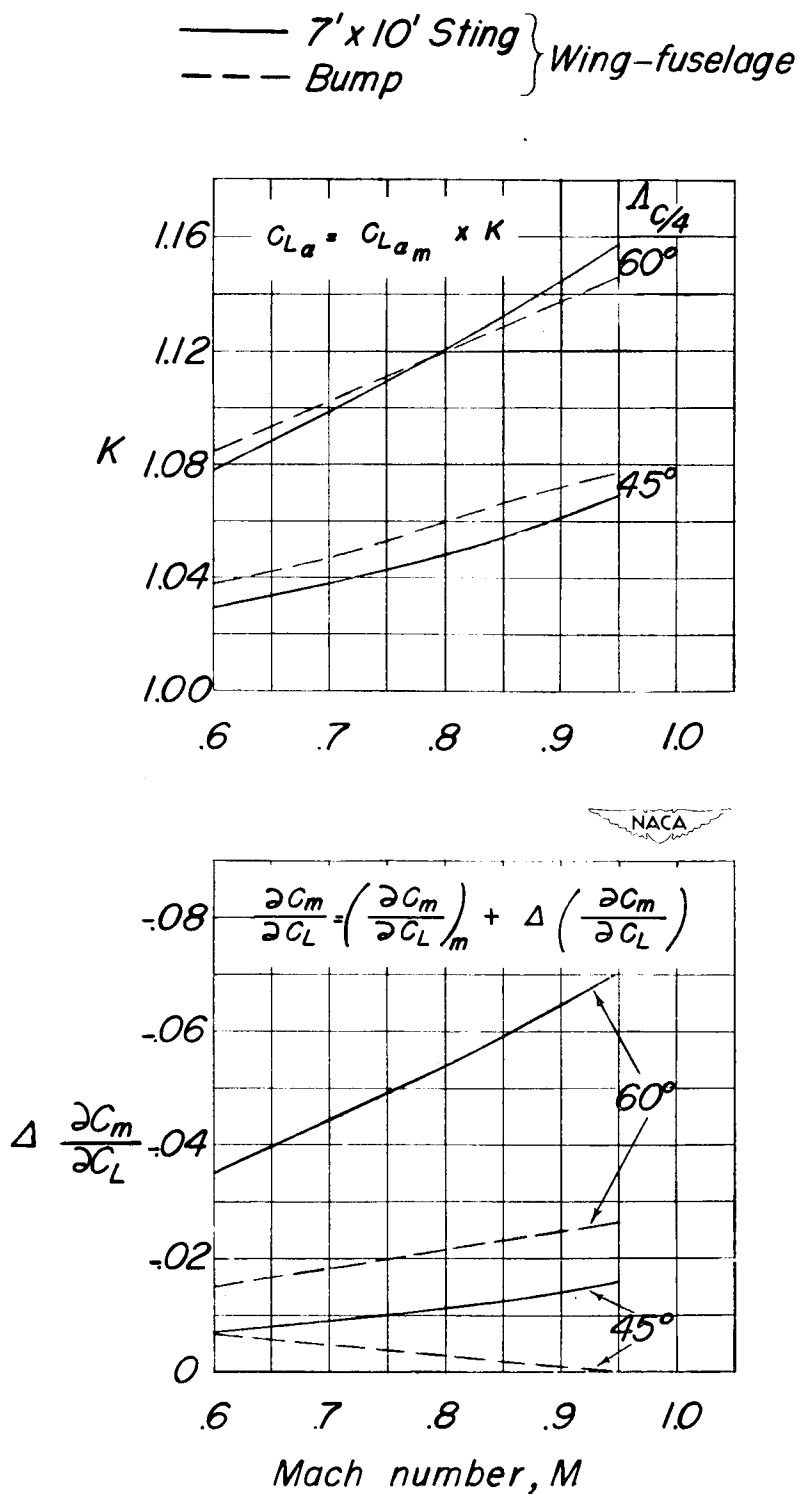


Figure 8.- Correction factors used to determine aerodynamic effects as a result of twist for two different models in two test facilities.

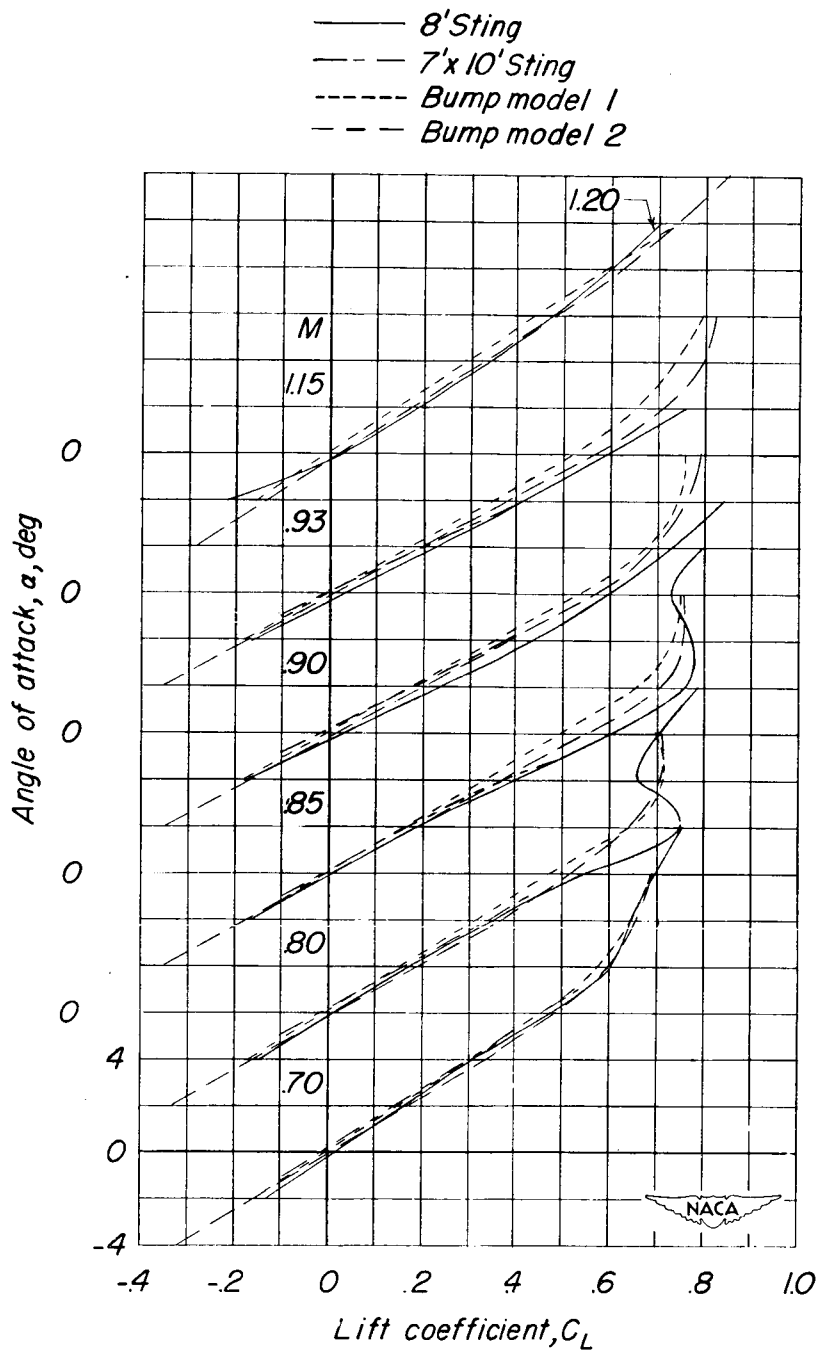
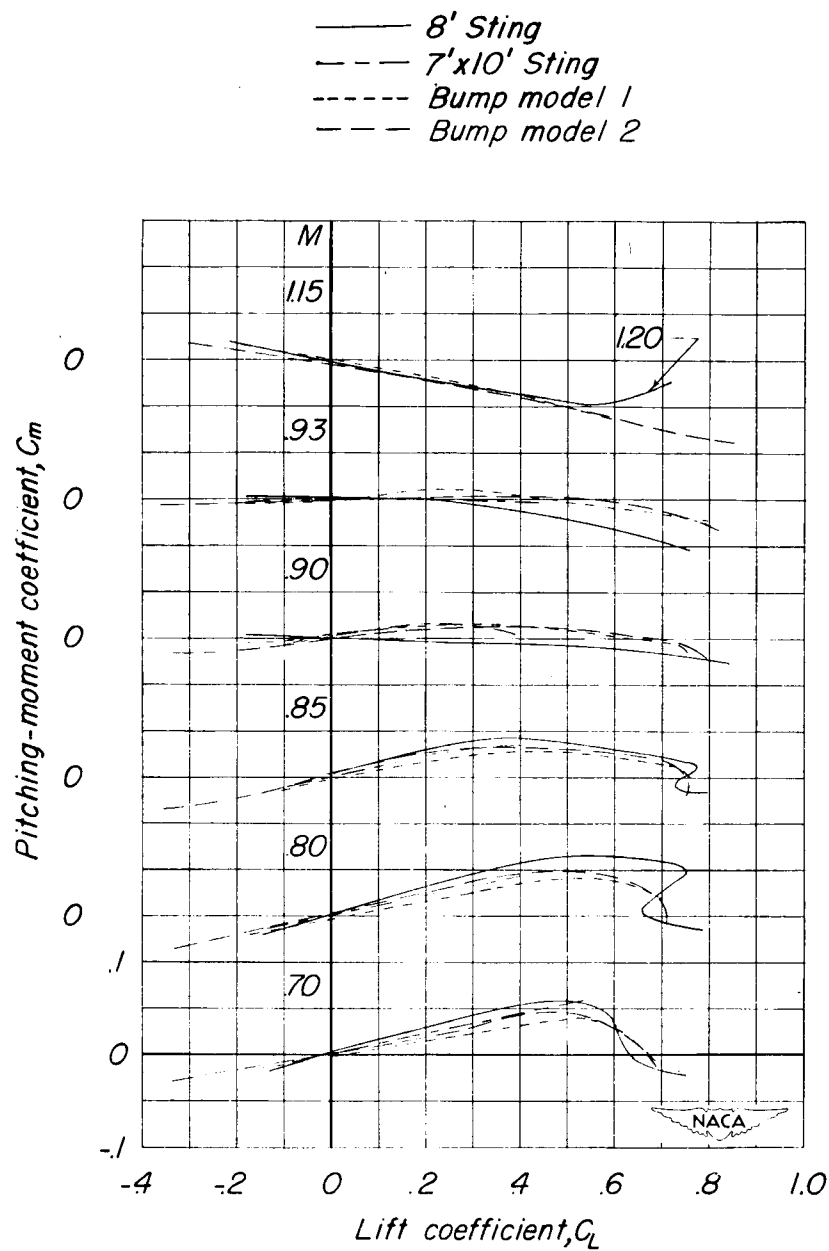
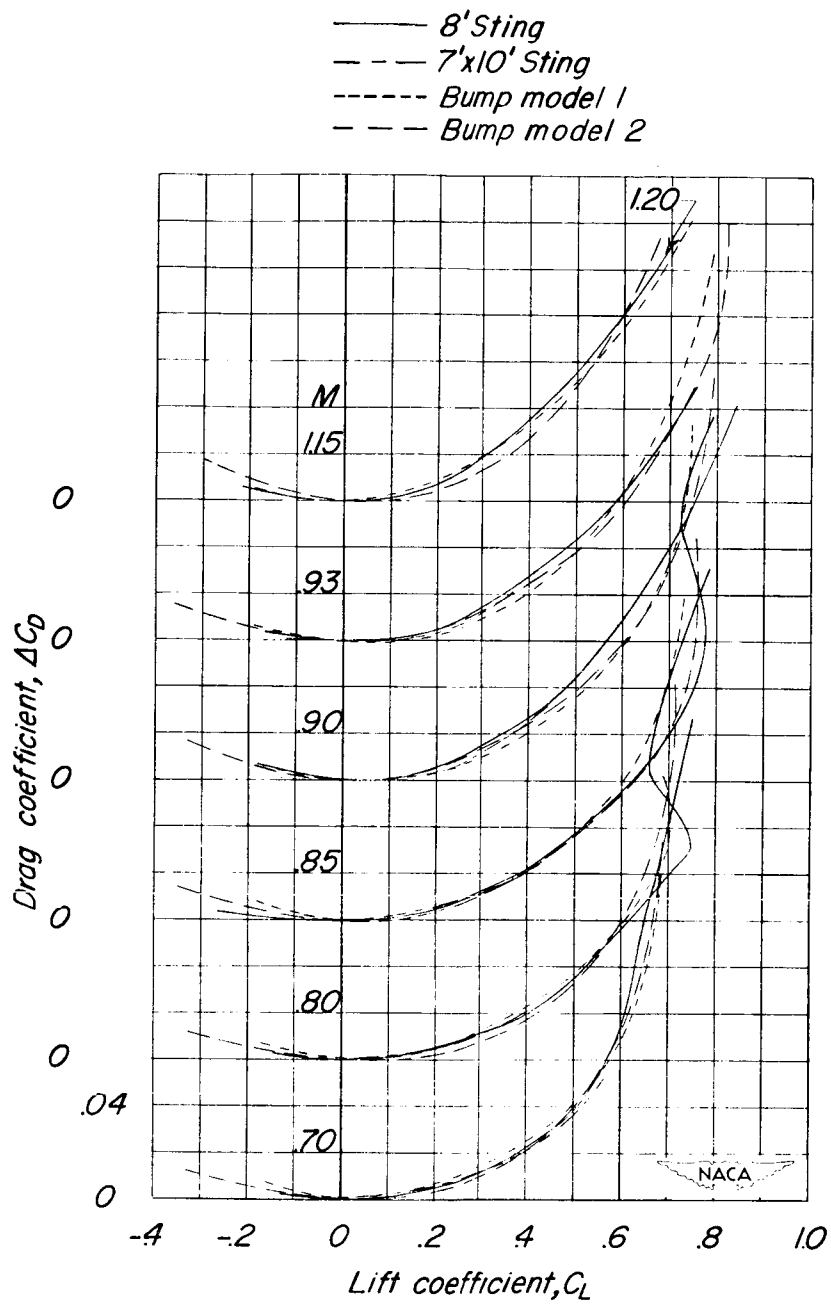
(a)  $\alpha$  against  $C_L$ .

Figure 9.- Wing-fuselage aerodynamic characteristics as determined from different test facilities for a model with unswept wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section parallel to free stream.



(b)  $C_m$  against  $C_L$ .

Figure 9.- Continued.



(c)  $\Delta C_D$  against  $C_L$ .

Figure 9.- Concluded.

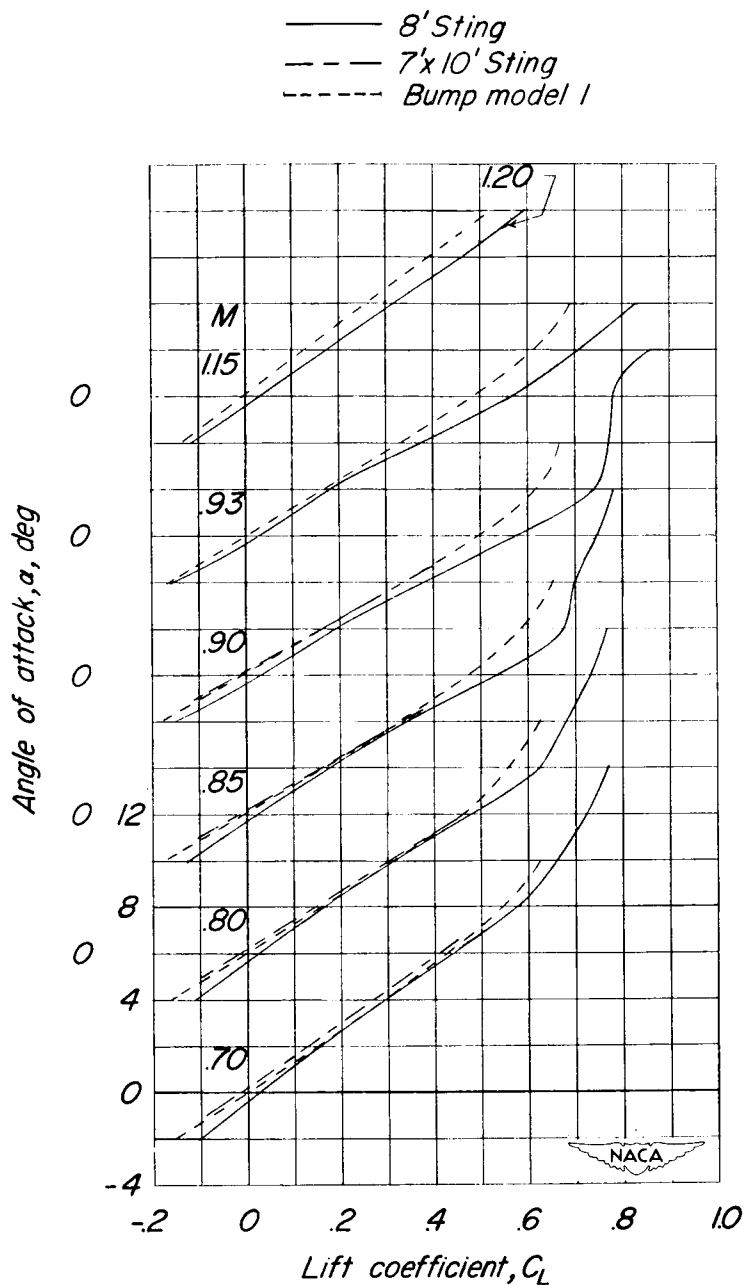


Figure 10.- Wing-fuselage aerodynamic characteristics as determined from different test facilities for a model with  $35^\circ$  sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section parallel to free stream.

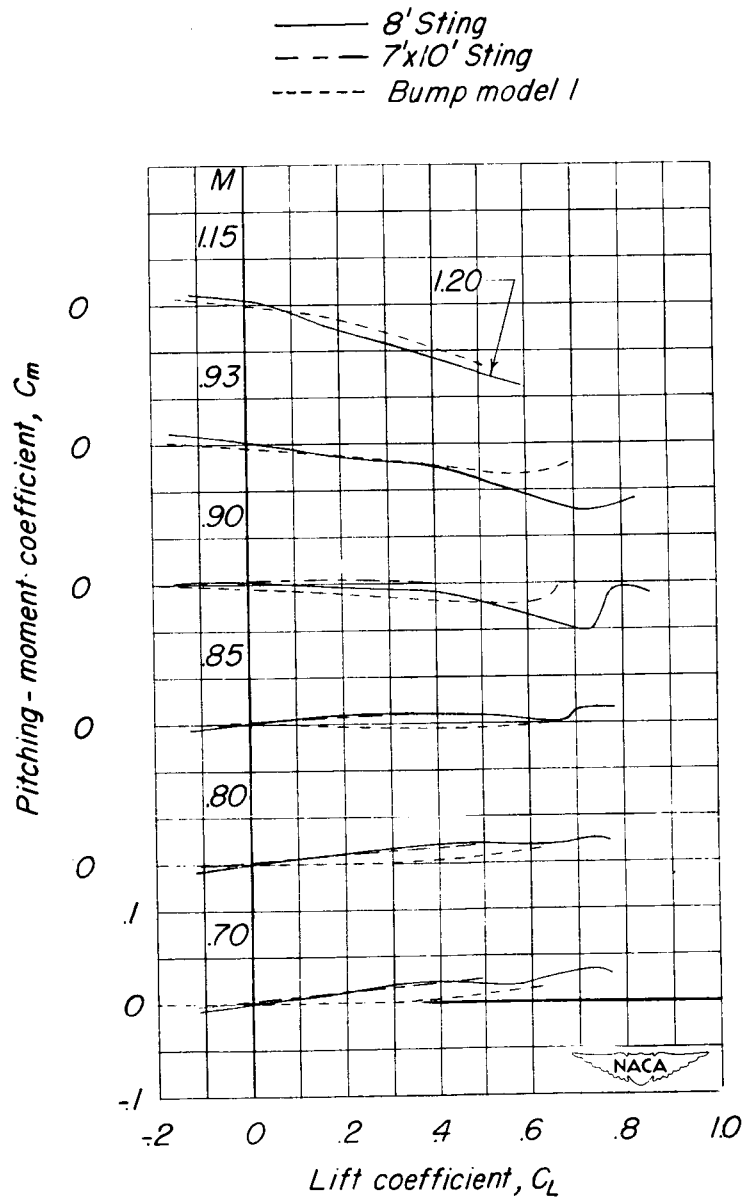
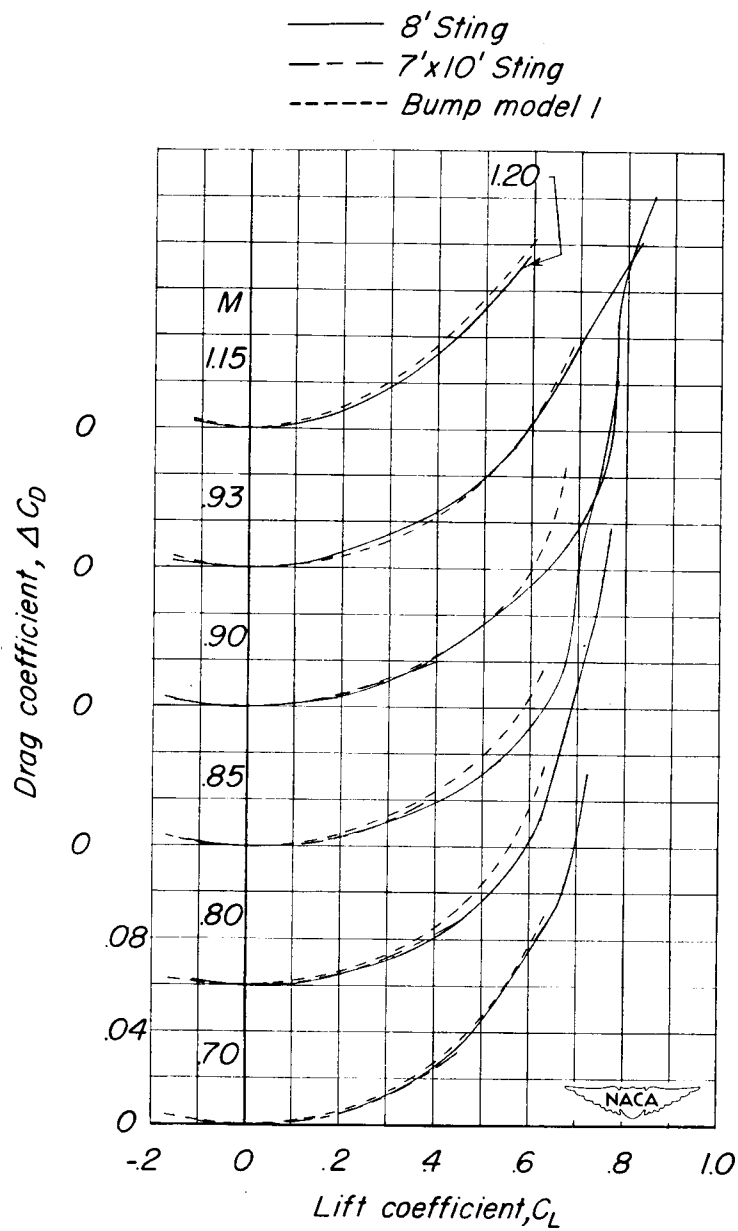
(b)  $C_m$  against  $C_L$ .

Figure 10.- Continued.



(c)  $\Delta C_D$  against  $C_L$ .

Figure 10.- Concluded.

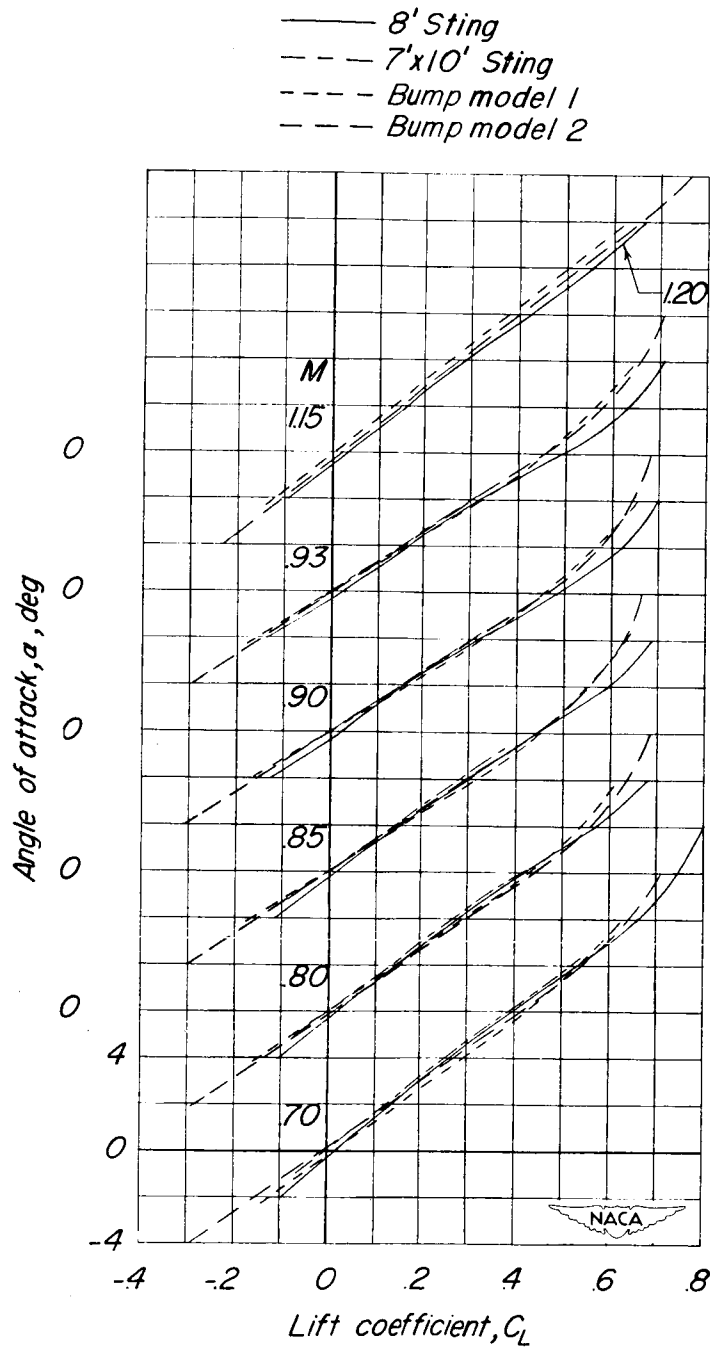
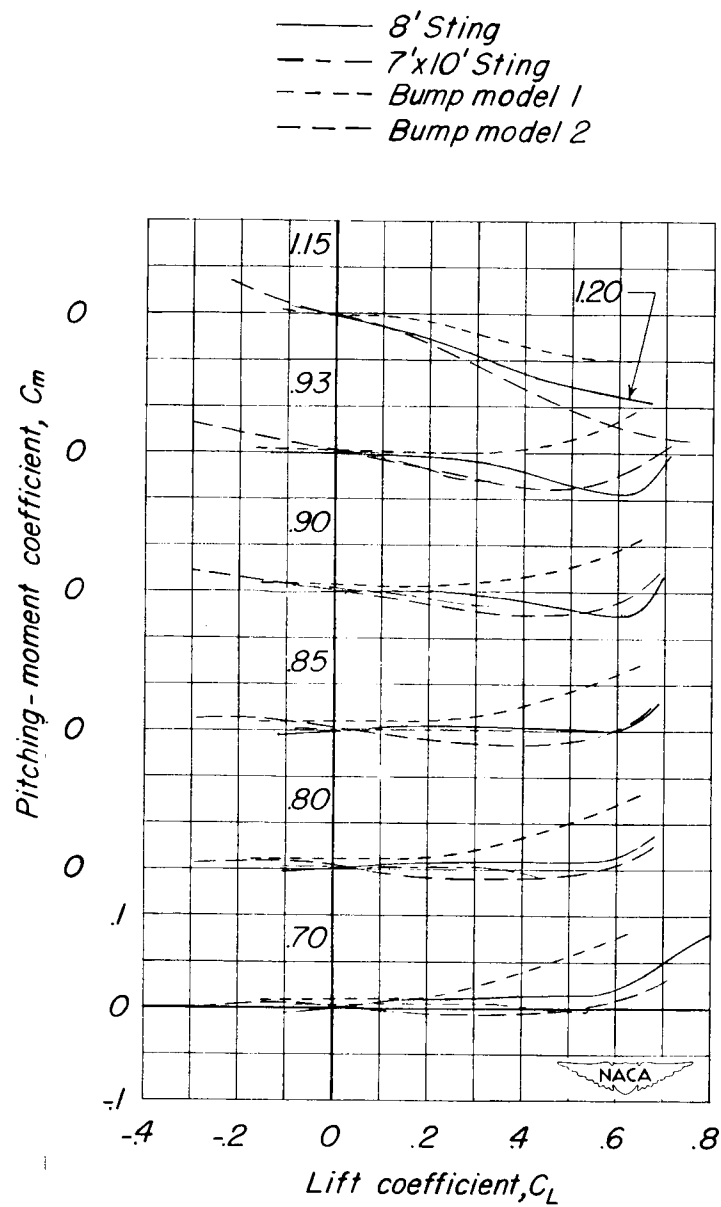
(a)  $\alpha$  against  $C_L$ .

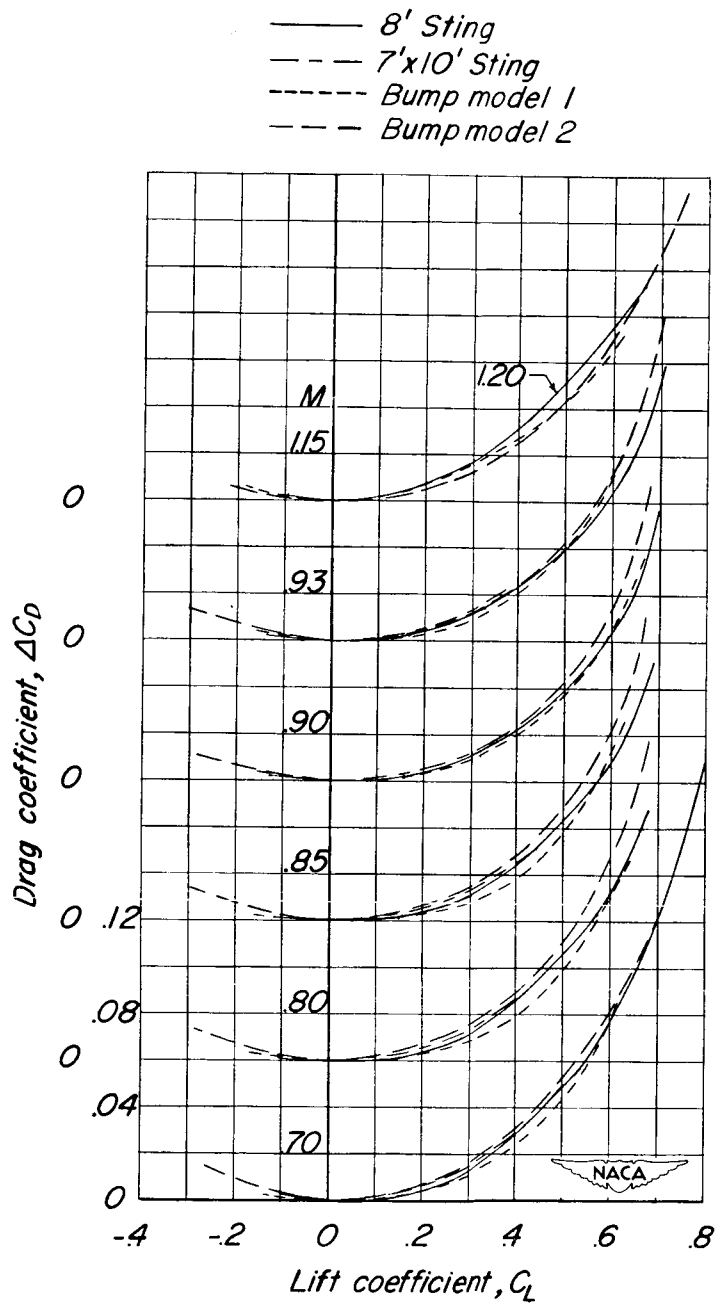
Figure 11.- Wing-fuselage aerodynamic characteristics as determined from different test facilities for a model with  $45^\circ$  sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section parallel to free stream.





(b)  $C_m$  against  $C_L$ .

Figure 11.- Continued.



(c)  $\Delta C_D$  against  $C_L$ .

Figure 11.- Concluded.

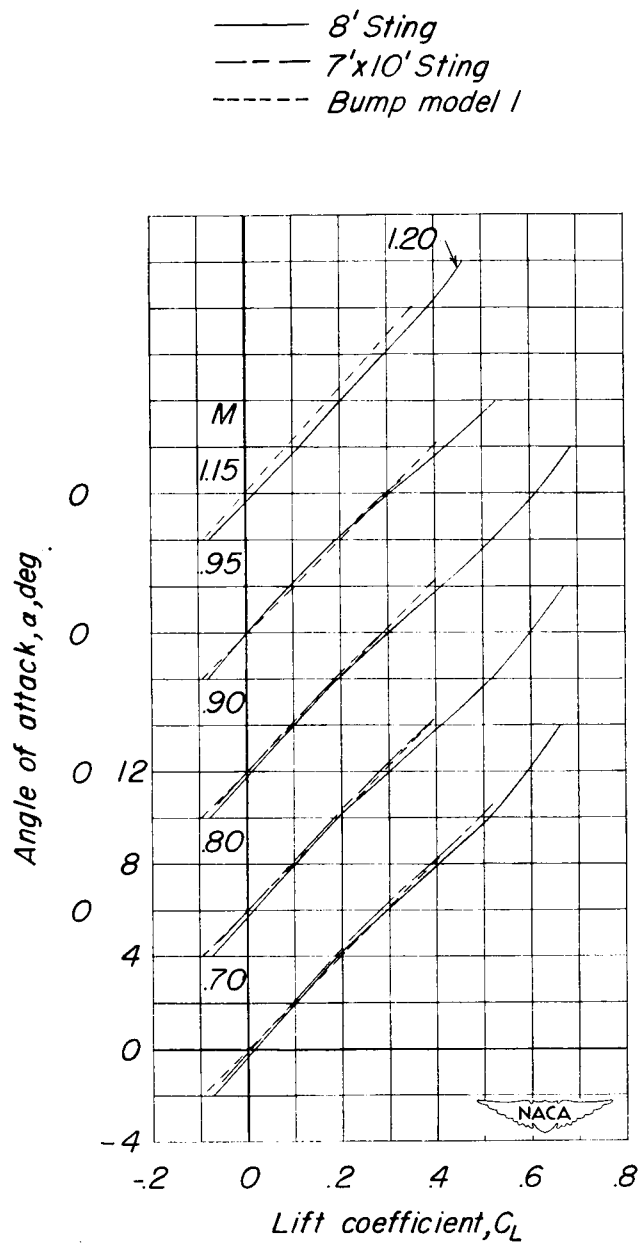
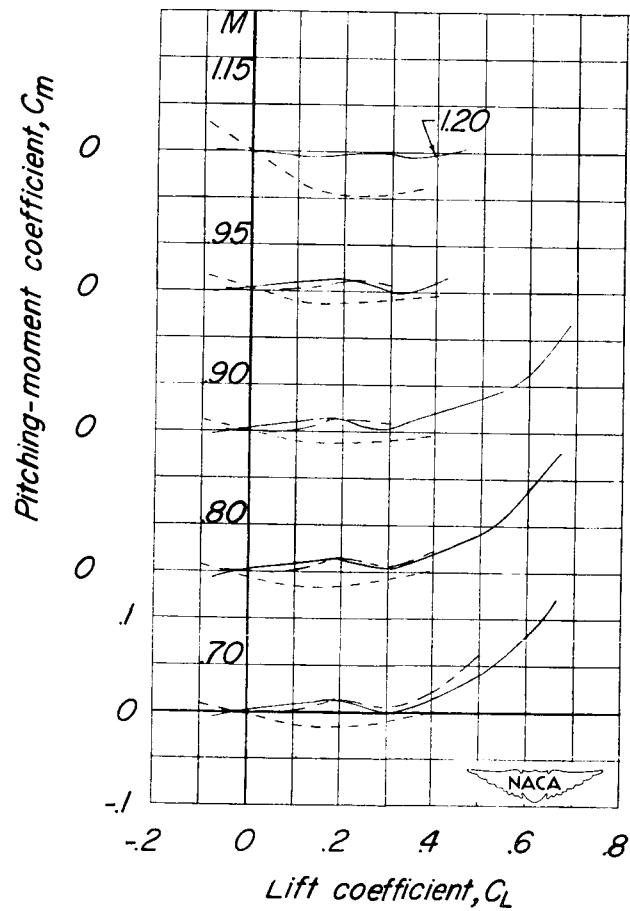


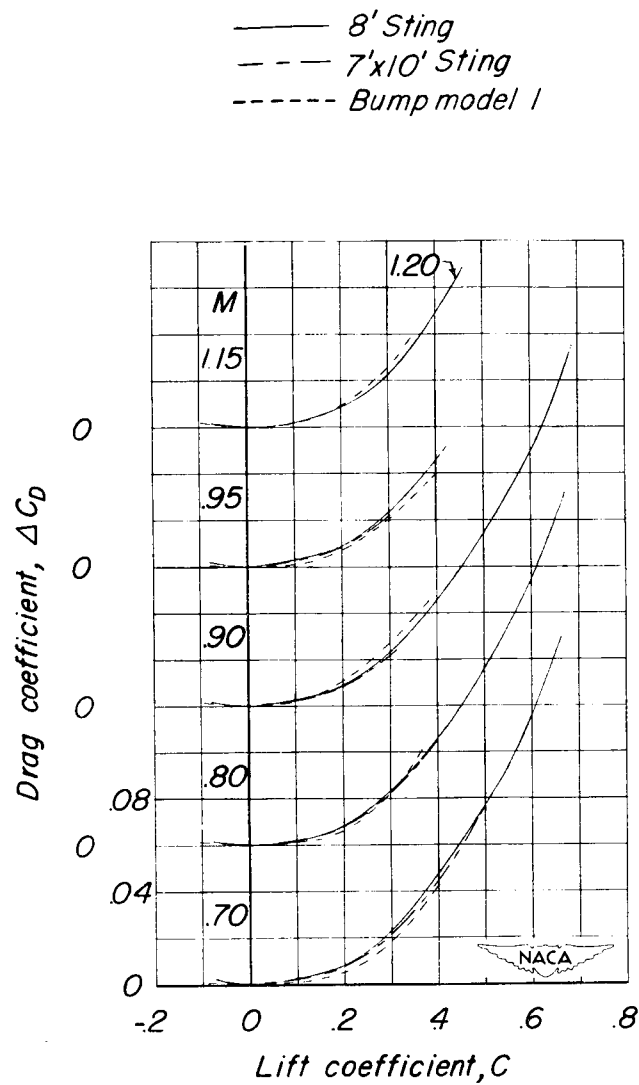
Figure 12.- Wing-fuselage aerodynamic characteristics as determined from different test facilities for a model with  $60^\circ$  sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section parallel to free stream.

— 8' Sting  
 - - - 7'x10' Sting  
 - - - - Bump model 1



(b)  $C_m$  against  $C_L$ .

Figure 12.- Continued.



(c)  $\Delta C_D$  against  $C_L$ .

Figure 12.- Concluded.

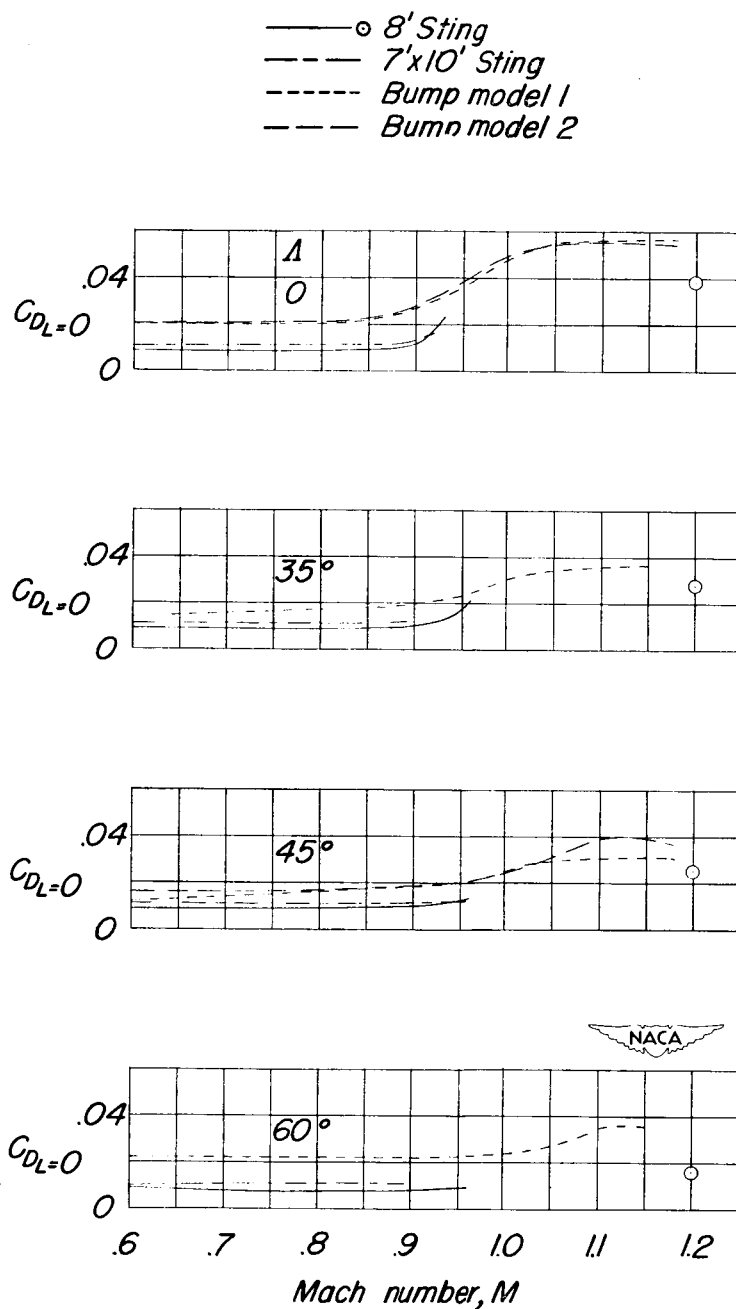


Figure 13.- Effect of sweep and Mach number on drag at zero lift for a wing-fuselage configuration in different test facilities.

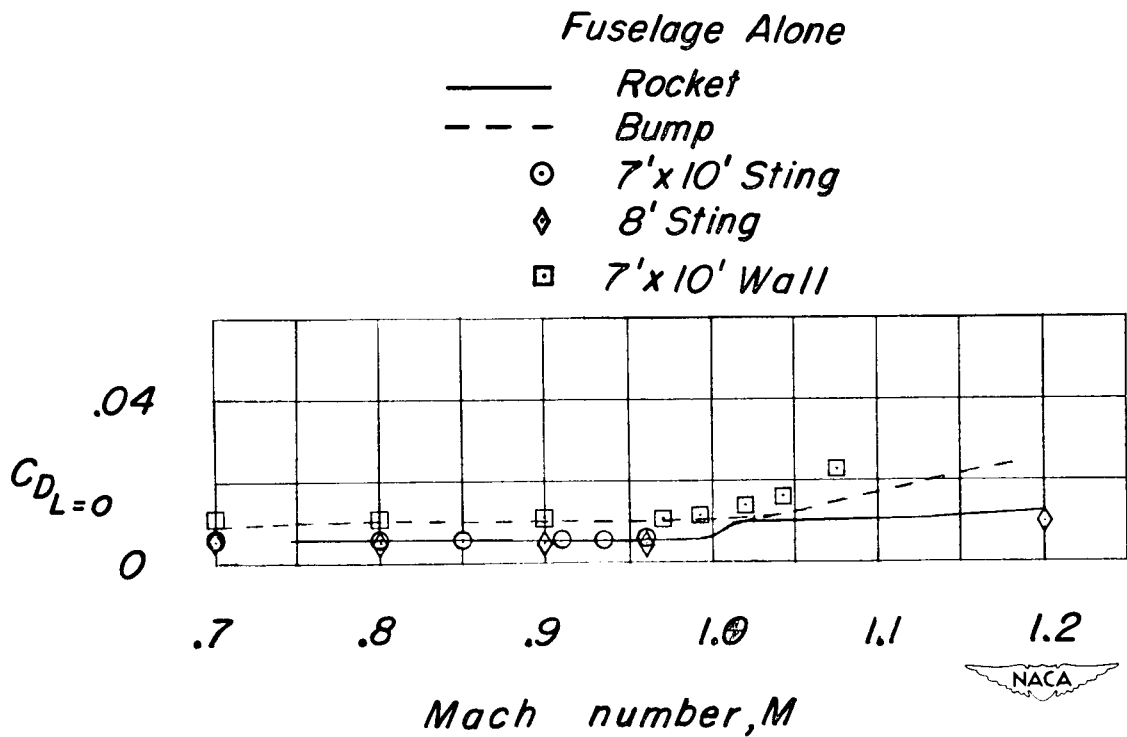
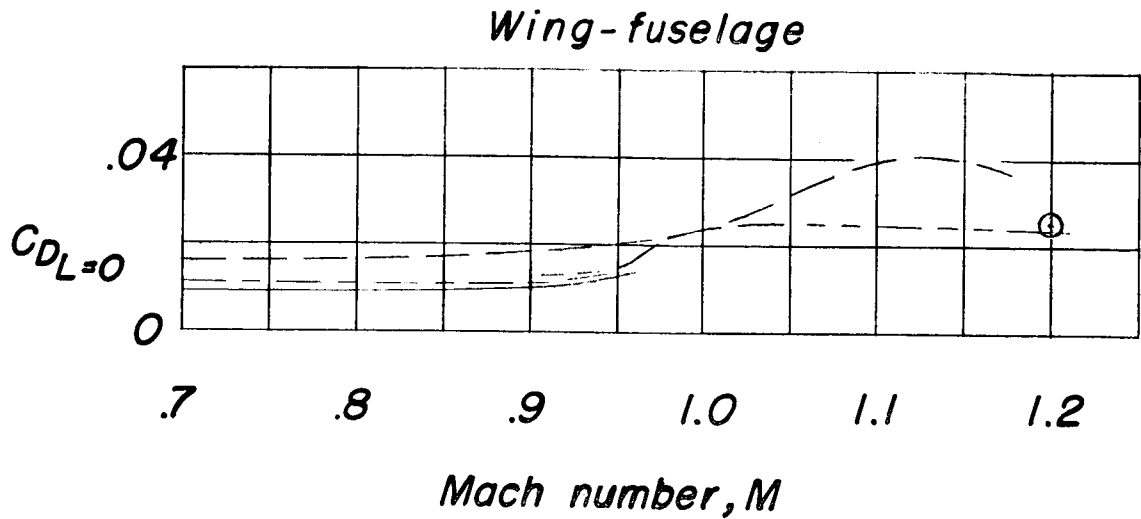


Figure 14.- Variation of drag at zero lift with Mach number for the transonic fuselage as determined from several test facilities.



|         | Source       | $\Delta C/4$ | $A$ | $\lambda$ | $R(M=0.95)$         |
|---------|--------------|--------------|-----|-----------|---------------------|
| — — —   | Bump model 2 | $45^\circ$   | 4   | .6        | $0.80 \times 10^6$  |
| — — —   | 7'x10' Sting | $45^\circ$   | 4   | .6        | $3.55 \times 10^6$  |
| — — — ○ | 8' Sting     | $45^\circ$   | 4   | .6        | $2.02 \times 10^6$  |
| — — —   | Rocket       | $45^\circ$   | 4   | .6        | $10.40 \times 10^6$ |

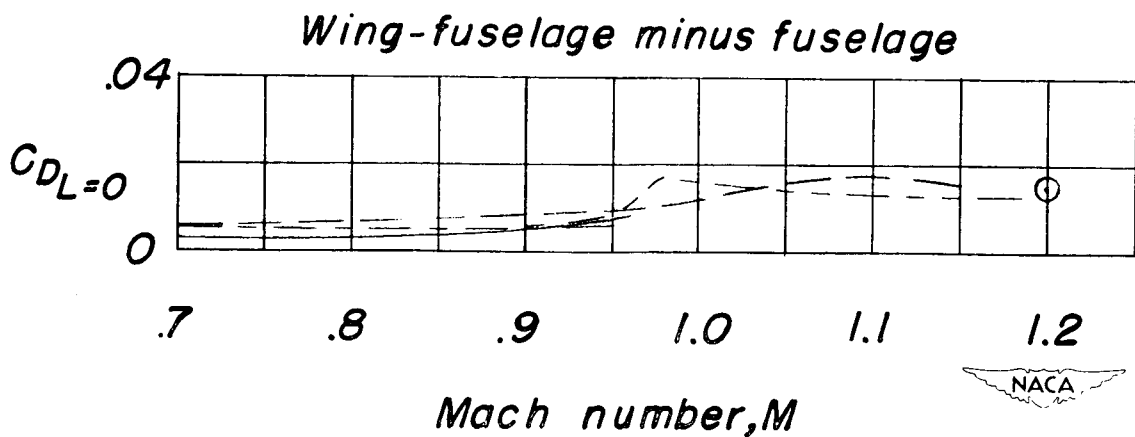
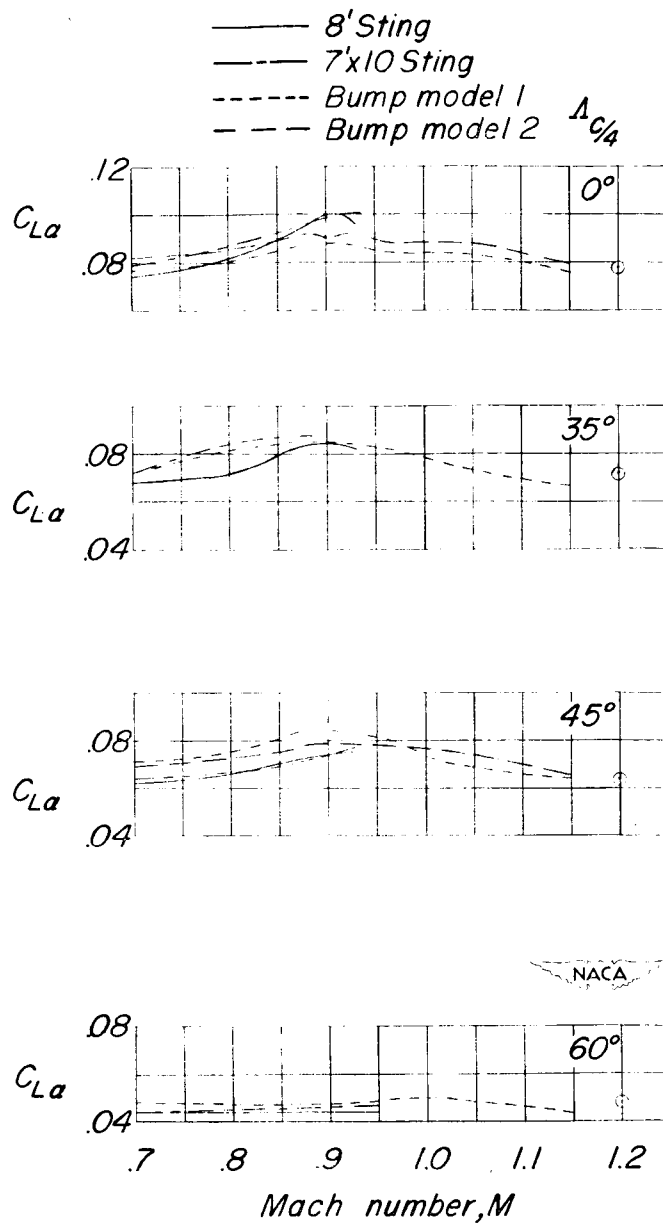


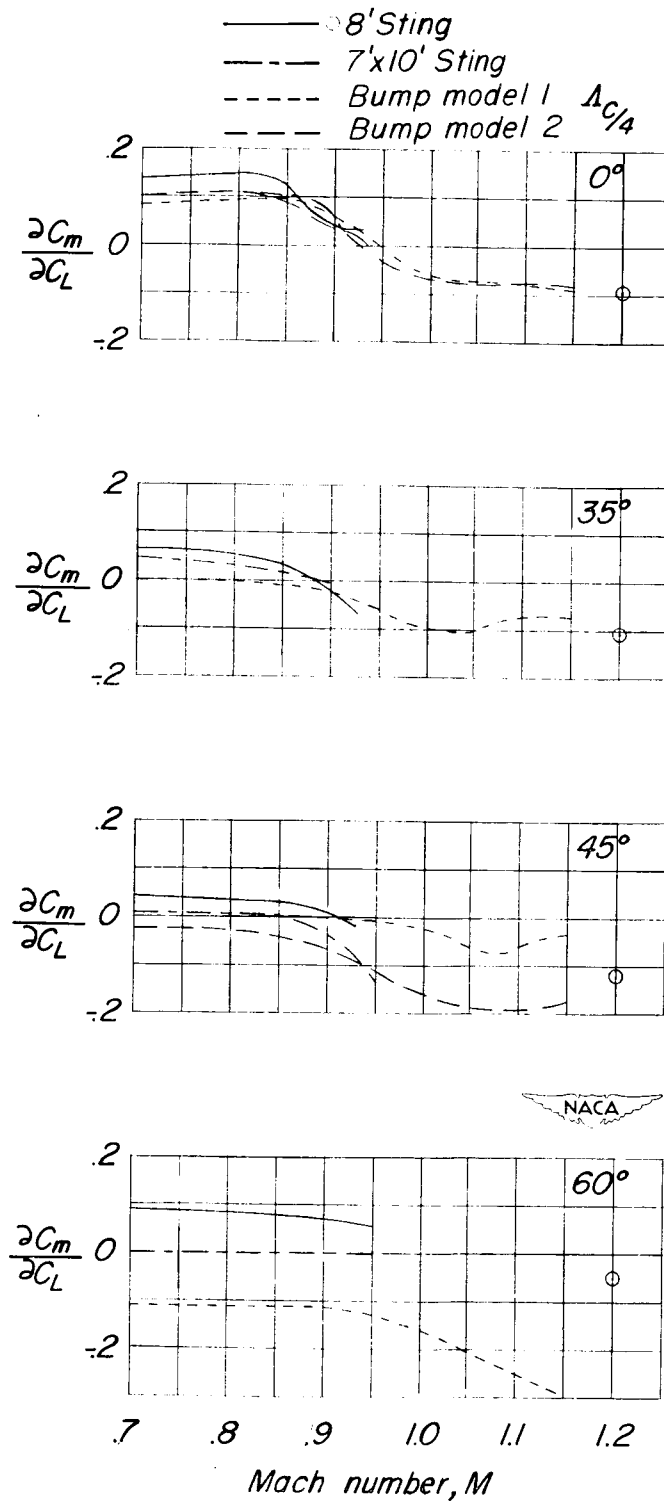
Figure 15.- A comparison of drag at zero lift for two configurations as determined from three different test techniques.





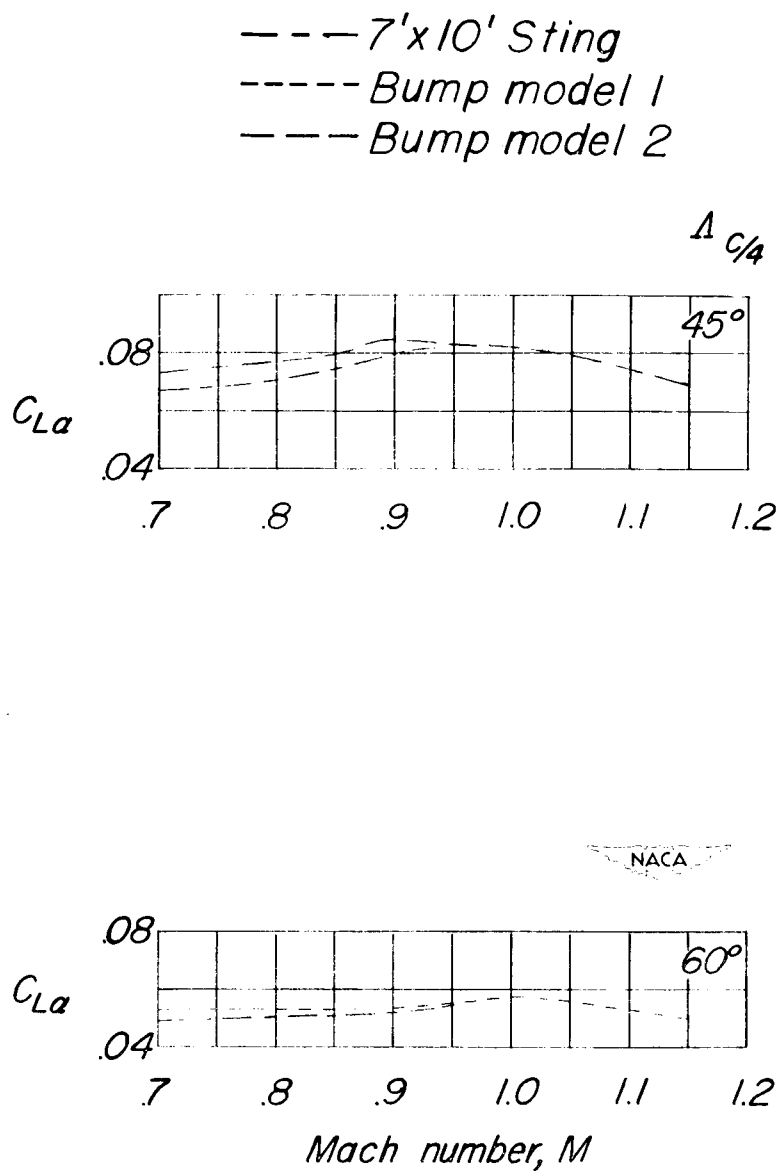
(a)  $C_{L\alpha}$  against  $M$ .

Figure 16.- Summary of aerodynamic characteristics as determined from basic data for the several wing-fuselage configurations in three different test facilities.



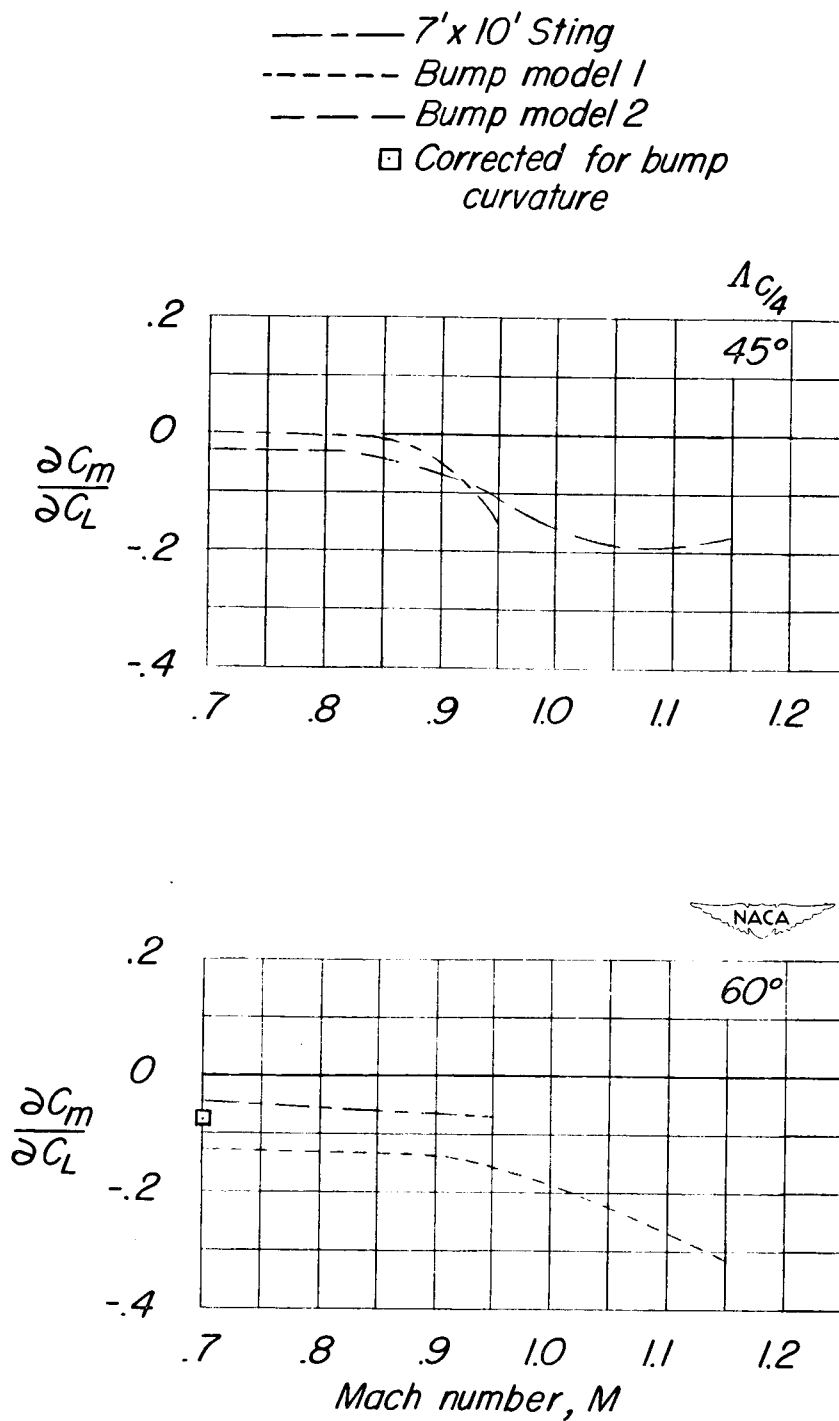
(b)  $C_m/C_L$  against  $M$ .

Figure 16.- Concluded.



(a)  $C_{L\alpha}$  against  $M$ .

Figure 17.- Summary of aerodynamic characteristics corrected for aerodynamic twist for two wing-fuselage configurations in two different test facilities.



(b)  $\partial C_m / C_L$  against  $M$ .

Figure 17.- Concluded.

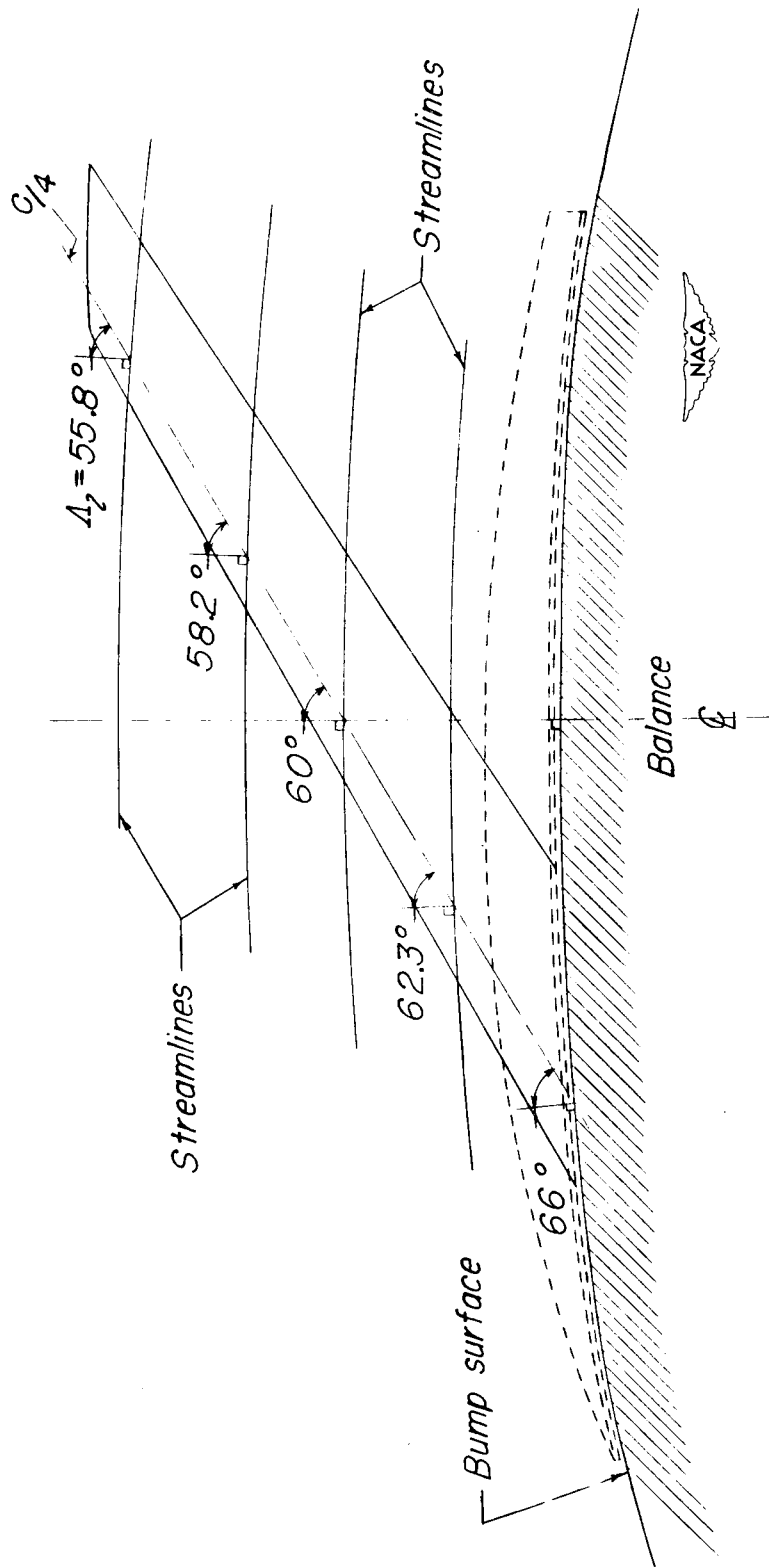


Figure 18.- The effective sweep angle at several spanwise stations on a  $60^\circ$  sweptback wing model when placed in the curved flow field over the bump.

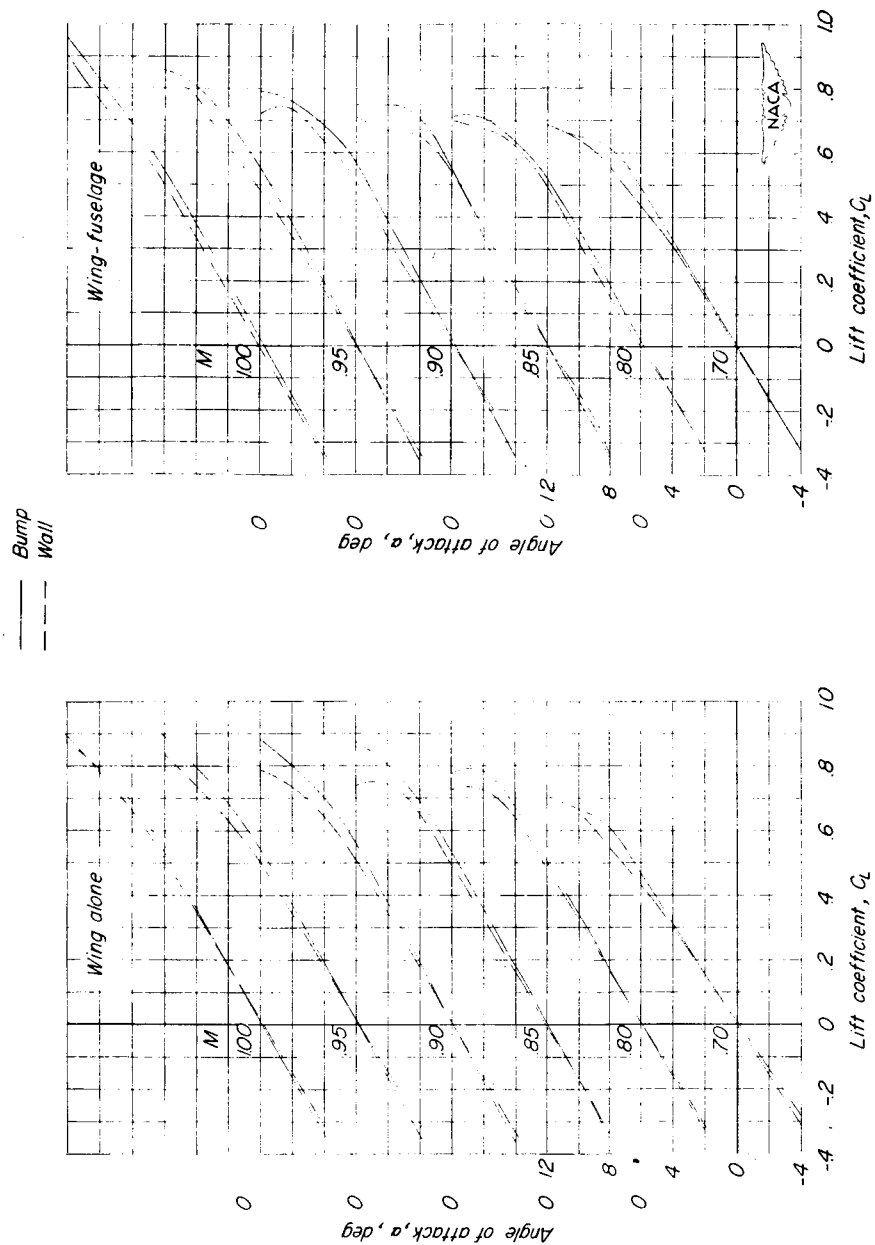
(a)  $\alpha$  against  $C_L$ .

Figure 19.- Aerodynamic characteristics of wing-alone and wing-fuselage configurations as determined from two different test techniques for an unswept wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section.

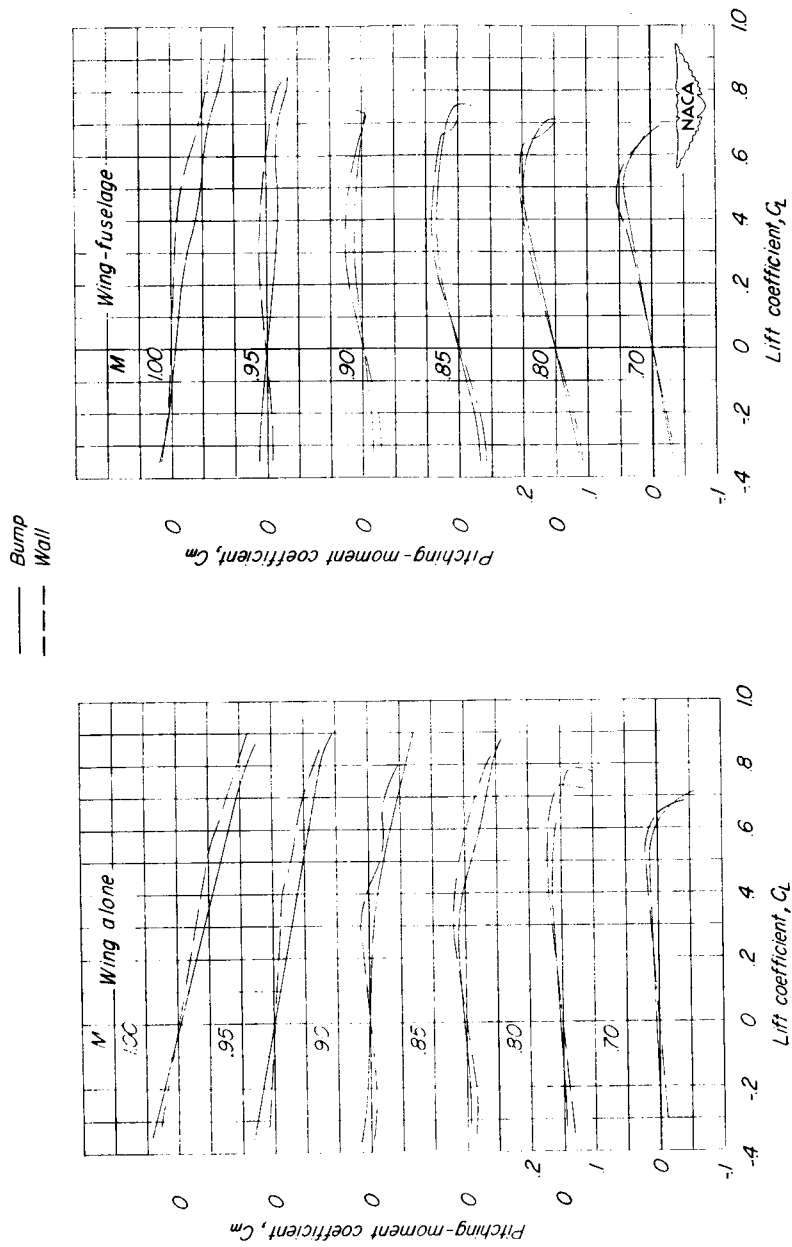
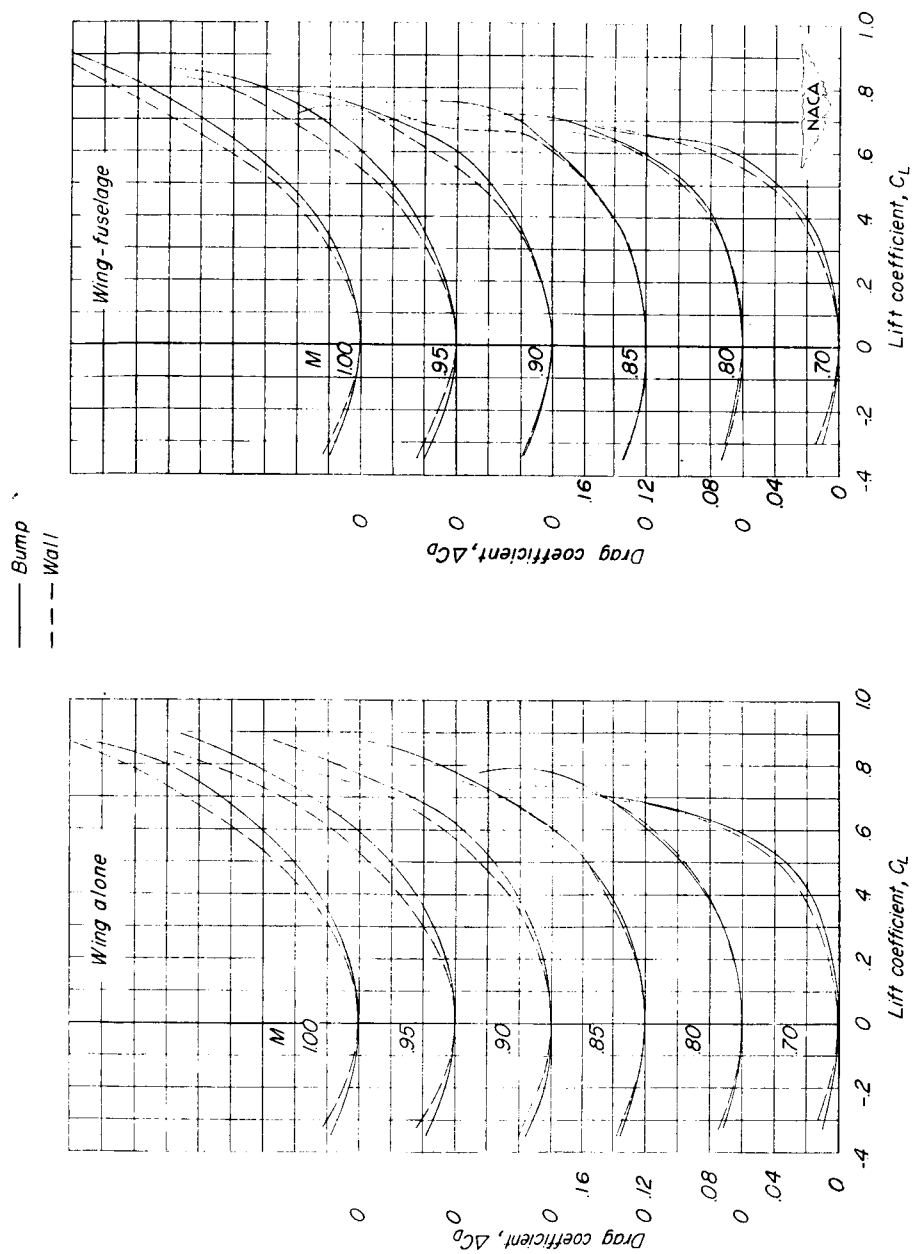
(b)  $C_m$  against  $C_L$ .

Figure 19.- Continued.



(c)  $C_D$  against  $C_L$ .

Figure 19.- Concluded.



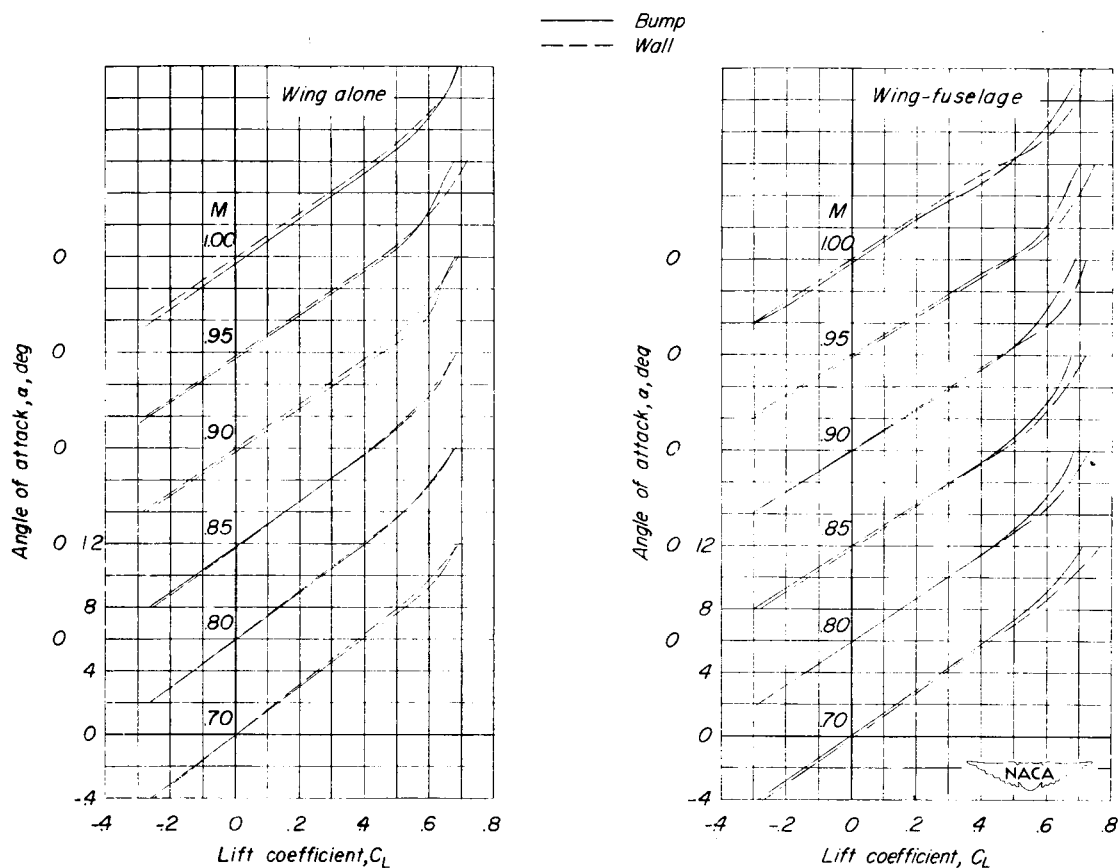
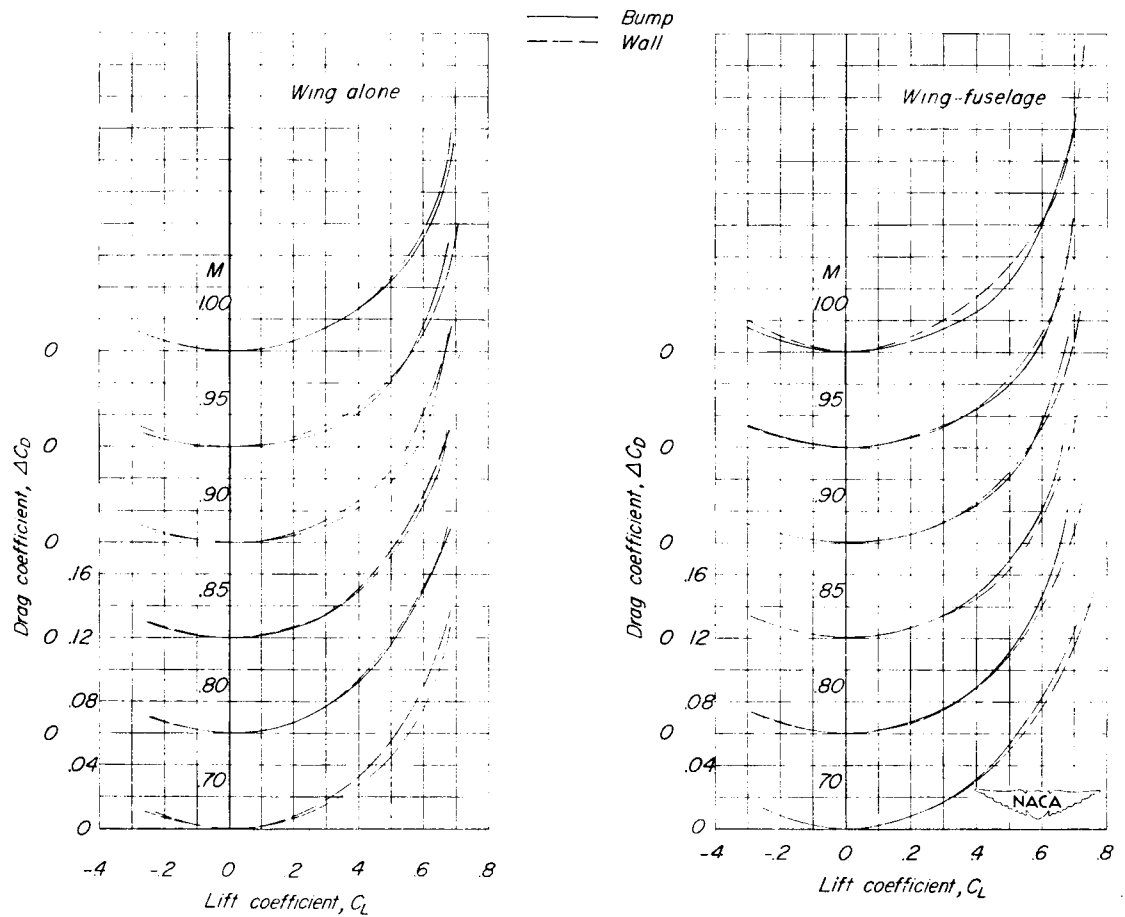
(a)  $\alpha$  against  $C_L$ .

Figure 20.- Aerodynamic characteristics of wing-alone and wing-fuselage configurations as determined from two different test techniques for a 45° sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section.



(c)  $C_D$  against  $C_L$ .

Figure 20.- Concluded.

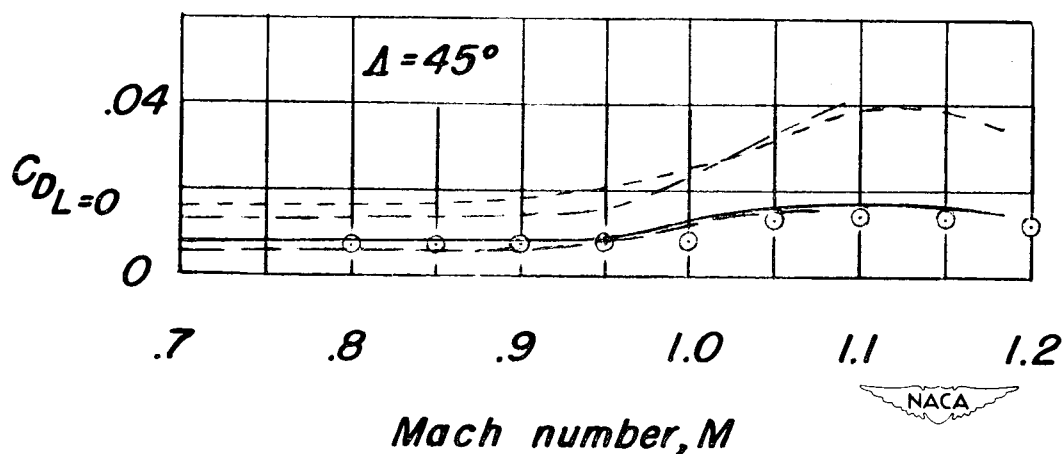
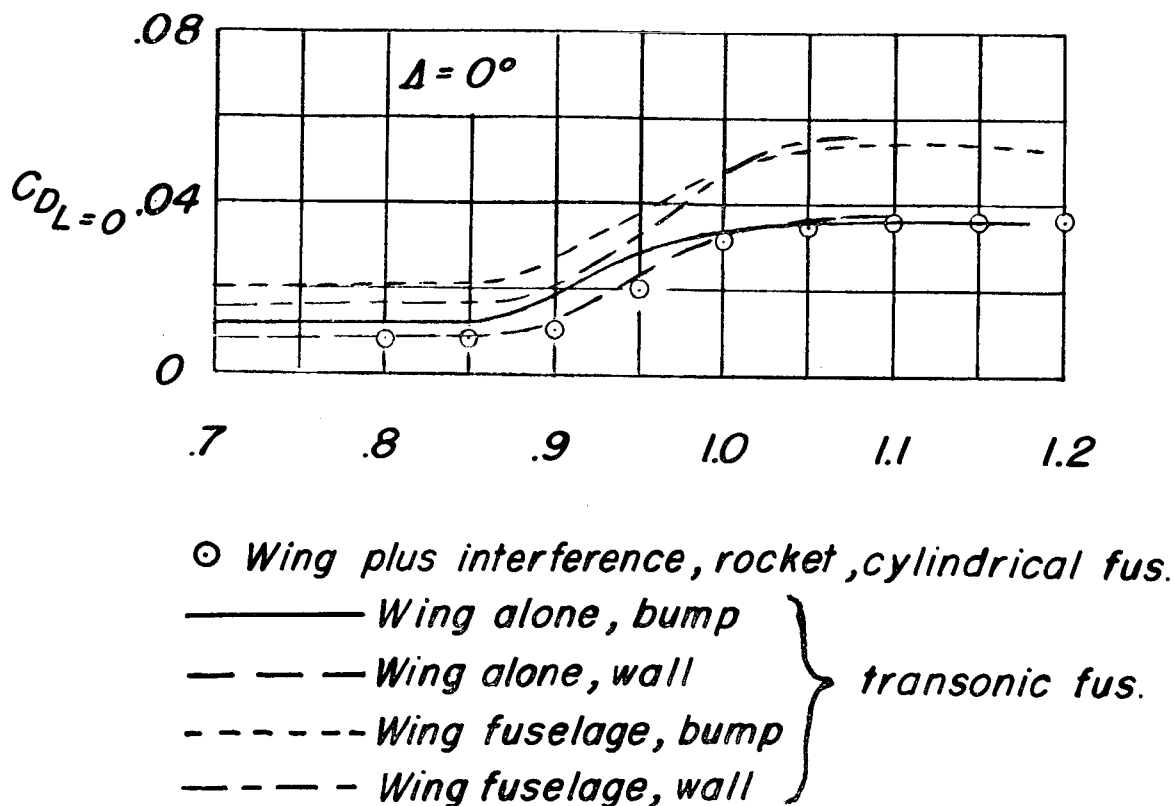


Figure 21.- Effect of sweep on the variation with Mach number of drag at zero lift for two test techniques and for two configurations.

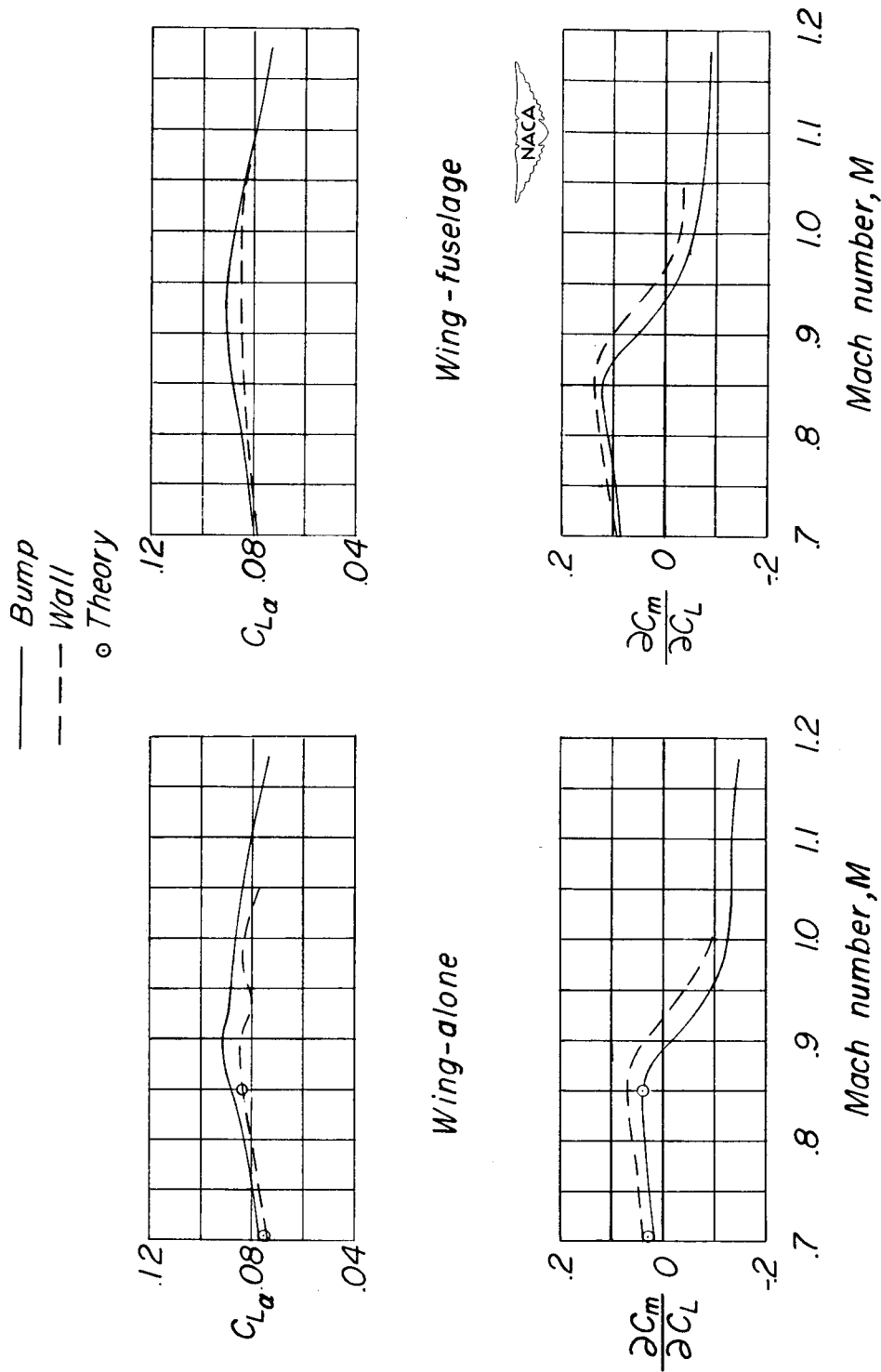
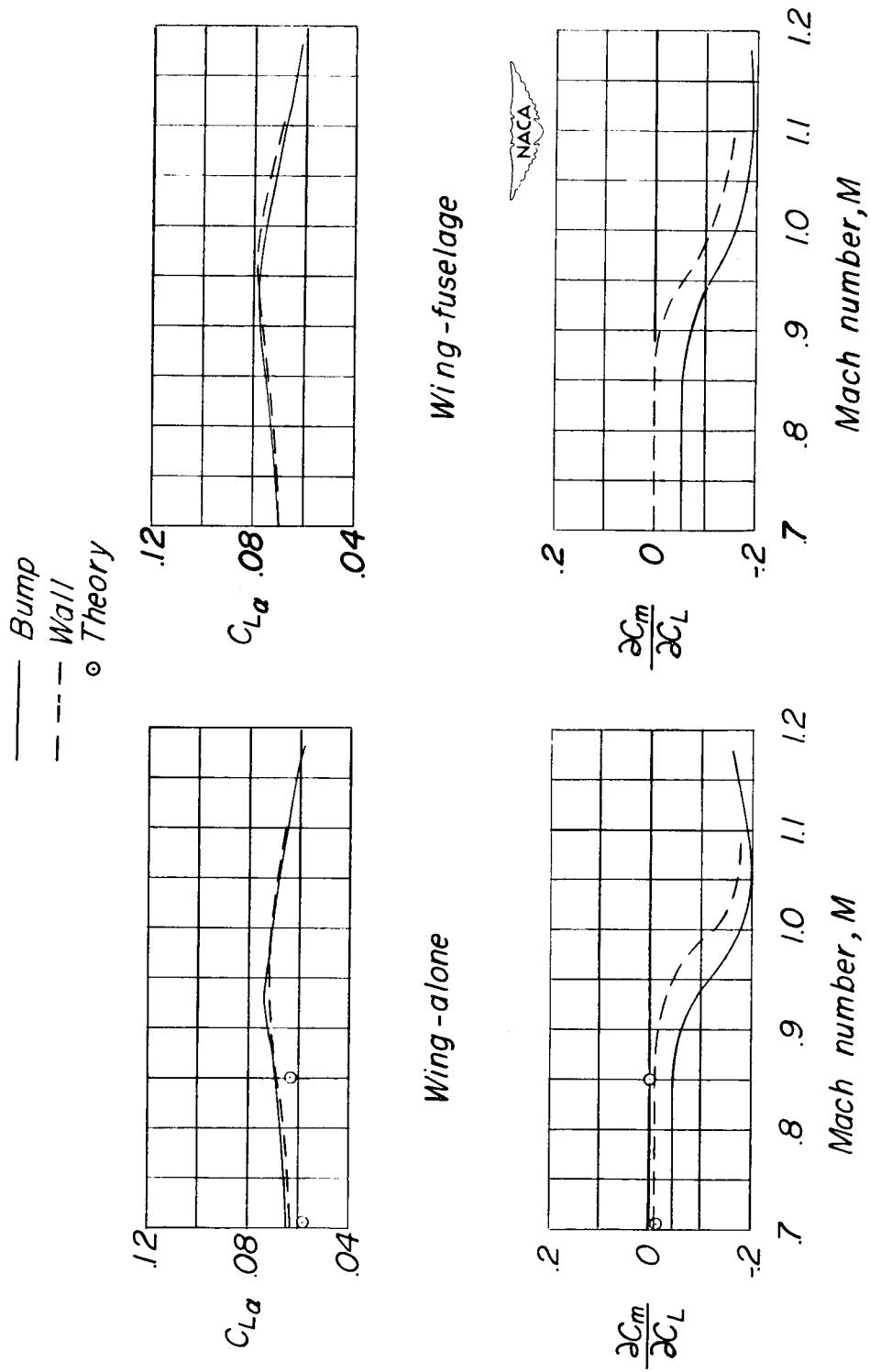
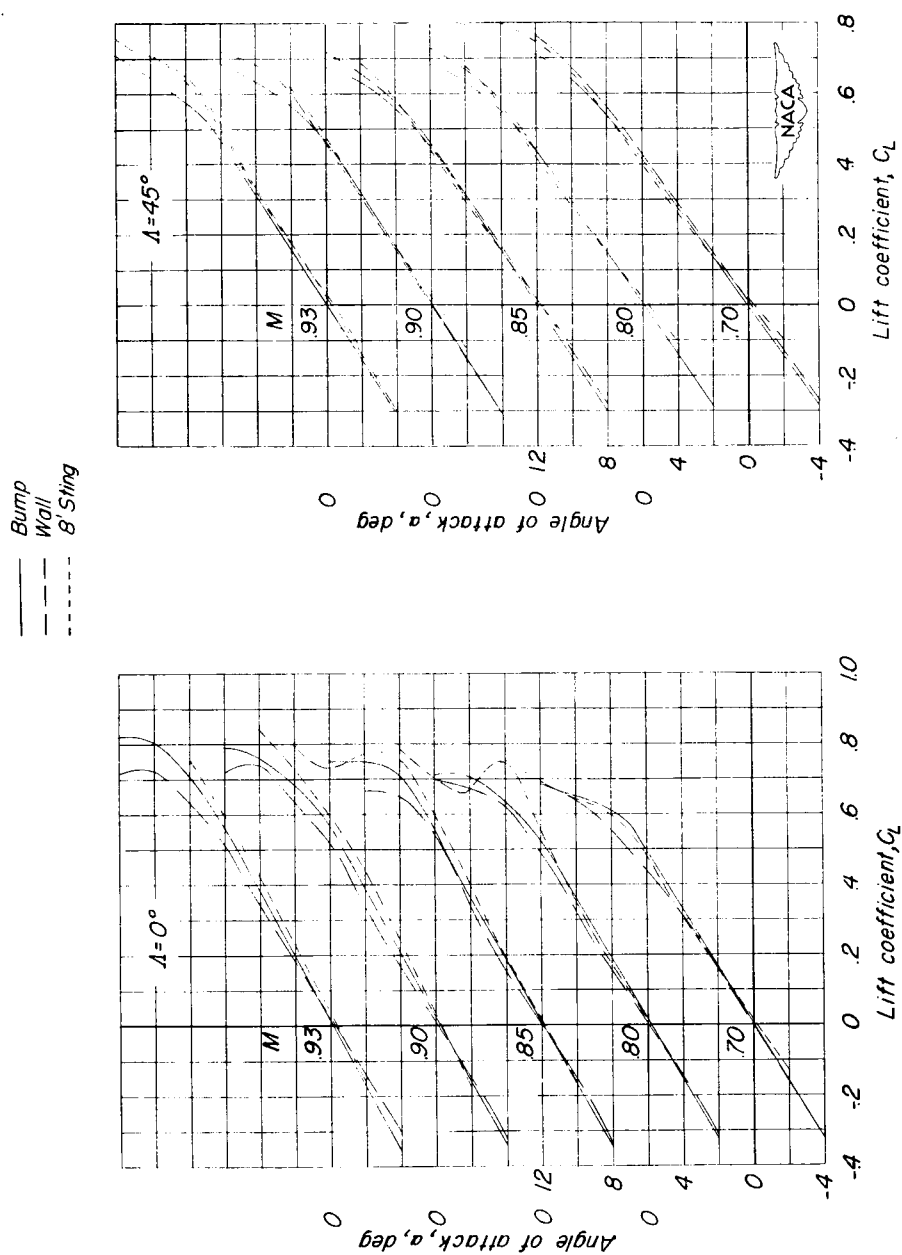
(a)  $\Lambda = 0^\circ$ .

Figure 22.- Variation of lift-curve slope and aerodynamic-center location with Mach number for two configurations at two sweep angles.



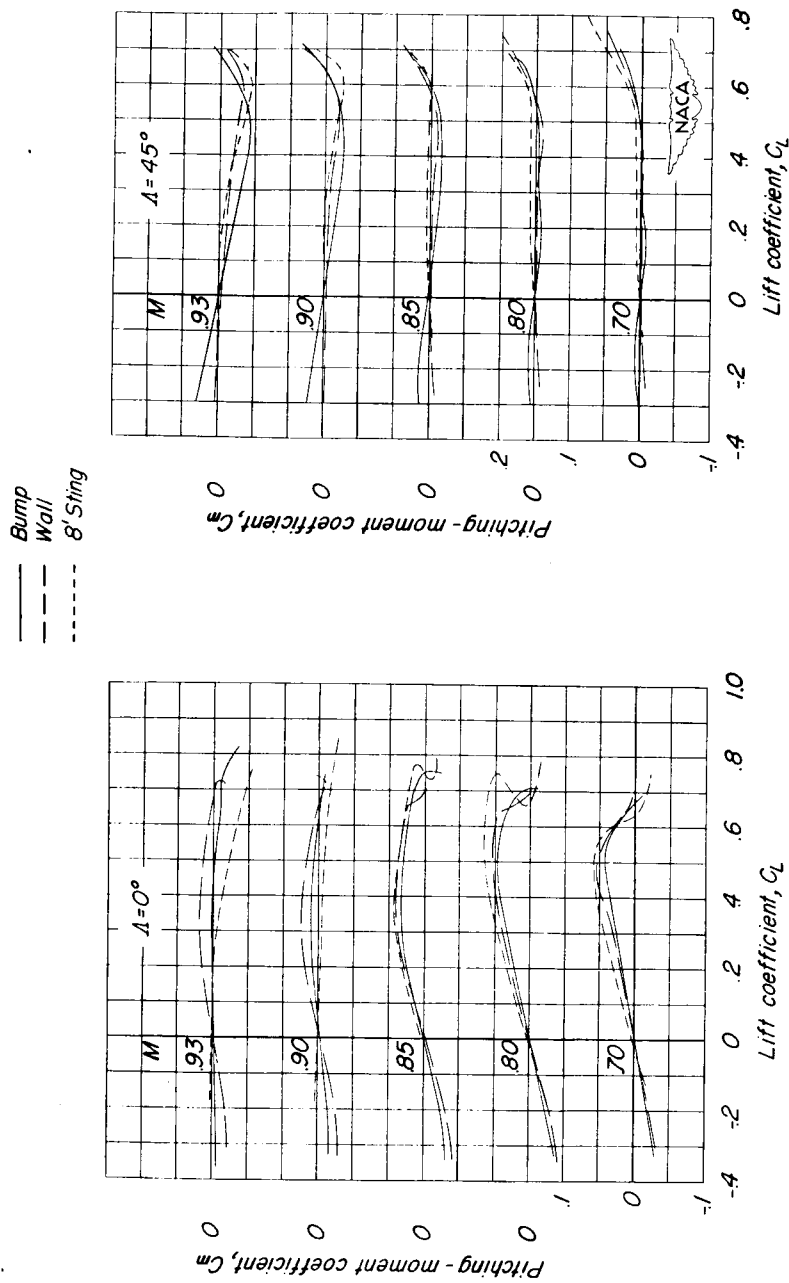
(b)  $\Lambda = 45^\circ$ .

Figure 22.- Concluded.



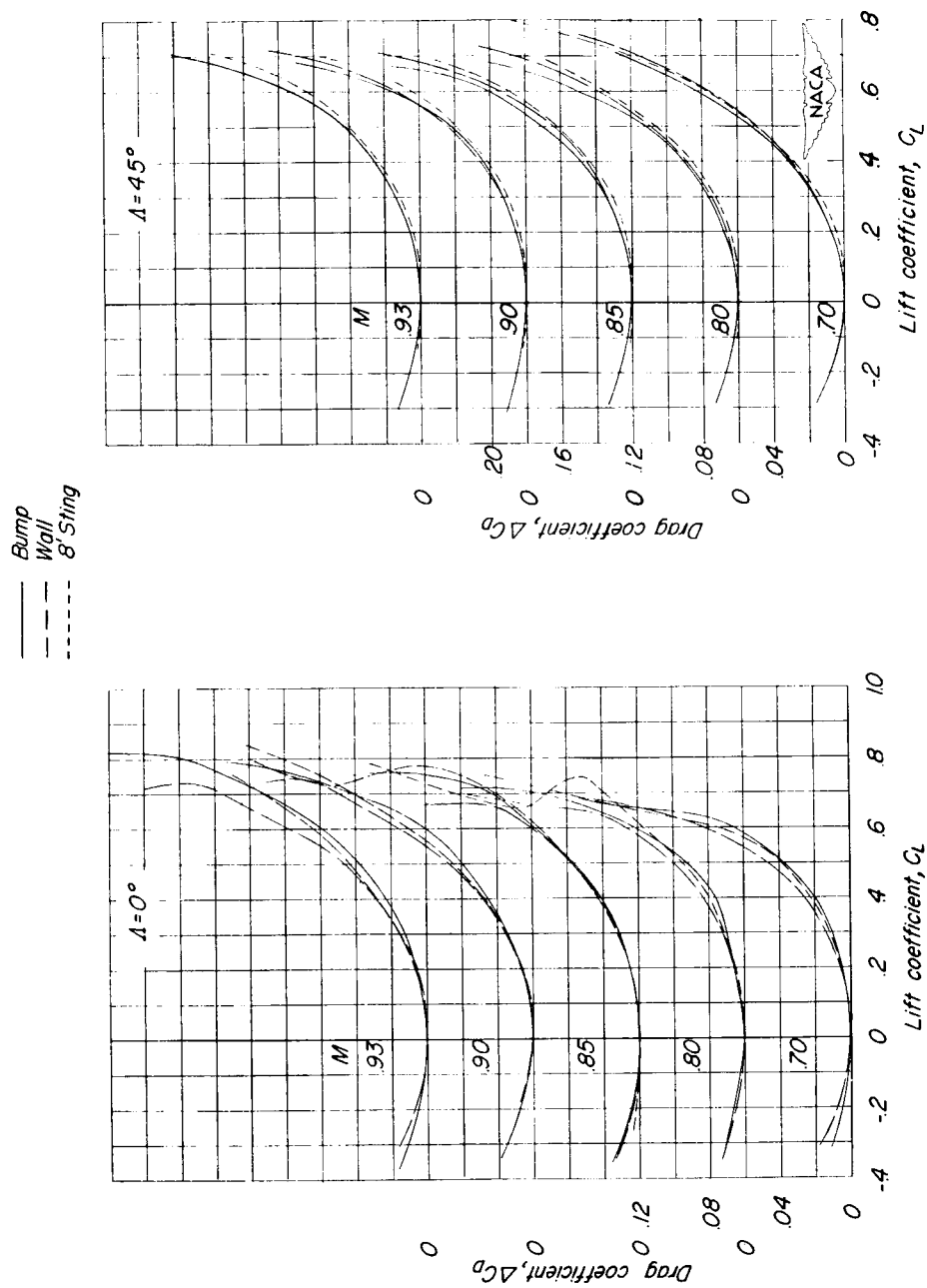
(a)  $\alpha$  against  $C_L$ .

Figure 23.- Aerodynamic characteristics of two wing-fuselage configurations as determined from three different test techniques for an unswept and  $45^\circ$  sweptback wing, aspect ratio 4, taper ratio 0.6, and NACA 65A006 airfoil section.



(b)  $C_m$  against  $C_L$ .

Figure 23.- Continued.



(c)  $\Delta C_D$  against  $C_L$ .

Figure 23.- Concluded.



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# RESEARCH MEMORANDUM

CHARACTERISTICS OF SWEEP WINGS AT HIGH SPEEDS

By Charles J. Donlan and Joseph Weil

Langley Aeronautical Laboratory  
Langley Field, Va.

CLASSIFICATION CHANGED

UNCLASSIFIED

To.....

By authority of *NACA Res. Lab. & Effective*  
*RN-112* Date *2-8-57*

*N12 4-5-57*

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## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON  
January 30, 1952

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NACA RM L52A15

## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

## RESEARCH MEMORANDUM

## CHARACTERISTICS OF SWEPT WINGS AT HIGH SPEEDS

By Charles J. Donlan and Joseph Weil

## INTRODUCTION

Progress is being made in subjecting to systematic study some of the many important factors affecting the behavior of swept wings at high speeds. In this paper are presented some of the results of recent swept-wing investigations that were undertaken to determine the effects of thickness and thickness distribution, camber and twist, nose-flap deflection, and devices or "fixes" for improving the wing pitching-moment characteristics at high lift coefficients.

## SYMBOLS

|               |                                              |
|---------------|----------------------------------------------|
| $C_L$         | lift coefficient                             |
| $C_D$         | drag coefficient                             |
| $C_{D_{min}}$ | minimum drag coefficient                     |
| $C_m$         | pitching-moment coefficient                  |
| $c_{l_i}$     | design section lift coefficient              |
| $L/D$         | lift-drag ratio                              |
| $RN$          | Reynolds number                              |
| $M$           | Mach number                                  |
| $y_{c.p.}$    | lateral center of pressure, percent semispan |
| $b$           | wing span                                    |
| $c$           | wing chord                                   |
| $t$           | maximum wing thickness                       |

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|-----------------|-----------------------------|
| $\alpha$        | angle of attack             |
| $\lambda$       | taper ratio                 |
| $\Lambda$       | sweep angle                 |
| $\Lambda_{c/4}$ | sweep of quarter-chord line |
| $\Lambda_{LE}$  | sweep of leading edge       |
| $\delta_n$      | nose-flap deflection        |

## EFFECT OF THICKNESS AND THICKNESS DISTRIBUTION

### Lift-Curve Slopes

Wings of aspect ratio 3.5.- The lift-curve slope is a useful index for comparing different wings because it gives a direct indication of lift effectiveness and has an important influence on the drag due to lift for thin swept wings having little leading-edge suction. In figure 1 are shown the results of a study of the effects of thickness and thickness distribution on the lift-curve slope (average slope from 0 to  $0.4C_L$ ) for various Mach numbers for a  $47^\circ$  sweptback wing of aspect ratio 3.5 and taper ratio 0.2. The data up to a Mach number of 1.15 were obtained in the Langley 8-foot high-speed tunnel (reference 1); the data at  $M = 1.6$  were obtained with the same model in the Langley 4-foot supersonic pressure tunnel (unpublished). The fuselage was a cylindrical body with an ogival nose that was just large enough to house the strain-gage balance. Variations in uniform thickness ratio for these wings indicate only minor differences in the lift-curve slope and all wings possess regular although rapid variations in the transonic speed range. In the lower part of figure 1, a comparison has been made of the lift-curve slopes of a wing with a constant thickness ratio of 6 percent and one having a 6-percent-thick section over only the outboard 60 percent of the semispan. The root was thickened to 12 percent and tapered to 6 percent at the 40-percent semispan station. It is of interest that the tapered-in-thickness wing having a thickness ratio at the fuselage juncture of about 10 percent does not show any sudden losses in lift in this range. The theoretical variation of the lift-curve slope (references 2 to 4) has been added to indicate the confidence with which the data can be faired in the Mach number range for which data were unavailable.

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Wings of aspect ratio 6.- An indication of the influence of aspect ratio on this thickness effect is afforded in figure 2. The data for the wing of aspect ratio 6 were obtained from bump tests in the Langley 7- by 10-foot high-speed tunnel. (See reference 5.) For this aspect ratio and sweep angle, it is apparent that a thickness ratio of 9 percent was not sufficient to avoid losses in lift in the transonic range, although the losses are nowhere near so severe as those encountered with the 12-percent-thick wing; however, it will be noted on the lower part of this chart that a tapered-in-thickness wing having a 9-percent-thick root section and a 3-percent-thick tip section was free of these losses and possessed characteristics similar to the constant 6-percent-thick wing. It appears, therefore, that some thickening at the root of a swept wing can be tolerated, provided the outer panel of the wing is thin enough.

It has been observed that the development of lift-slope "buckets" such as those shown for the 9- and 12-percent-thick wings occurs because of a breakdown in flow over the outer wing panel. Consequently, swept wings of high aspect ratio having this characteristic are likely also to have undesirable pitching-moment effects at low lift coefficients as illustrated in figure 3. These are the same wings for which lift-curve slopes are shown in figure 2. The wings exhibiting the very erratic pitching-moment behavior at  $M = 1.0$  are seen to be the same wings that reveal a bucket-type lift-curve-slope variation in this range. The slopes of the curves of pitching moment against lift are plotted for different Mach numbers in the lower left part of the figure to provide an indication of the range over which this difficulty was experienced. The location of the lateral center of pressure is plotted against Mach number in the lower right part of figure 3. The inboard movement of the lateral center of pressure in the vicinity of Mach number 1 for the 9- and 12-percent-thick wings is evidence that the losses in lift are occurring over the outer panels of these wings. It will be noticed that this behavior did not occur on either the 6-percent-thick wing or the 9 to 3 percent tapered-in-thickness wing.

Effect of Sweep Angle.- The effects of thickness on the lift-curve slope are, of course, influenced by sweep angle as well as by aspect ratio. Although systematic data on this effect are scarce, some idea of the interdependency of these factors can be obtained from figure 4. This figure furnishes a convenient guide for estimating those combinations of aspect ratio, thickness ratio, and sweep angle that are likely to provide a smooth variation in lift-curve slope through the transonic range. The preferred combinations of these factors are those falling in the cross-hatched regions below the lines of constant sweep angle. The boundary for the zero-sweep line is well established from experimental data but the boundary for  $45^\circ$  sweep is only qualitative. Attention is called to the fact that the streamwise thickness ratio

constituting the abscissa of this chart may be the root-mean-square value for a wing that is tapered in thickness and plan form.

#### Minimum Drag

Wings of aspect ratio 6.- Data obtained from rocket-model investigations (reference 6 and unpublished) showing the effects of thickness and thickness distribution on minimum drag are presented in figure 5. All wings were of aspect ratio 6, taper ratio 0.6, and were tested on a parabolic body. For purposes of comparison, the body drag has been subtracted and the minimum drag of the wing-fuselage combination less fuselage is presented. Although some mutual interference effects are undoubtedly present at Mach numbers close to 1, it is believed that at  $M = 1.25$  these interference effects are small. Therefore, at this Mach number, the drag is cross-plotted against the square of the root-mean-square thickness ratio of these wings. It appears from this plot that the wings with the thickness modifications are probably contributing an amount of drag consistent with their weighted thickness.

Wings of aspect ratio 3.5.- Another thickness and thickness-distribution study has been completed at transonic speeds in the Langley 8-foot high-speed tunnel (reference 1) and the Langley 4-foot supersonic pressure tunnel (unpublished) and the results are presented in figure 6. These are the same wings for which lift-curve slopes were presented in figure 1. The wing-fuselage minus fuselage drag coefficients (WF-F) show the expected increases with thickness ratio. A cross plot of the data at Mach number 1.6 again demonstrates that the wing with the thickened root is contributing an amount of drag consistent with its weighted thickness. The reason for the delayed drag rise of the wing with the thickened root is not understood. It may be that, even with the small fuselage used on this model, favorable interference effects were encountered.

A study of the effect of sweep angle on the characteristics of the 4-percent wing of this family has also been made and the minimum drag results for the wing-fuselage less fuselage combination are given in figure 7. The results reflect the expected changes with sweep angle insofar as drag rise is concerned. Of some interest is the fact that the drag rise for this series of wings seems to have occurred at just about Mach number 1, as contrasted with the aspect-ratio-6 wings which showed generally increasing drag coefficients up to Mach number 1.4.

The calculated pressure-drag variation as determined from linear-theory charts (references 7 and 8) is shown for each of the swept wings. Even when adjusted to account for frictional effects as was done here, it is evident that these theoretical variations supply an inadequate guide for estimating the magnitude of the maximum drag rise that occurs at slightly supersonic Mach numbers.

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Correlation of minimum-drag results.- Inasmuch as the excess of thrust over drag for a jet-powered airplane equipped with afterburning may be critical at Mach numbers just above 1, a knowledge of the drag in this region is especially important. It has been observed that a number of wings of aspect ratio 4 possess the characteristics exhibited by the wings of aspect ratio 3.5; that is, that the maximum pressure-drag rise occurs very close to  $M = 1$ . A correlation of this drag rise in the vicinity of  $M = 1$  has been made for a number of wings of aspect ratio about 4 by using an empirical approach outlined in reference 9 and the results are shown in figure 8. The lines are those given by the empirical formula at the top of the chart where the thickness is taken in the stream direction. The points represent data from the indicated sources. That some orderly variation seems to exist for the pressure-drag rise in the vicinity of  $M = 1$  seems evident but whether such correlations can be extended to other aspect ratios or even to other sections must await further study.

#### Pitching-Moment Behavior

Innumerable low-speed investigations have demonstrated the instability occurring at high and even moderate lift coefficient for those combinations of sweep and aspect ratio that may otherwise be attractive from performance considerations. It may be of interest to examine the effects of Mach number on this pitching-moment behavior (fig. 9). The wings of aspect ratio 6 were tested on the transonic bump in the Langley high-speed 7- by 10-foot tunnel (reference 5) and both were tapered in thickness as indicated, although thickness distribution has been found to have little effect on the particular characteristics under discussion. The wings of aspect ratio 4 were investigated in the Langley 8-foot high-speed tunnel (reference 10). It is seen that the severity of the pitching-moment reversal is lessened and delayed to higher lift coefficients as the transonic range is negotiated. Of particular importance is the rearward movement of the aerodynamic center at transonic speeds which tends to minimize the importance of the pitching-moment changes that are eventually encountered at high lift coefficients. It would appear, therefore, that, as far as the pitching-moment behavior is concerned, the most critical speed regime for high-aspect-ratio swept wings is likely to be at high subsonic Mach numbers where the pitching-moment characteristics are similar to those encountered at low speeds. Of course, tail-on tests will be required before any such conclusion can be stated with certainty.

## METHODS FOR IMPROVING PERFORMANCE AND STABILITY

## Performance

Twist and camber.- The use of twist and camber has been explored as one method of improving both the stability and performance of swept wings. Some results of one investigation of this kind are shown in figure 10. This investigation was conducted in the 12-foot pressure tunnel at the Ames Laboratory at a Reynolds number of two million. (See reference 11.) Both wings were of identical plan form. One had an NACA 64A010 section perpendicular to the quarter-chord line, whereas the other had an NACA 64A810 section perpendicular to the quarter-chord line and was washed out at the tip as indicated. Insofar as any improvement in pitching-moment behavior is concerned, the cambered and twisted wing was not much better than the basic wing at  $M = 0.85$  and at  $M = 0.94$  it was worse. From this and other investigations, it appears that something more than camber and twist is needed to improve the high-speed pitching-moment behavior of swept wings at high lift coefficients. To a certain extent, however, the study of this problem is complicated because at low speeds those wings which possess a fairly large amount of camber and twist show large Reynolds number effects. For high-speed investigations, the Reynolds numbers have been relatively low. It is evident, however, that, at this Reynolds number at least, a substantial improvement in performance at  $M = 0.85$  resulted from the use of camber and twist even though for this wing the beneficial effect had vanished at  $M = 0.94$ .

It has been possible to twist and camber wings for improved performance at supersonic speeds, although specific examples of this kind are admittedly few. Figure 11 presents the maximum lift-drag ratios obtained for three wings designed from lifting-surface considerations specifically for operation at supersonic speeds. The aspect-ratio-3 wing was tested on the transonic bump at a Reynolds number of about one million and was designed for a uniform pressure loading for a lift coefficient of 0.25 at a Mach number of 1.1 (reference 12). The aspect-ratio-3.5 wing was designed for a uniform loading at a lift coefficient of 0.25 at a Mach number of 1.53 and was tested in the Ames 6- by 6-foot supersonic tunnel at Reynolds numbers up to 3.5 million (reference 13). The arrow wing was tested in the Langley 9-inch supersonic tunnel and was designed for  $c_{l_i} = 0.20$  at  $M = 1.62$  (reference 14). For each of these twisted and cambered wings a comparison with the corresponding flat wing was available. It is evident that some improvement was obtained throughout the Mach number range, at least for the Reynolds numbers of the tests. In fact, the configurations with aspect ratio 3.5 and 2.6 represent the highest lift-drag ratios that have been obtained in their respective Mach number ranges for purely swept wings. The

results for the aspect-ratio-3 wing are for wing-alone data and the values of the lift-drag ratio are merely indicative of what might be accomplished.

Nose flaps.- One convenient device for studying camber effects is the nose flap. Preliminary tests have been made to explore the use of nose flaps at transonic speeds and the results obtained with a small semispan model in the Langley high-speed 7- by 10-foot tunnel (unpublished) are shown in figure 12. The ratio  $\frac{[(L/D)_{\max}]_{\delta_n}}{[(L/D)_{\max}]_{\text{flat}}}$  indicates that the advantages of nose droop alone essentially disappeared at  $M = 0.975$ ; however, with  $3.5^\circ$  of washout, the beneficial effects of nose-flap deflection were apparent at the highest Mach number for which data were available.

### Stability

Regardless of the improvements in performance that some of the modifications discussed may lead to, the problem of instability at high lift coefficients is still the one for which it is most difficult to find a successful solution. Inasmuch as the use of leading-edge slats has been found to be an unattractive method for overcoming pitching-moment difficulties at high speeds, a considerable amount of research is being devoted to other methods of improving the high-speed, high-lift stability characteristics of swept wings.

Fences and chord-extensions.- Most of the progress that has been made in improving the pitching-moment characteristics of swept wings so far is attributable to "fixes." Fences, for example, are known from low-speed investigations to result in considerable improvements in stability. More recently, chord-extensions have indicated considerable promise at low speeds. Some recent results have been obtained on the relative effects of fences and chord-extensions at high subsonic Mach numbers and the results of one such investigation are shown in figure 13. The investigation described in figure 13 was conducted on a sting-supported model in the Langley high-speed 7- by 10-foot tunnel at a Reynolds number of 3.8 million (unpublished). The particular chord-extensions used were determined from extensive low-speed investigations on this wing. It will be observed that at  $M = 0.8$  the single fence was nearly as good as the chord-extension in improving the characteristics up to a lift coefficient of 0.8 but, at higher lift coefficients and especially at higher Mach numbers, the chord-extension was more effective. Data (fig. 14) for a very similar wing but with a 0.15c chord-extension were obtained at a Reynolds number of  $6 \times 10^6$  in the Langley 16-foot transonic tunnel (unpublished). At  $M = 0.8$ , the improvement



obtained with the chord-extension was remarkable and, although the benefits of the chord-extension were less at high Mach numbers, some improvement is still evident at a Mach number of 0.98.

Inasmuch as the nose-flap results discussed earlier showed performance gains, the effect of chord-extensions and fences was also investigated in the Langley high-speed 7- by 10-foot tunnel with nose flaps deflected on the sting-supported model described in figure 13. The results of the nose-flap study are shown in figure 15 and it will be observed that the model is the same as that described in figure 13, except for the 0.20c leading-edge flaps that were drooped  $6^\circ$ . A comparison of the pitching-moment data with that of figure 13 shows that the nose flaps in themselves are relatively ineffective in producing any improvements in stability but that some improvement was obtained with a chord-extension or a fence. In general, the chord-extension was more effective than the fence for the wing regardless of whether the nose was drooped or not.

The effect of the chord-extensions on the maximum lift-drag ratio of this model, both with and without nose droop, is shown in figure 16. Of particular interest is the fact that whereas the chord-extension on the flat wing decreased the  $(L/D)_{\max}$  at the higher Mach number for the flat wing, the chord-extension on the wing with nose flaps actually increased the  $(L/D)_{\max}$  throughout the Mach number range. Although these arrangements are not considered optimum, they do indicate one possibility for improving both the performance and stability of swept wings. At the present state of our knowledge, these devices must be explored individually on each wing for which they are proposed.

### RÉSUMÉ

In brief summary, data have been presented which indicate that:

1. Swept wings having appreciable taper in thickness appear to have characteristics similar to those which would be expected from a similar wing of constant thickness ratio equal to the root-mean-square thickness ratio of the tapered wing.
2. The maximum lift-drag ratios of sweptback wings can be improved by the judicious addition of camber and twist up to a Mach number of the order of 2.4.
3. The unstable pitching-moment variation developed by swept wings at high lift coefficients is characteristic throughout the subsonic and transonic Mach number range, although the severity of this effect appears to be somewhat diminished at transonic speeds partly as a result of the rearward movement of the aerodynamic center.

4. Arrangements of fences and chord-extensions can be developed on particular swept wings which materially improve the stability characteristics at high speeds without excessive loss in performance.

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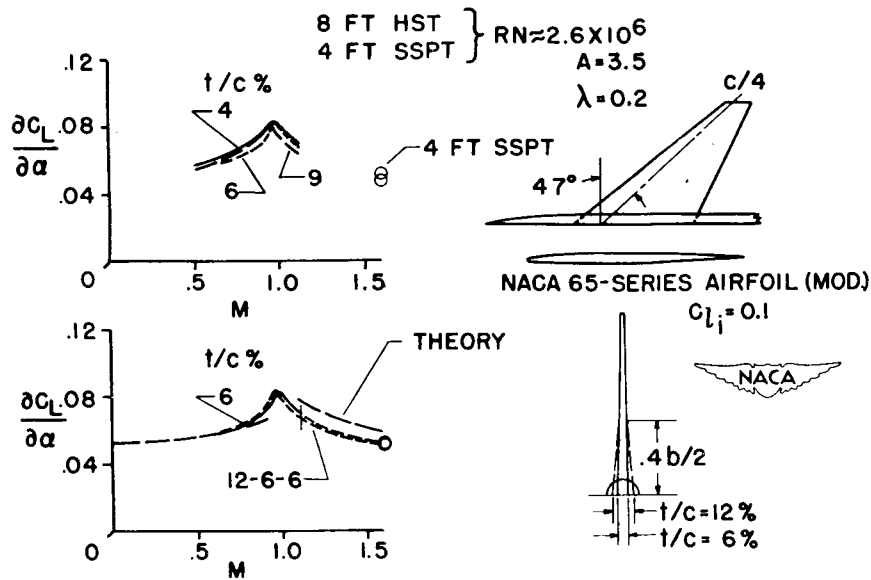


Figure 1.- Effect of thickness and thickness distribution on lift characteristics.  $A = 3.5$ .

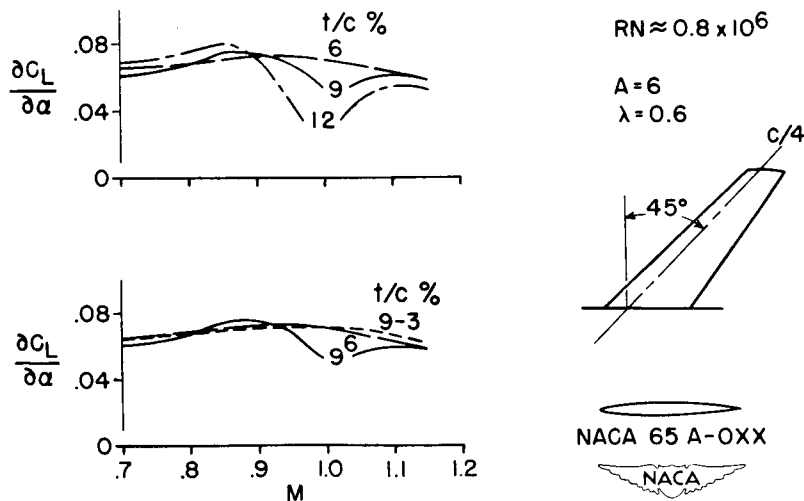


Figure 2.- Effect of thickness and thickness distribution on lift characteristics.  $A = 6.0$ .

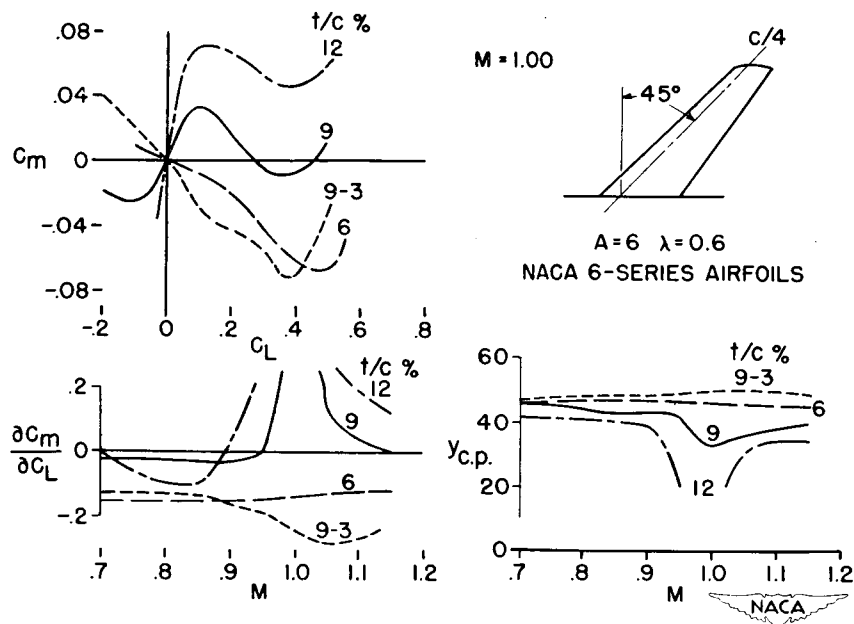


Figure 3.- Effect of thickness and thickness distribution on the pitching-moment and lateral center-of-lift characteristics.  $A = 6.0$ .

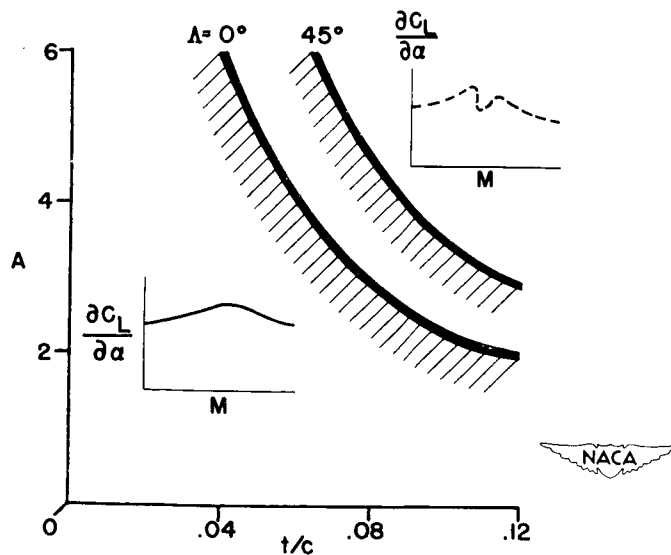


Figure 4.- Influence of aspect ratio, thickness ratio, and sweep angle on the type of transonic lift-slope variation.

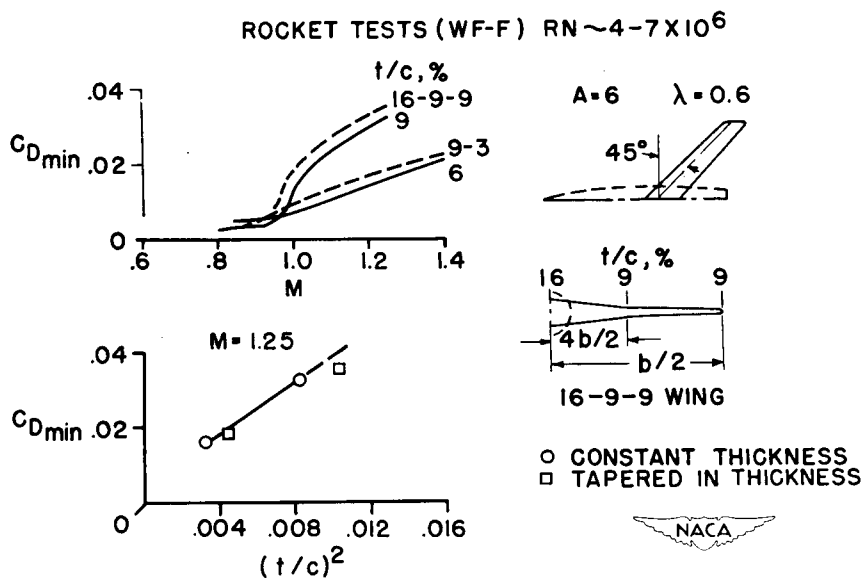


Figure 5.- Influence of thickness and thickness distribution on minimum drag.  $A = 6$ .

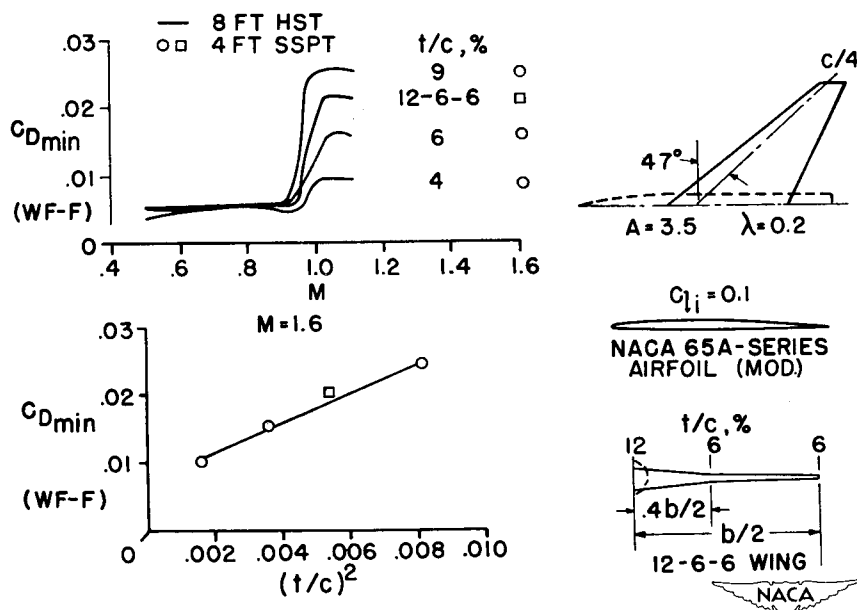


Figure 6.- Influence of thickness and thickness distribution on minimum drag.  $A = 3.5$ .

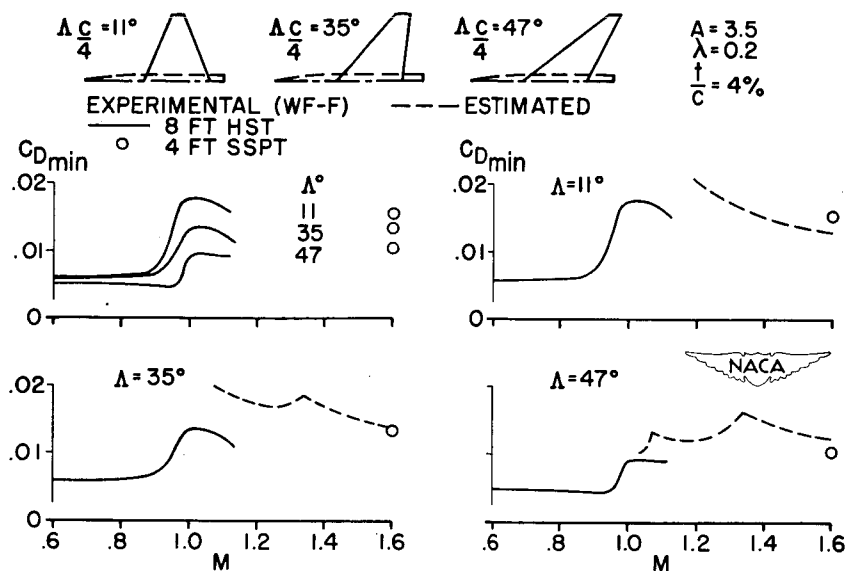


Figure 7.- Effects of sweep angle on minimum-drag characteristics.  $A = 3.5$ .

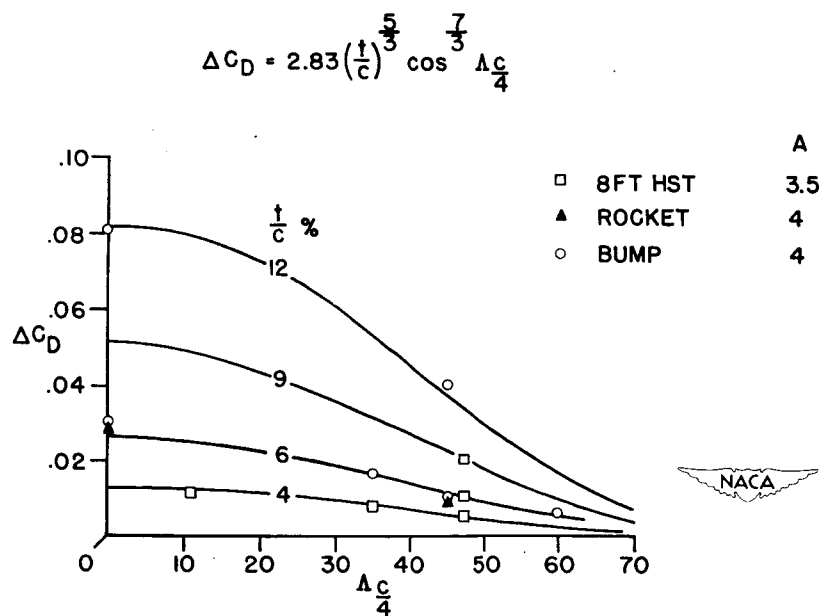


Figure 8.- Correlation of wave drag at low supersonic speeds.



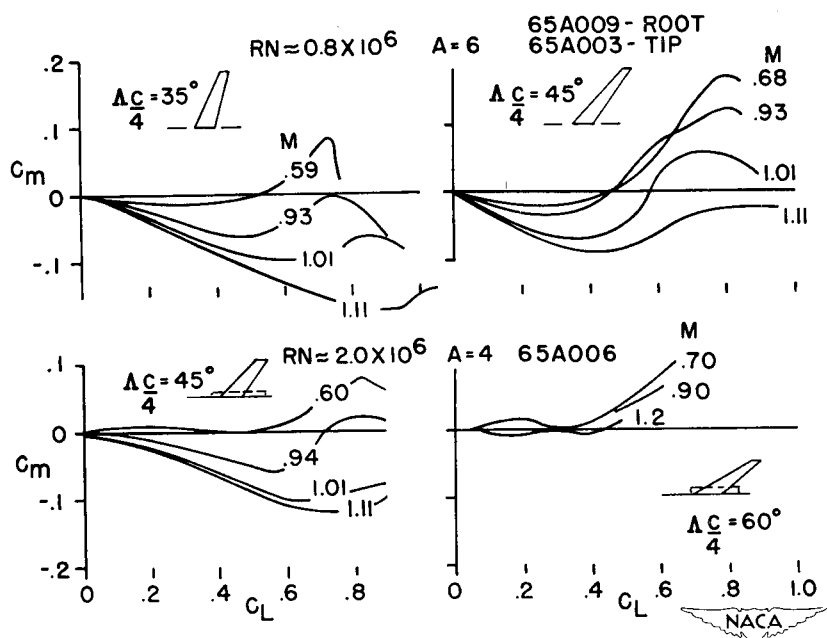


Figure 9.- Mach number effect on the pitching-moment characteristics of a number of swept wings.

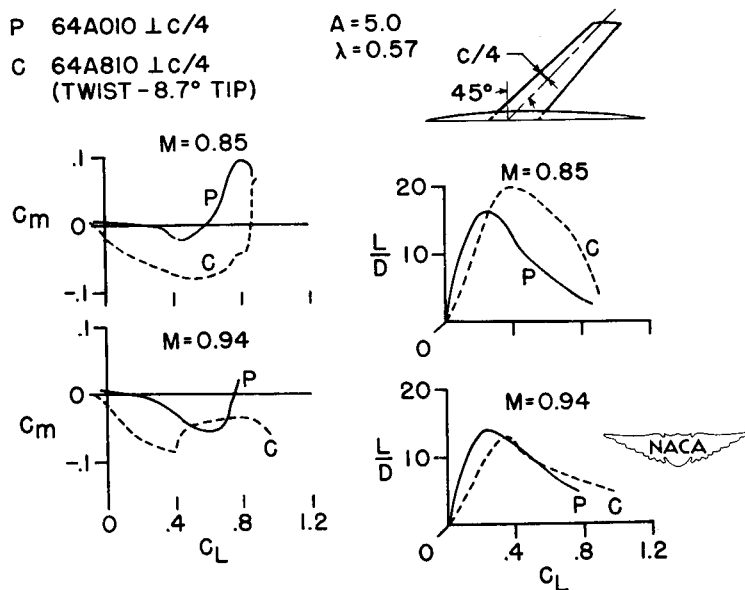


Figure 10.- Comparison of pitching moment and lift-drag ratios of a flat and a twisted and cambered wing.

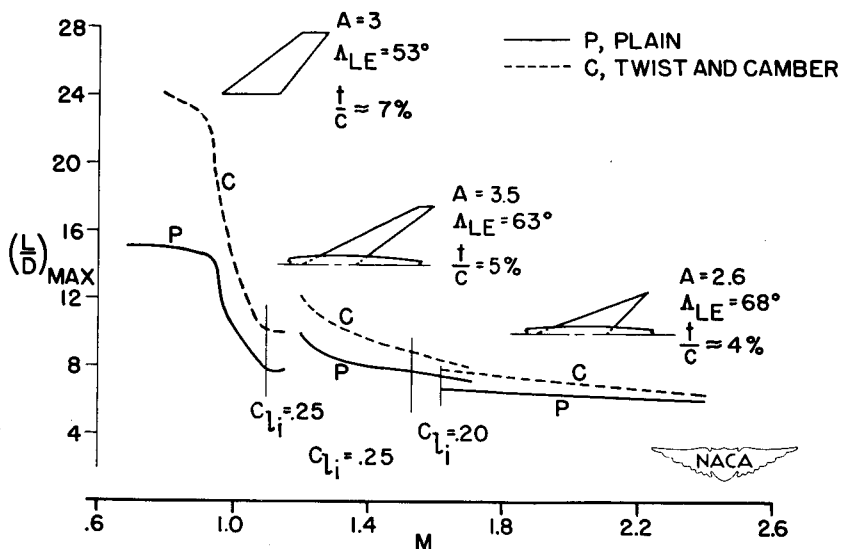


Figure 11.- Summary of effects of twist and camber on the performance of swept wings.

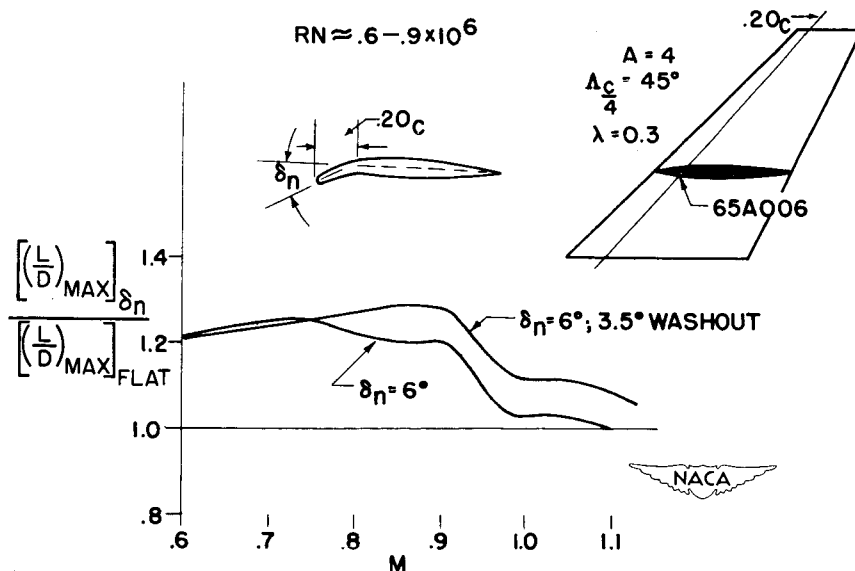


Figure 12.- Relative improvement in  $(\frac{L}{D})_{MAX}$  of flat wing attributable to nose-flap deflection.

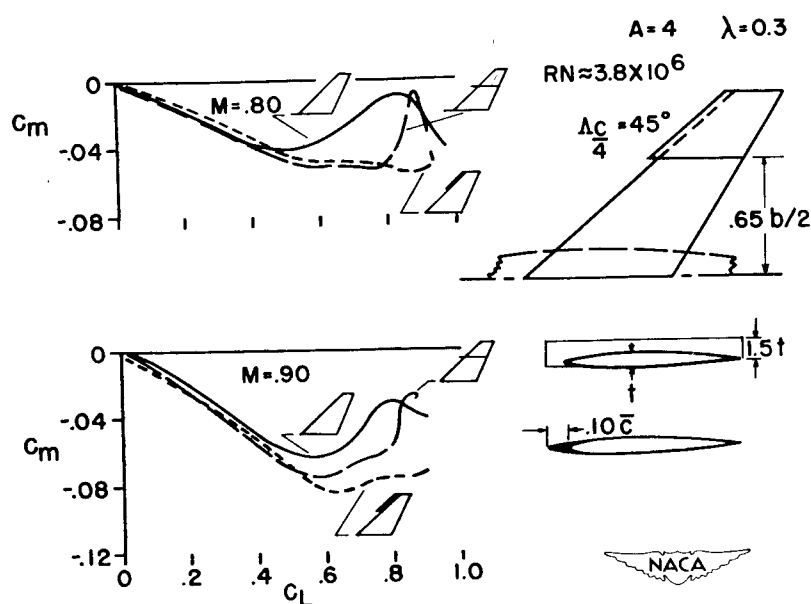


Figure 13.- Separate effects of chord-extensions and fences on the pitching-moment characteristics.  $RN = 3.8 \times 10^6$ .

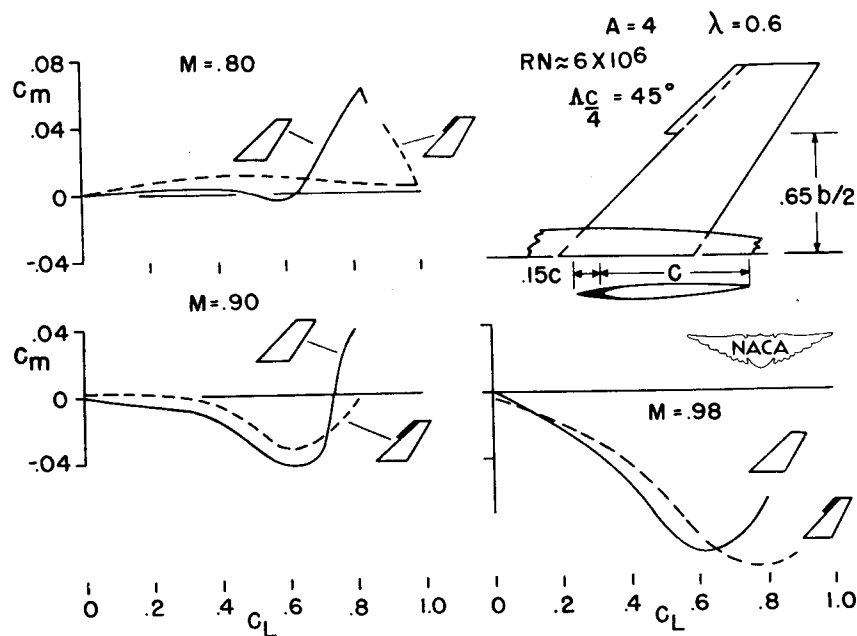


Figure 14.- Effect of chord-extensions on the pitching-moment characteristics of a swept wing at transonic speeds.  $RN = 6.0 \times 10^6$ .

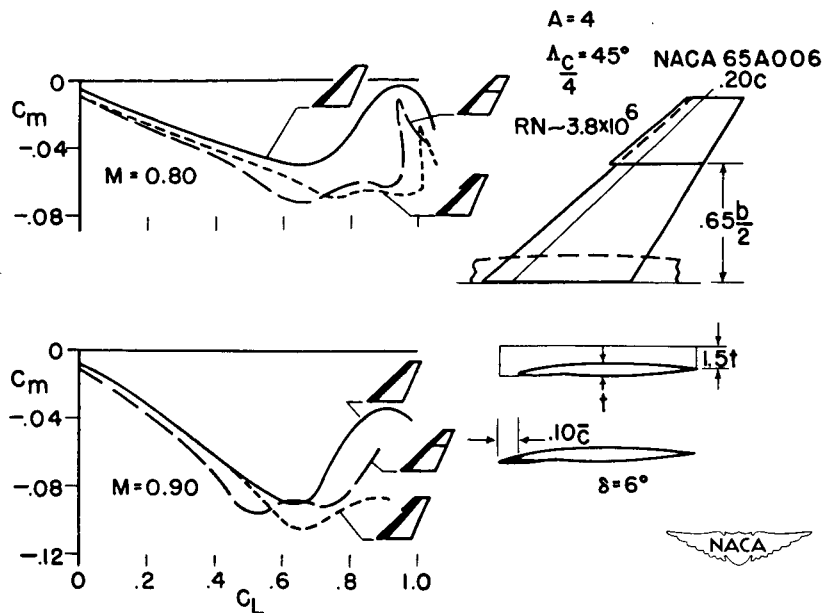


Figure 15.- Separate effects of chord-extensions and fences on the pitching-moment characteristics of a wing with nose flaps.  $RN = 3.8 \times 10^6$ .

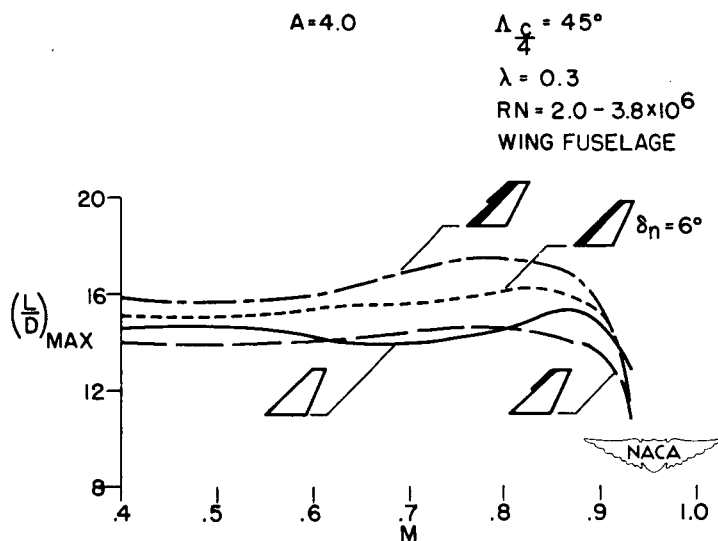


Figure 16.- Effects of chord-extensions and nose flaps on  $(L/D)_{\max}$ .

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# RESEARCH MEMORANDUM

AN ASSESSMENT OF THE AIRPLANE DRAG PROBLEM  
AT TRANSONIC AND SUPERSONIC SPEEDS

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## RESEARCH MEMORANDUM

AN ASSESSMENT OF THE AIRPLANE DRAG PROBLEM  
AT TRANSONIC AND SUPERSONIC SPEEDS<sup>1</sup>

By Charles J. Donlan

## SUMMARY

The airplane drag problem at transonic and supersonic speeds is discussed. The area rule is shown to be a powerful tool that provides guidance for the airplane designer in selecting aerodynamic features compatible with low wave drag. The factors influencing the drag of bodies of revolution are reviewed and the effectiveness in reducing wave drag of various methods of improving the cross-sectional area distribution of aircraft configurations is illustrated. It is demonstrated that, irrespective of the method adopted for improving area distribution, a high effective fineness ratio and smooth area progressions along the equivalent body are essential to the achievement of low drag.

## INTRODUCTION

The airplane designer is interested in achieving the lowest possible drag for his configuration. For aircraft that fly at less than the speed of sound, this requires the minimization of the profile drag - essentially friction drag - and the induced drag. For aircraft that are to fly at transonic and supersonic speeds, another source of drag must be considered - the wave-making drag. The drag from this source alone can create formidable design problems as illustrated in figure 1. For the flight condition assumed, the drag coefficient associated with level flight increases markedly with Mach number as the speed of sound is approached and exceeded. It is evident that, while the friction-drag component and the trim-drag component (including induced drag) are still of significance at supersonic speeds, the wave-drag component is responsible for the large increase in drag coefficient shown. The wave-drag component is primarily independent of the lift and thus can usually be analyzed for the zero-lift condition ( $C_{D0}$ ). Because it is of such great

<sup>1</sup>This paper was originally prepared as an informal talk and was presented at a meeting of the NACA Committee on Aerodynamics held at the Langley Aeronautical Laboratory on April 28, 1954.

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importance in establishing the level of the drag curve, a large research effort has been devoted toward methods of reducing this source of drag. This paper, therefore, is concerned primarily with the progress that has been made in coping with the wave-drag problem.

## SYMBOLS

|                       |                                                  |
|-----------------------|--------------------------------------------------|
| $A$                   | cross-sectional area of body; also, aspect ratio |
| $b$                   | wing span                                        |
| $C_D$                 | drag coefficient                                 |
| $C_{D_{min}}$         | minimum drag coefficient                         |
| $C_{D_0}$             | zero-lift drag coefficient                       |
| $\Delta C_{D_{CALC}}$ | calculated drag-rise coefficient                 |
| $\Delta C_{D_{EQ.B}}$ | drag-rise coefficient of equivalent body         |
| $\Delta C_{D_{EXP}}$  | experimental drag-rise coefficient               |
| $(\Delta C_D)_F$      | drag-rise coefficient based on frontal area      |
| $\Delta C_{D_{WB}}$   | drag-rise coefficient of wing-body configuration |
| $C_T$                 | thrust coefficient                               |
| $c$                   | airfoil chord                                    |
| $d_{max}$             | maximum body diameter                            |
| $l$                   | body length                                      |
| $M$                   | Mach number                                      |
| $t$                   | airfoil thickness                                |
| $\Lambda_{LE}$        | leading-edge sweep                               |

$\lambda$  taper ratio

$X$  distance from nose of body

### THE AREA RULE

The key to understanding why certain arrangements of airplane components result in less wave drag at transonic speeds than others is furnished by the aerodynamic principles expressed so conveniently in the so-called area rule. These important aerodynamic principles were demonstrated experimentally less than two years ago by Whitcomb (ref. 1) and the effect on airplane design philosophy has been noteworthy. In this brief period of time nearly every transonic and supersonic airplane design has been influenced in some way by area-rule concepts. The reason for its acceptance is that the area rule provides the airplane designer with a tool that permits him to assess those features of a design that should be sought for - or avoided - if wave drag is to be kept to a minimum.

The equivalent-body concept.- Figure 2 illustrates the basic tenet of the area rule, which states that the wave drag of an airplane configuration depends primarily on the longitudinal distribution of the total cross-sectional area. This concept results in the proposition that the wave drag of a simple equivalent body of revolution (that is, a body having the same longitudinal distribution of total cross-sectional area) would be the same as the drag of the more complex wing-body arrangement. The cross-sectional distributions shown are obtained by passing planes perpendicular to the body axis. This procedure is correct for a design Mach number of 1.0. Although originally developed as a Mach number 1.0 concept, the area rule has been extended to supersonic speeds by Whitcomb (ref. 2) and R. T. Jones (ref. 3), who has elegantly molded this concept into the framework of linearized body theory. In the supersonic applications of the area rule, the geometric considerations involve the cross-sectional distributions formed by planes tangent to the Mach cone at various orientations about the longitudinal axis of the configuration.

Experimental verification.- The equivalent-body concept has been subjected to experimental verification. In figure 3 (from ref. 4), the measured drag-rise increments at  $M = 1.03$  for various swept-wing, delta-wing, and unswept-wing-body combinations and complete airplanes (all symbolized by  $\Delta C_{D_{WB}}$ ) are compared with the experimental increments for the equivalent bodies of revolution ( $\Delta C_{D_{EQ.B}}$ ). The aspect ratios of the wings are 4 or less and the wing thickness-chord ratios (streamwise) are 0.07 or less. Deviations from exact agreement are apparent but, in general, qualitative agreement exists between the drag-rise increments, and thus the basic tenet of the area rule appears to be substantiated.



Progress in calculations.- Utilizing the linear-theory development of Jones (ref. 3), Holdaway (ref. 5) has computed the drag rise of several wing-body-tail configurations at various Mach numbers and compared his results with the experimental drag-rise values obtained with freely falling models. This comparison is presented in figure 4. The agreement is good in the regions where the drag rise is lowest, although precise determination of drag - for example,  $\Delta C_D \lesssim 0.0015$  - should not be expected from any computation based on an approximate theory. It is significant, however, that the calculated results appear to be as reliable for drag predictions near  $M = 1$  as actual experiments with equivalent bodies (fig. 3), and for supersonic applications the theoretical procedure is to be preferred. The theoretical method outlined in reference 5 is perhaps the only published procedure available that will handle transonic and supersonic calculations of this nature in a relatively routine manner. Nelson and Stoney (ref. 6) have published an empirical correlation of drag-rise data that permits estimates of drag-rise increments around  $M = 1$  but the method has not been extended to supersonic speeds.

#### BODY DRAG

Knowing that the transonic drag rise of his airplane will reflect the merits of its equivalent body, it is obvious that the airplane designer would want his design to have a low-drag equivalent body shape. Figure 5 shows some of the factors governing the drag of bodies at transonic and supersonic speeds and illustrates the penalties imposed by low fineness ratios and deviations from low-drag shapes. The drag-rise coefficient  $(\Delta C_D)_F$  in this figure is based on body frontal area. The points represent all types of smooth parabolic bodies for which rocket-model drag data are available. The lowest drags are obtained with bodies possessing smooth parabolic profiles. The minimum drag variation will be found to be inversely proportional to the square of the fineness ratio, as theory predicts. For a given fineness ratio ( $l/d_{max} = 9$ , for example), the bodies well removed from the minimum drag curve are found to possess steep gradients in their area distributions, as shown. It is clear that the requirement for low drag is a high effective fineness ratio and a smooth area distribution free of rapid rates of change of area along the body. The position of the maximum diameter for minimum drag is a function of the base area and for bodies pointed on either end it is at 0.5 $l$ . One could substitute a Sears-Haack shape or some other profile for the parabolic shape used here with similar results. At the higher fineness ratios that are required for really low drag, however, the particular choice of body shape is a minor variable as far as drag is concerned.

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## AIRPLANE DRAG

Figure 6 is prepared in the same fashion as figure 5 except that the points are for complete airplane configurations. The drag-rise coefficient  $(\Delta C_D)_F$  again is based on the frontal area and overall fineness ratio of the equivalent bodies of revolution. The minimum measured body-drag curve and the theoretical variation are replotted from figure 5. The open points represent the drag of basic or initial configurations. The closed and half-closed points are experimental results obtained with modifications that were made to the basic configuration in order to improve the effective fineness ratio of the equivalent body. These modifications were of several kinds. It is of interest to examine a few of the cases shown to highlight some of the experiences of the NACA in utilizing various methods such as body indentation, body lengthening, body buildup, and the juggling of the aircraft components themselves in order to approach the high-fineness-ratio smooth equivalent body for which there is no substitute if low drag is to be achieved.

Example A. - A partial case history of a delta-wing model identified as example A is summarized in figure 7. The composition of the area diagram of the prototype configuration is shown in the upper left-hand portion of the figure. Note how each element piles on top of the other, especially at the rear of the fuselage. The rate of change of area with length at the rear of the fuselage is thus very rapid and the large suction forces associated with this area gradient at transonic speeds is responsible for the high drag rise of the prototype configuration. In the lower right of the figure, the area distribution of the prototype is compared with the area distribution of the modification. By "waisting" or indenting the body to accommodate the wing, partially, and by lengthening the fuselage, the area distribution of the revised configuration is obviously improved and this improvement is reflected in the lower drag levels for the revised design. Even larger reductions in drag are possible if, in addition to the modifications incorporated in the revised design, the nose of the aircraft could be changed in the region shown in the area diagram.

In order to show what these drag gains mean in airplane performance, the drag in pounds has been evaluated for a selected flight condition and compared with the available thrust in figure 8. It will be seen that, while the prototype will just about attain sonic speed in level flight, the revised airplane could fly level at  $M = 1.15$ . More startling performance gains are possible with the improved nose configuration.

Example B. - Figure 9 illustrates how adding volume to an unswept-wing configuration (example B) so as to improve the area distribution can result in a lower drag configuration. The left plot of the figure

illustrates the  $M = 1$  modification that is example B on figure 6. The right portion of the figure illustrates a supersonic application at  $M = 1.14$ . The experiments were conducted with free-fall models and the calculations were carried out by Holdaway by use of the procedures in reference 5. The volume is added in such a manner that a Sears-Haack minimum-drag shape is created although for the  $M = 1.14$  case, such a shape must necessarily represent only an average distribution. The dashed curve represents the drag of the original configuration. The points are for the modified configuration. The solid lines are from theoretical calculations. The drag improvements are significant and the agreement between the experiments and the calculations remarkable.

Example C.- In figure 10 is shown an example of how adding volume rearward of the maximum cross section of a swept-wing airplane (example C) reduced the severe area gradient and thus significantly reduced the drag rise. The volume is added fairly symmetrically around the fuselage. Eighty percent of the inlet-stream-tube area was allowed for in preparing the area diagram. The wing-root inlet location precluded any improvement in the area diagram ahead of the wing in this case.

It is of interest that, in this case, the transonic drag was reduced by virtue of an improvement in the afterbody area distribution despite the fact that the maximum cross-sectional area was actually increased somewhat. Although not shown in this figure, some reduction in the drag level has been found to be still evident at a Mach number of 2.0, notwithstanding that the modification was laid out for a Mach number of 1.0.

Example D.- Thus far, the examples have all been of configurations characterized by engines installed in the fuselage. An interesting example of the application of area-rule principles to a configuration with nacelles (example D) is summarized in figure 11. The undesirable superposition of components on the original arrangement is evident from the area diagram on the left-hand plot of the figure. The modified design at the right uses elements in themselves of high fineness ratio and so positioned that the resulting area distribution approaches a parabola over much of its length. Although the area distribution was laid out for  $M = 1$ , the components are all of such excellent proportion in themselves and the contouring is accomplished over such long lengths and with such gentle slopes that the distribution apparently remains good even for supersonic speeds up to  $M = 1.4$ , as the drag results indicate. Even in this case, however, the irregularities in the area distribution are costly and the drag of the equivalent body is actually considerably higher than that of a true parabolic body of equal fineness ratio, as can be seen from figure 6.

It is not always possible to use a completely symmetrical arrangement such as was used here and items such as off-center position of the wing, incidence of the wing, and noncircular cross sections constitute items for continued research.

## LOW-DRAG ARRANGEMENTS

In the examples cited, the effectiveness of certain kinds of modifications in reducing the zero-lift drag has been illustrated. Such applications have not been 100 percent successful and continued research is needed to determine how the drag of such configurations can be made to approach more nearly that which might be expected of an idealized area distribution. There are a few examples, however, of wind-tunnel models that have evidenced a very low drag rise. The wings of these configurations were also twisted and cambered in order to minimize the drag at lift. Thus, these configurations have yielded about the highest lift-drag ratio thus far obtained at transonic and supersonic speeds. The maximum lift-drag ratios  $(L/D)_{MAX}$  are shown in figure 12. Of special interest is the wing-body combination designed for  $M = 1.4$  (ref. 2) that uses a wing with a streamwise root thickness of 12 percent chord. This design was made possible by careful contouring of the wing-body juncture according to supersonic area-rule concepts. Up to  $M = 1.4$ , the characteristics of this wing compare very favorably with the  $63^\circ$  swept wing of thinner and more flexible construction that was designed for  $M = 1.53$  (ref. 7). Details of the  $68^\circ$  swept wing designed for  $M = 1.6$  can be found in reference 8. Figure 12 also illustrates the well-known fact that the values of  $(L/D)_{MAX}$  obtainable at supersonic speeds - at least by conventional means - are low compared with subsonic values.

## CURRENT RESEARCH TRENDS

The possibility of obtaining more complete cancellation of wave drag for wing-body combinations operating at supersonic speeds is being explored analytically and experimentally by the NACA. Some recent theoretical work of Heaslet and Lomax of the Ames Laboratory is of interest. Briefly, they are exploring the effectiveness of distorting the cross-sectional shape in varying degrees along the body length according to a calculated pattern in order to achieve a more substantial reduction in drag. The NACA is continuing experimental work of a similar nature on the effects of asymmetric wing-body modifications under lifting conditions. Important research on the interference effects of jets on drag is under way. The problem of handling inlet air and assessing its effect on the drag is also being actively pursued. New methods of analysis are being explored. Most of these developments in general have not reached a stage where concrete recommendations regarding their use can be made and are mentioned here only to emphasize the fact that this is a rapidly changing field of research, in which many people are working and new contributions may be expected.

## CONCLUSIONS

At the present time, there are certain conclusions that can be drawn relative to the work done thus far on the problem of reducing wave drag, as follows:

1. The area rule is a powerful tool that provides guidance for the airplane designer in selecting aerodynamic features compatible with low wave drag.
2. Analytical methods have been developed that permit quantitative evaluation of the wave-drag level likely to be experienced with a given design.
3. For any airplane configuration, a high effective fineness ratio and smooth area progressions along the equivalent body for the design Mach number are essential to achieving a low wave-drag level.
4. Continued research is needed especially to exploit the possibilities of more complete cancellation of wave drag and regarding the effects of jet inlet and exit flows on airplane drag.

Langley Aeronautical Laboratory,  
National Advisory Committee for Aeronautics,  
Langley Field, Va., June 8, 1954.

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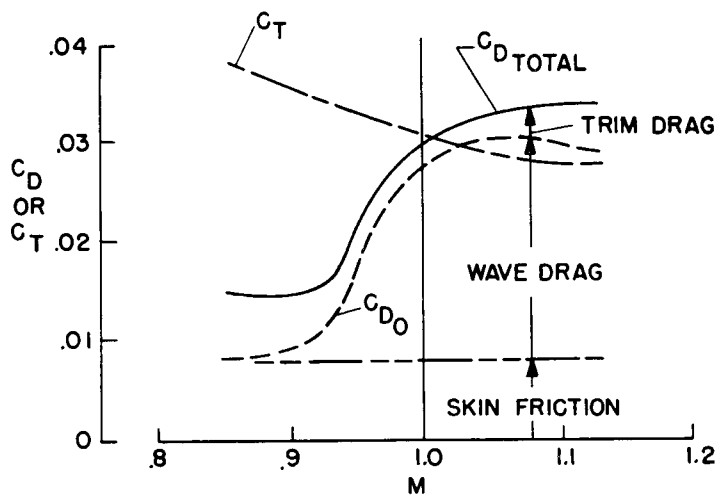


Figure 1.- Composition of drag for delta-wing design in level flight at an altitude of 35,000 feet.

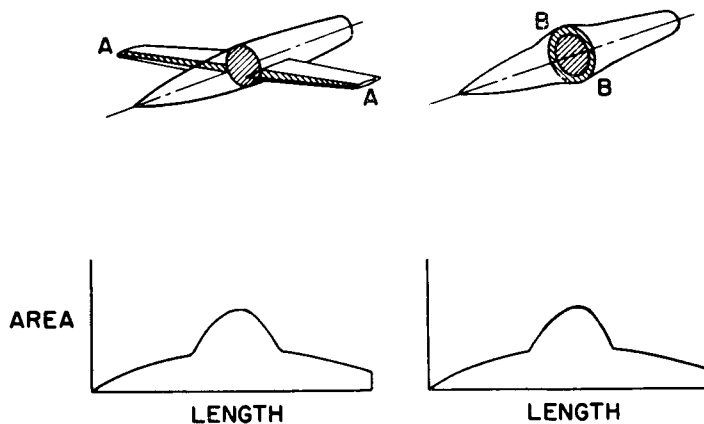


Figure 2.- Wing-body combination and equivalent body.

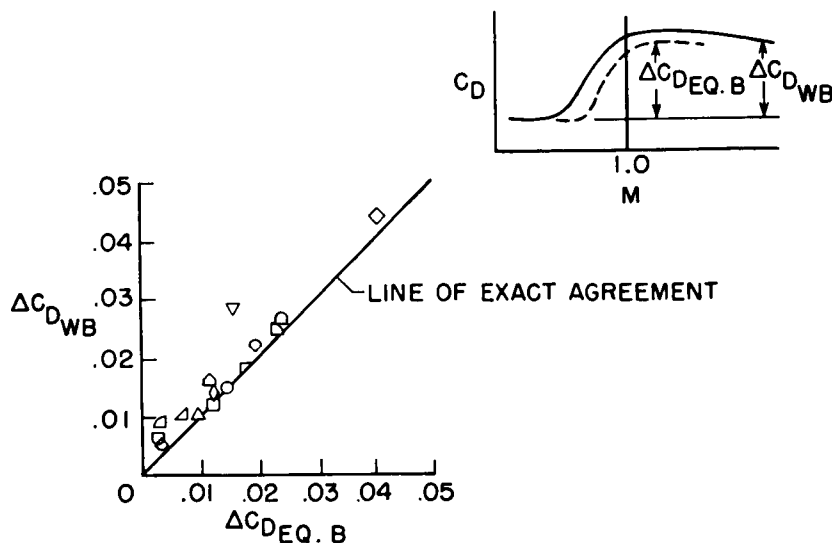


Figure 3.- Comparison of experimental drag-rise coefficients at  $M = 1.03$ .

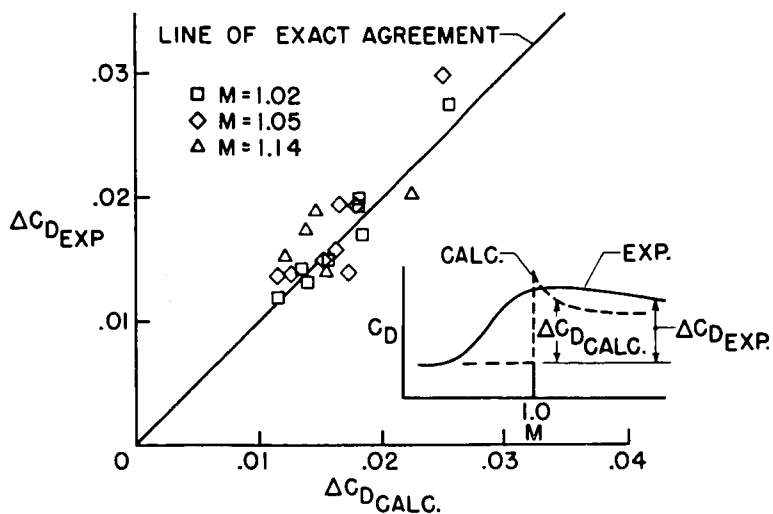


Figure 4.- Comparison of calculated and experimental drag-rise coefficients.



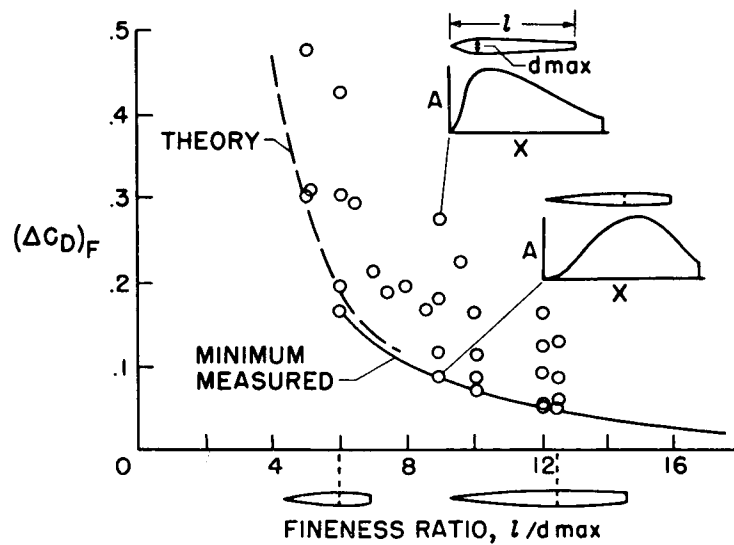


Figure 5.- Effects of fineness ratio and profile shape on body drag.

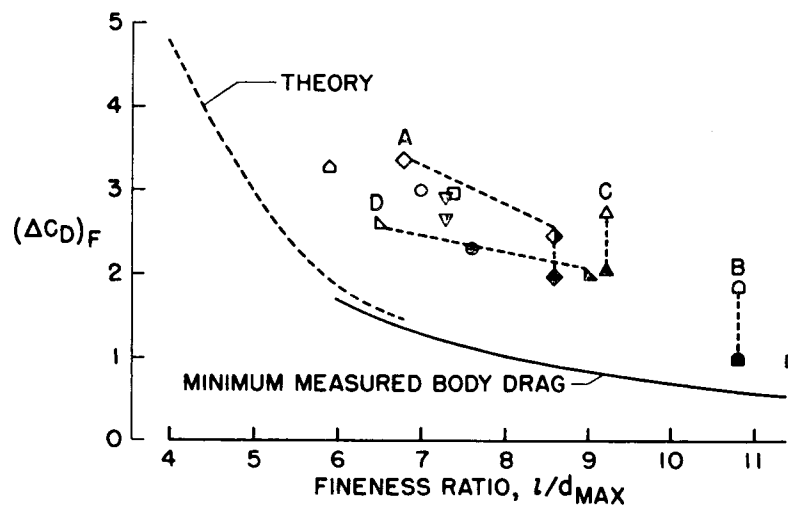


Figure 6.- Airplane drag rise based on fineness ratio and frontal area of equivalent body.

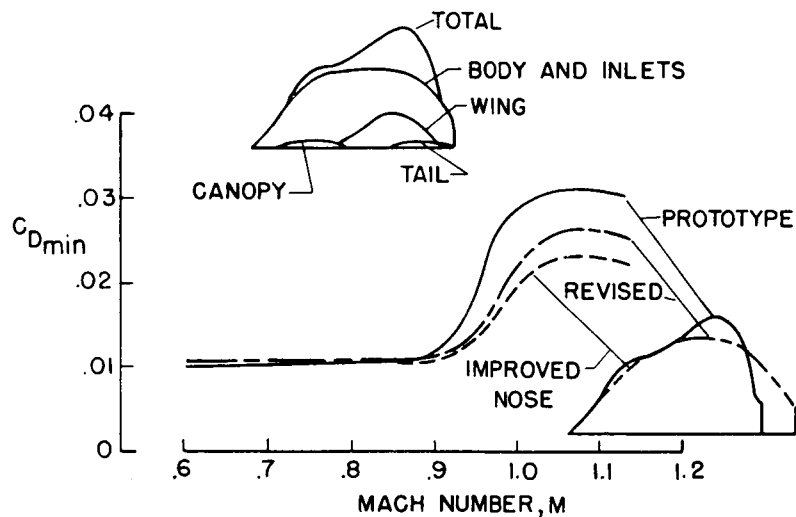


Figure 7.- Example A. Effect of body modifications on delta-wing airplane.

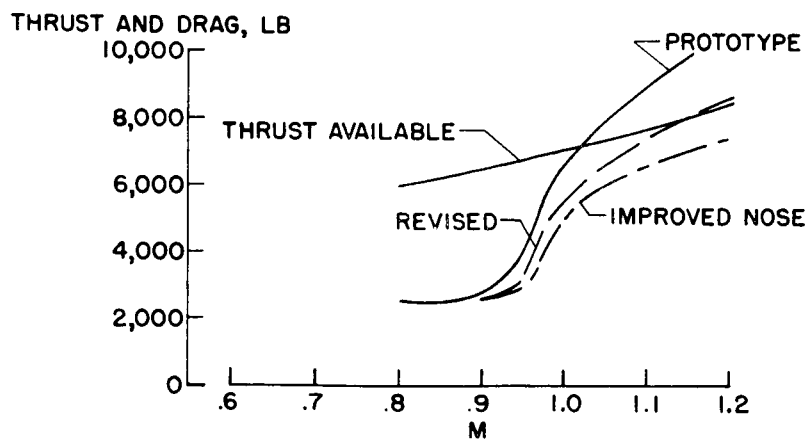


Figure 8.- Effect of body modification on performance of delta-wing airplane at 35,000 feet.

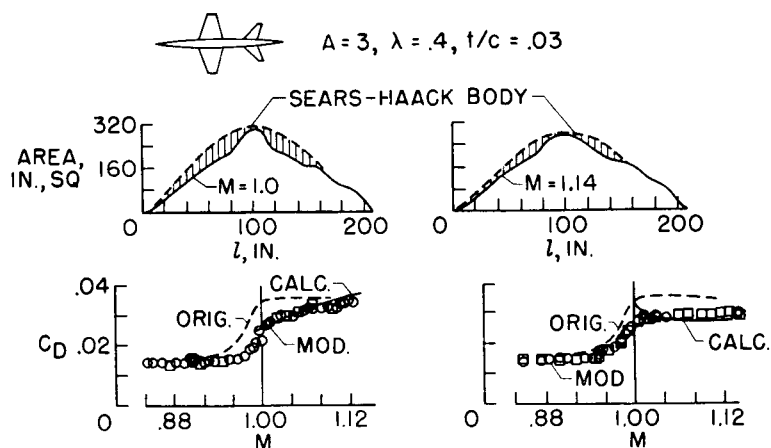


Figure 9.- Example B. Effect of adding volume to unswept-wing airplane.

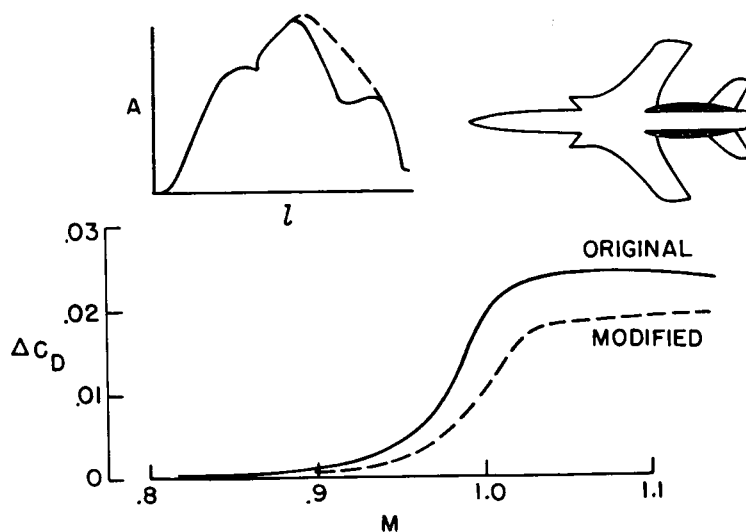


Figure 10.- Example C. Effect of adding volume to swept-wing airplane.

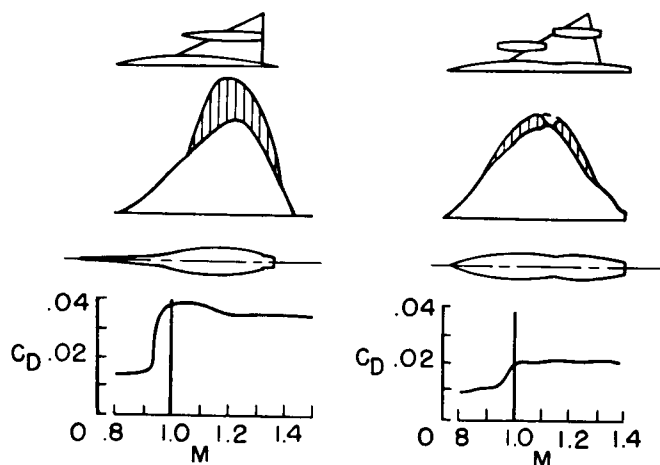


Figure 11.- Example D. Applications of area-rule concepts to multiengine airplane.

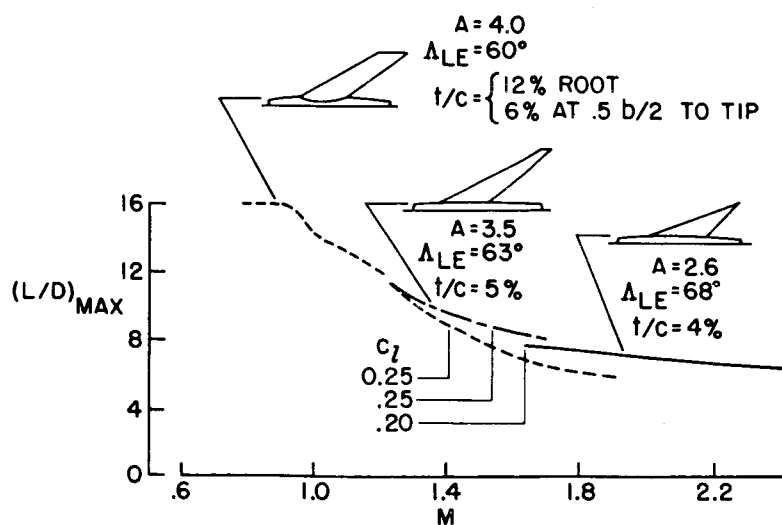


Figure 12.- High  $\left(\frac{L}{D}\right)_{\text{MAX}}$  configurations.

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PROJECT MERCURY

THE PROGRAM AND ITS OBJECTIVES

By Charles J. Donlan ✓ and Jack C. Heberlig ✓

NASA Space Task Group  
Langley Field, Va.

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# PROJECT MERCURY

## THE PROGRAM AND ITS OBJECTIVES

By Charles J. Donlan\* and Jack C. Heberlig\*\*

NASA Space Task Group

### SUMMARY

Project Mercury is reviewed in the light of experience gained thus far in the technical implementation of this Nation's initial program for manned orbital flight. Initial guidelines formulated for the Mercury program are reviewed together with some of the technical considerations that have influenced the design of the capsule and the operational plan for the booster-capsule vehicle.

The role of the astronaut in the Mercury program is discussed and some observations are made concerning the impact of the Mercury program on future manned space ventures.

### PROGRAM CONCEPTS

Project Mercury received official status on October 5, 1958, when it became an active NASA program, although a good deal of the preliminary planning in research activity extended back 12 to 18 months preceding this date. At that time, certain guidelines were adopted by NASA for the implementation of this project. These original guidelines are listed in figure 1 and they still constitute the basic program concepts for Project Mercury.

### Objectives

The main objectives of Project Mercury are (1) to insure safe orbital flight and recovery of a man in the space capsule and (2) to study man's capabilities in a space environment. In the initial manned orbital flight of Project Mercury, the first objective will undoubtedly receive much popular emphasis. However, the more far-reaching and scientific objective of the Mercury program is to study man in a space environment. This study may well influence the next generation of spacecraft. This objective will be discussed more fully subsequently.

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\*Associate Director of PROJECT MERCURY

\*\*Technical Assistant, Office of the Project Director

## Basic Principles

In regard to the basic principles, the approach taken was to use the simplest and most reliable methods known and to minimize new developments. Also, systems reliability was to be demonstrated by a progressive buildup of tests. In this way, the primary objectives could be attained in the most reliable manner in the least amount of time.

## Method

The method used to carry out the program is also outlined in figure 1. A high-drag ballistic reentry shape was selected for the basic design. In the present state of technology, this decision still appears to be a sound one. By modifying the adapter section of the Atlas ICBM booster to receive the capsule, this booster can be used to put the capsule in orbit. A cluster of solid-propellant retrorockets were adapted for positive reentry. After the high reentry velocity is reduced because of aerodynamic drag on the blunt shape, a parachute recovery system will lower the capsule to earth. In order to provide the astronaut with a positive means of escape from a malfunctioning rocket, an escape system was provided. Recent tests of this system simulating an off-the-pad abort, an abort at maximum dynamic pressure on exit, and an abort at a very high altitude and velocity have all been successful. The parachute recovery system is also being extensively tested to assure high reliability. Nevertheless, the astronaut is provided with a backup reserve parachute in case malfunction of the primary parachute occurs.

## CAPSULE AND ESCAPE SYSTEM CONFIGURATION

Figure 2 shows the capsule with and without its escape system. The retrorocket package and heat shield can be seen on the bottom of the capsule. The heat shield is of the ablative type and is made of a phenolic resin. Three retrorockets are used to initiate the descent from orbit. The metal shingles on the afterbody are made of cobalt alloy which dissipates the reentry heat by radiation. This type of construction allows full expansion in all directions. The parachute recovery system is contained in the upper cylinder. The maximum diameter of the capsule is about 74 inches. The launch configuration shown on the left of figure 2 is approximately 25 feet in height. The tower escape system allows the capsule to be pulled away from the immediate vicinity of the booster at any time. To date, there have been no booster malfunctions that would have prevented a safe escape of the capsule from the scene of immediate danger.

Figure 3 has been prepared to show the relative position of the astronaut, the window, and the emergency egress hatch. The astronaut is shown in a semisupine position with his back toward the heat shield. The window and egress hatch are two modifications that have been made to the capsule since its initial design. The window, directly above the astronaut's head, has been added to allow him to make observations independent of the periscope. The hatch, on the astronaut's right, has been developed so the astronaut could, in case of some unforeseen emergency after landing, exit from the capsule quickly. As the astronaut turns the hatch handle, primacord is exploded in such a manner as to cause the hatch structure to fail and thus fall free.

When all the many systems and subsystems are consolidated and integrated within the capsule structure, the Mercury capsule interior arrangement would be similar to the sketch presented in figure 4. The capsule is complicated by the redundant systems which have been added for increased safety. Starting from the small end of this sketch (from right to left), one can distinguish the antenna can, the two horizon scanners, the parachute package, the pitch and yaw jets and associated plumbing, the periscope housing, the astronaut's instrument panel, the side-arm controller, and sundry electronic and power packages. The environmental-control system is mainly below the astronaut's couch. The capsule is completely automatic, but provisions have also been made for piloted operation. Each system has a backup, and in the area of communications, there are several backups. There are two telemetry systems for sending back capsule data and pilot data. Also, there are onboard recorders which collect the same information sent by telemetry.

The astronaut may take over completely and perform all the functions of the automatic control system. Because of the parallel manual system, he will also be able to fly any mode that might fail on the automatic system.

## MAJOR CAPSULE SYSTEMS

All the major Mercury capsule systems have been described in reference 1. Some facts on a few of the systems of special interest are discussed briefly in the following sections.

### Instrument Panel

The instrument panel is shown in figure 5. The controls and displays on this panel are grouped as to function. The group on the left side has various pilot controls such as those concerned with the attitude control system and the retrorockets. The two large handles are for decompression



and repressurization. Decompression would be the method used for extinguishing a fire.

The next group is a sequencing display. This consists of a series of lights arranged in sequence and designed so as to indicate whether the various functions occurred, or did not occur, at the proper time. A green light comes on when the function does indeed occur and a red light comes on when there is some failure in the automatic functioning. The handle just to the left of each light is the pilot's control to override and correct the failure of any particular function. The light at the very top of this group is the abort light which comes on in the event an abort is initiated. If the abort does not occur, the pilot can abort by using his left-hand grip. The switch immediately under the abort light is the pilot's "ready" switch. This switch is used prior to launch to light up a "ready" light on the test conductor's panel in the blockhouse.

The next series of dials are flight instrumentation. The dials indicate acceleration, rate of descent, altitude, and the fuel supply in the hydrogen peroxide tanks. The top center of the panel is the attitude display. This combination display shows rate and attitude in pitch, roll, and yaw. The rate display is in the center, surrounded by the attitude displays consisting of three dials. This particular display for attitude control was arrived at after a number of simulation flights were made with various types of displays. The astronaut control of attitude will be aided by the periscope, which is located in the center and below the instrument panel, and a window, which is located directly above the panel. The astronaut will be able to see the horizon in back of the capsule through this window in the normal orbital attitude of  $14^{\circ}$  as well as in the retrofiring attitude of  $34^{\circ}$ . The periscope provides the astronaut with a view of the earth beneath the capsule. He can control the attitude by location of the earth's image in the periscope.

The instrument in the left center is a dead-reckoning type of device which consists of a small model of the earth, clock-driven to rotate in time with the orbit. In the right center is the satellite clock, which, in addition to firing the retrorockets, indicates time of day, the lapsed time since launch, and time-to-go for retrorocket firing. The astronaut is provided with an additional time-to-go dial which he may set for any desired event and also a separate stop watch.

The environmental-control-system instrumentation is grouped in the upper right-hand corner of the instrument panel. This group indicates the following environmental-control-system instrumentation: cabin pressure, temperature, relative humidity, and oxygen partial pressure; primary and emergency oxygen supply pressure; and carbon-dioxide partial pressure downstream of the lithium hydroxide canister in the pressure-suit

control system. Controls for the cabin-pressure-system fans are provided on this panel. A warning-light panel is installed adjacent to the environmental-control-system instrumentation. This panel gives the astronaut visual warning of major systems failures. At the same time, an auditory warning system is actuated when failure occurs. The astronaut would turn off the auditory warning signal by actuating a switch located by the light and then take corrective action. Warning lights are provided for loss in cabin pressurization, depletion of primary oxygen supply, emergency-rate mode of operation, decrease in cabin oxygen partial pressure below 3 psi, increase of carbon-dioxide partial pressure to 3 percent in the pressure-suit circuit, and excessive cooling water to the suit and cabin heat exchangers. A telelight panel is located on the left console to give the astronaut indication of flight-event sequential operation. On this panel the launch oxygen supply and snorkel operation are presented. Manual backup controls are installed adjacent to the lights. A green light indicates that the event has occurred; a red light indicates the event has not occurred. Heat-exchanger water-flow controls and the emergency-rate valve control are located on the right console which is not shown in this figure.

The instrument panel and astronaut will be photographed in flight. In addition, the data from the system instrumentation will be telemetered and recorded by onboard recorders. Continuous physiological data on the astronaut's condition in flight will be made with electrocardiogram, body temperature, and respiration rate and depth measurements. These data will be recorded by onboard recorders and will be telemetered.

#### Environmental Control System

The Mercury environmental control system (fig. 6) maintains the environment for the astronaut at a temperature between 50° and 90° F. (See ref. 2.) The astronaut may manually select the temperature in this range that makes him most comfortable. During reentry, when the cabin air temperature may approach 180° F, the suit environmental-control systems will still maintain the air in the pressure suit within the desired temperature range. The system is capable of operating for about 32 hours. Body odor is removed by use of activated charcoal, and carbon dioxide is removed by the use of lithium hydroxide. Fresh oxygen enters the suit at the torso and exits at the helmet. As the oxygen leaves the helmet, it enters a debris trap where small particles are removed and then it moves through the odor absorber and carbon dioxide absorber. A filter is provided downstream of the carbon dioxide absorber in order to prevent lithium hydroxide dust from contaminating the suit oxygen. The temperature of the gas is lowered in the heat exchanger and water is removed in the water absorber. The resulting drop in pressure is counteracted by an increased supply of gas from the oxygen bottle. Ten cubic feet per minute is the flow rate through the suit at approximately

5 psi. Considering the fact that this system only weighs about 130 pounds it represents indeed a noteworthy engineering accomplishment.

#### Automatic Stabilization and Control System

One of the more complex systems in the capsule is the automatic stabilization and control system. Figure 7 shows the normal operation of this system. The Atlas and capsule configuration at lift-off is shown in the left-hand part of this figure. When the Atlas drops its booster engines, an event known as staging, the capsule removes the escape rocket by firing a special jettison rocket. The Atlas booster then follows a tilt program which yields an orbital altitude of approximately 105 nautical miles at the time of insertion. At booster cutoff, the capsule's posigrade rockets fire, separating the capsule from the booster as shown in position B in figure 7. The stabilization system maintains capsule rate damping for 5 seconds until all oscillations of the capsule have been stopped. The capsule then is oriented to the retrorocket firing attitude as shown in position C. The capsule is maintained in this attitude for 5 minutes. In this time period, ground tracking stations can determine if capsule orbital velocities and insertion angles are correct enough to allow the capsule to maintain itself in orbit. The capsule is then oriented into the normal orbital attitude as shown in position D. The capsule is maintained in a  $14\frac{1}{2}^{\circ}$  blunt-end-up position. This permits the astronaut, with the aid of the periscope, to see past the larger heat shield end. Also the retrorockets are in the right position for reentering immediately, in case of emergency. The automatic stabilization and control system next orients the capsule into the retrofiring attitude as shown in position F. After retrofiring, the capsule is oriented to the reentry attitude as shown in position G, and the retropackage is jettisoned. As the capsule enters the atmosphere the control system introduces a roll rate of 10 to 12 degrees per second and also maintains rate damping, which prevents large capsule oscillation from building up. At about 42,000 feet, the drogue parachute is deployed and its action gives stability to the capsule prior to deploying the main parachute at 10,000 feet. Upon impact into the water, a small balloon is released carrying an antenna aloft to aid in capsule recovery.

#### ACCELERATION AND IMPACT ATTENUATION

One of the major concerns in the design of the Mercury capsule is to protect the pilot from excessive accelerations during flight and in landing. The escape rocket, if fired in an off-the-pad abort or at high altitude where the drag forces are low, would place upon the astronaut

an acceleration level of 20g for a period of almost 1 second. Impact on the water would give acceleration levels as high as 40g for a few milliseconds. A contoured couch which evenly distributes a man's weight was developed and tested. Normal exit G profiles were run, and the subjects demonstrated ability to control and do tasks which are necessary for normal capsule functions. Tests in which accelerations reached values as high as 25g were also conducted and demonstrated the validity of this support system in case of the escape-rocket firing. However, a major problem still existed for the case of land impact or impact on the water with a high horizontal wind profile. It is believed that a satisfactory solution to this problem has been accomplished by the accordion-like air cushion design shown in figure 8. The air cushion consists of a 4-foot skirt made of rubberized Fiberglas that connects the heat shield to the rest of the capsule. After the main parachute is deployed, the heat shield is released from the capsule and the bag fills with air. Upon impact, the airtrap between the heat shield in the capsule is vented through holes in the skirt as well as through portions of the capsule which are not completely airtight. Thus the desired cushioning effect is provided.

#### MERCURY FLIGHT TEST VEHICLES

In the preceding sections the capsule and its features have been discussed. In this section the booster for the Mercury capsule and the steps taken to assure its reliability are discussed.

Figure 9 shows the three rocket flight test vehicles used in Project Mercury. The Little Joe System shown at the left is a cluster of solid-propellant rocket motors used in the research and development program. No manned flights are contemplated with this booster although successful flights with small primates aboard the capsule have been made. The Redstone Booster shown in the center of figure 9 will be used for training flights for the astronauts. Before the astronauts' training program begins, an animal will be sent aloft in one of the earlier flights to demonstrate the system. The Mercury-Atlas combination is shown at the right. In the Atlas program, animals will also be sent aloft before a man is put in orbit.

The reliability of each of these systems is of obvious importance in this program. Figure 10 outlines a quality assurance program being followed in the booster programs. Standard booster hardware is being used to take advantage of all the experience gained in conducting the weapons system program of which the basic boosters are a part. The reliability of each system will be based upon the nominal performance of each individual component and subsystem. Also, as these systems are assembled in the various industrial plants, each part is being marked

or tagged with a Project Mercury label. Although the assembly-line technique has been established for these boosters, 100-percent inspection of all parts will take place to assure the highest quality control possible. Upon assembly of a booster system, a comprehensive rollout inspection will be made. The history of all parts going into a particular booster will be carefully documented and reviewed. In this manner, the operational proficiency of all systems and their individual components are made known. After the capsule has been mated to the booster and the mechanical and electrical integration is completed, a flight safety review board will be held. This review board reports on the overall expected reliability of the two systems - that of the capsule and booster together. Only after the capsule-booster combination meets the requirements of the flight safety review board will approval for flight be given.

## MERCURY ORBIT, TRACKING, AND RECOVERY

### Orbit Selection Criteria

Selection of the firing azimuth is very important in the overall Mercury operation. Figure 11 presents the orbit selection criteria used. It was most advantageous to use the existing tracking stations established for other National programs. (See ref. 3.) This resulted in the minimum number of additional stations which had to be implemented. A total of 17 stations, including the control center at Cape Canaveral, will be active while the Mercury capsule is in orbit. Of these 17, two of the stations will be located onboard ship in the mid-Atlantic and Indian Oceans. At each tracking station, there will be telemetry read-out giving the personnel real-time indication of the pilot's condition and the capsule systems. Prior to each flight, aeromedical and capsule systems specialists will be sent to each of the monitoring stations facilities. After each flight, these specialists will aid in evaluating the pilot and capsule systems' performance. The down-range tracking facilities of the Atlantic Missile Range will be used for the normal reentry tracking. In this way, the capsule may be followed on radar from the time it begins its reentry until impact. Constant communication by voice will be possible during this time. In order to determine the exact altitude and velocity of the capsule, the tracking facilities in the continental United States will be used extensively for determining the correct time of retrofiring so that the capsule will land in the predicted impact area. In the United States, there are six such tracking installations. Another requirement for the orbit selection is that the capsule should pass over friendly territory and maintain itself in a temperate zone. It is of interest to review the philosophy used in establishing the ground-station requirements.

## Ground Stations

Figure 12 presents the ground-station selection criteria. Continuous tracking is required from Hawaii to Bermuda. During each orbit, the retrotiming device must be capable of being reset along with other ground command capabilities. Continuous contact during launch and through insertion is required to determine the orbit characteristics and pilot's well-being. Frequent voice and telemetry contact must be maintained throughout the orbit. There must exist sufficient tracking information to provide continuous impact prediction; that is, at any time during the flight, the recovery forces should know exactly where the capsule would impact if the retrorockets were fired at that time.

## Recovery

It is not feasible to provide world-wide recovery forces; therefore, certain areas have been designated as primary recovery areas (fig. 13). A long recovery area is provided (from Cape Canaveral toward the Bermuda Islands) to include all impact points which might occur due to an early abort. Between the Bermuda Islands and the Canary Islands are other primary recovery areas. These areas are so located that if orbital velocity or insertion angle is not correct, firing of the retrorockets would cause the capsule to impact in these designated areas. If the initial orbit is not correct and conditions were to prevent the capsule from making three orbits, a second impact area is established at the end of the second orbit, as shown. The impact area from a nominal flight, however, occurs at the end of the third orbit down the Atlantic Missile Range, designated here as area 3. As mentioned previously, this would mean continuous tracking of the capsule during entry from Hawaii until impact.

## THE ASTRONAUT AND HIS TASK

The flight of a Mercury capsule into orbit and return, fundamentally, is an extension of the flight envelope of aircraft. The majority of sensations and demands on the astronaut are similar to those imposed on the pilot of any highly maneuverable, high-performance aircraft. For this reason, the prime technical qualification for the Mercury astronaut was that he should be an experienced pilot, trained in objective flying; i.e., test flying. His presence in the Mercury capsule can be expected to increase the chances of a completely successful mission because he has the capability and ability to take control of the capsule in the event of a malfunction of many of the automatic systems.

The astronaut in Project Mercury will furnish other important milestones in man's conquest of space. Some examples of the kind of scientific information that this first venture of man-into-space makes possible are listed in figure 14.

The Mercury orbital missions will permit the study of the effects of prolonged weightlessness on the physiological reactions, the subjective psychological reactions, and the performance of the astronaut. Of particular interest will be the effects of weightlessness on the function of the respiratory and circulatory systems. Telemetered electrocardiogram and breathing rates will make it possible to determine not only the effects of prolonged weightlessness itself, but the effects of transition from high accelerations to 0g and back to the high accelerations associated with launch and reentry. Problems associated with eating and drinking in prolonged weightless flight can also be studied. Stress hormone studies, together with pre- and post-physical examinations, will allow assessment of the body's physiological reaction to the total envelope of flight stresses.

Subjective psychological phenomenon likely to be effected by the weightless environment can be studied through voice reports and capsule instrumentation. The ability of the astronaut to orient himself in space can be determined. The effects of the reduction in proprioceptive cues in time estimation can also be determined. The extent of the oculogyric and autokinesis and oculogyric illusions in a weightless environment should also be of interest.

Of utmost practical importance to future manned space flight efforts will be the quality of the astronaut's performance in space. Accuracy of manual attitude control should provide a good test of psychomotor performance capability. Monitoring of the capsule orbital systems should provide an indication of the vigilance and perceptual accuracy. Navigation will test reasoning and visual discrimination of earth terrain features and heavenly bodies outside the capsule.

In reference 4, the following statement is made in regard to the importance of Project Mercury. "It has become increasingly evident that the full utilization of space flight for scientific and exploration purposes will depend on man's ability to participate in the flight mission. While, in principle, it is possible to devise increasingly complex instruments and automatic systems, in practice, the availability in the vehicle of human intelligence and decision-making capacity will prove to be essential for successful accomplishment of many advanced space missions."

It is believed that Project Mercury will provide the baselines by which the future space vehicles will be built around the most valuable and useful machine - man himself.

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## **OBJECTIVES**

1. ORBITAL FLIGHT AND RECOVERY
2. MAN'S CAPABILITIES IN SPACE ENVIRONMENT

## **BASIC PRINCIPLES**

1. SIMPLEST AND MOST RELIABLE APPROACH
2. MINIMUM OF NEW DEVELOPMENTS
3. PROGRESSIVE BUILD-UP OF TESTS

## **METHOD**

1. DRAG VEHICLE
2. ICBM BOOSTER
3. RETRO ROCKET
4. PARACHUTE DESCENT
5. ESCAPE SYSTEM

Figure 1.- Project Mercury.

NASA

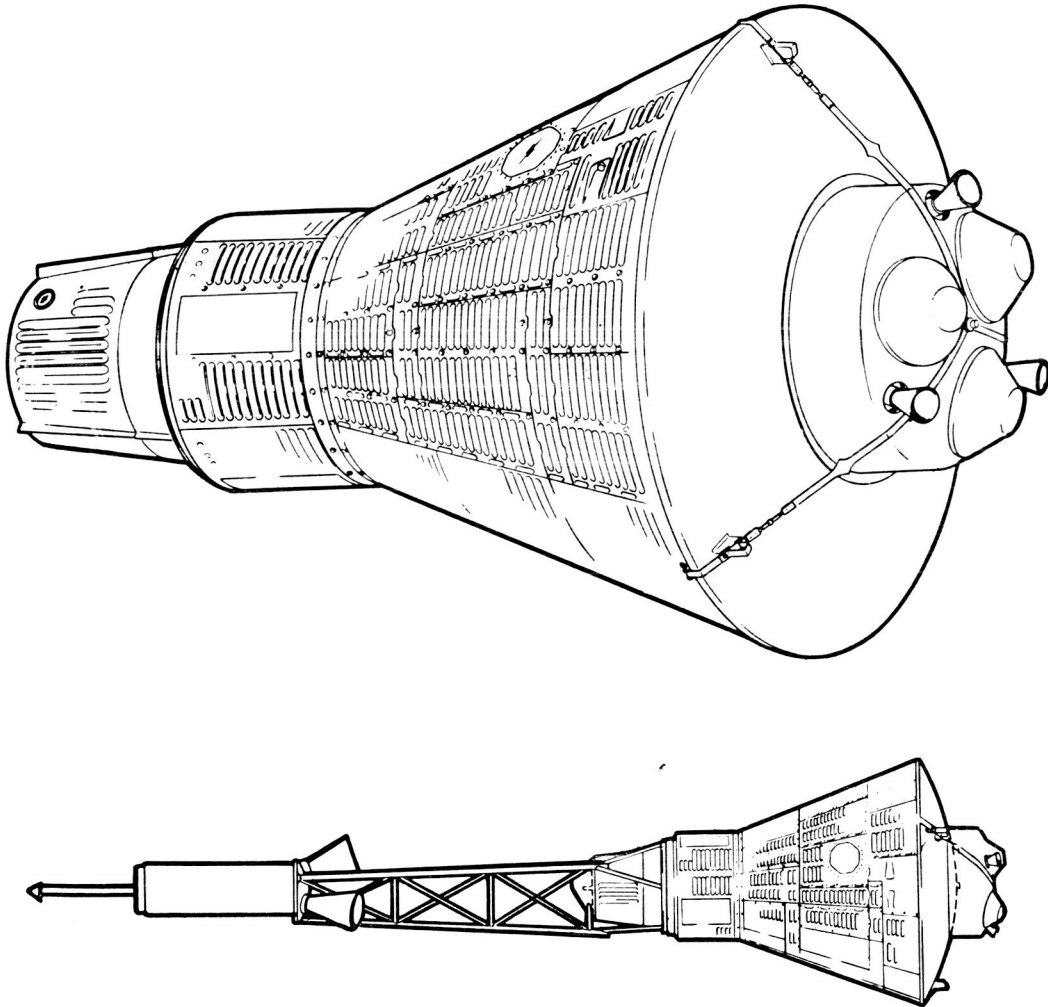


Figure 2.- Capsule and escape system configuration.

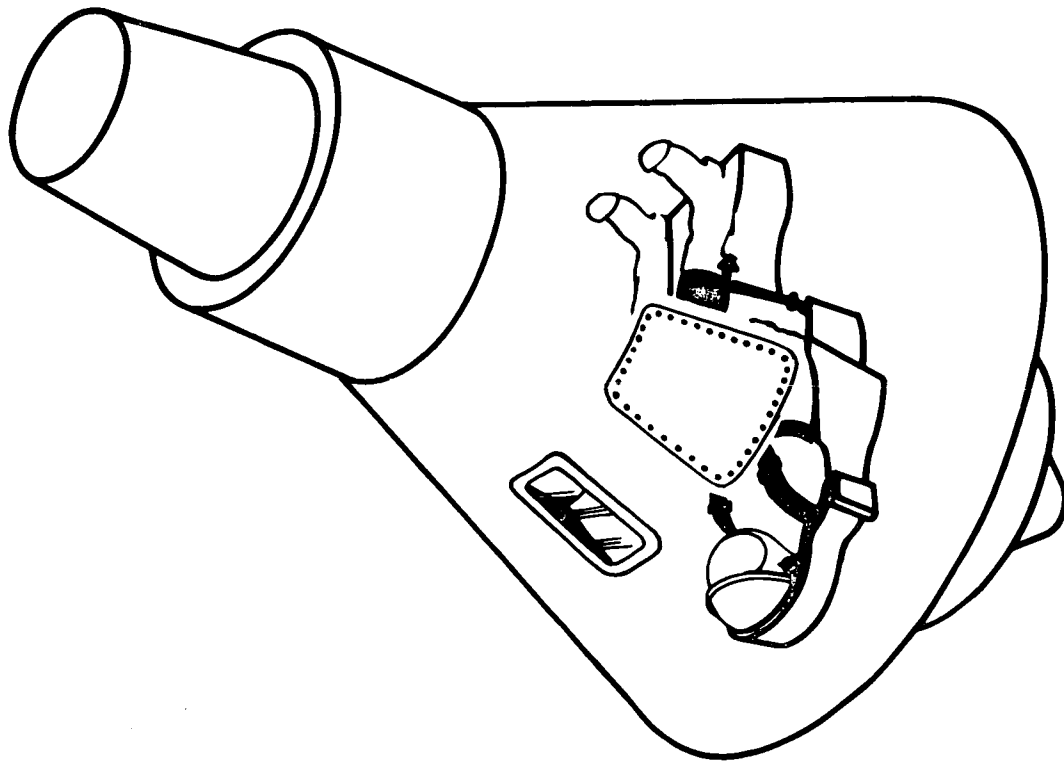


Figure 3.- View showing relationship of pilot, window, and emergency egress.

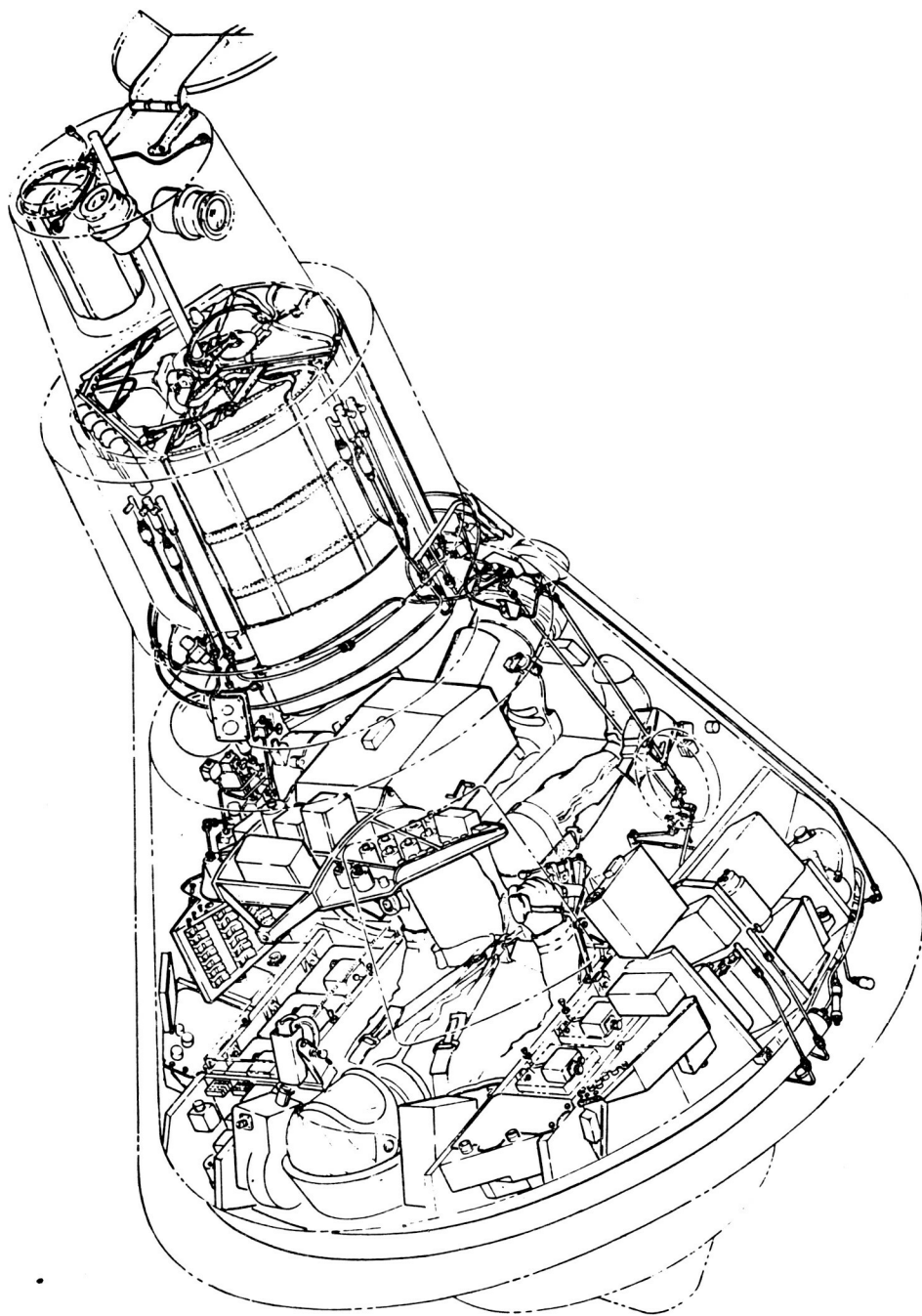


Figure 4.- Capsule internal arrangement.

NASA

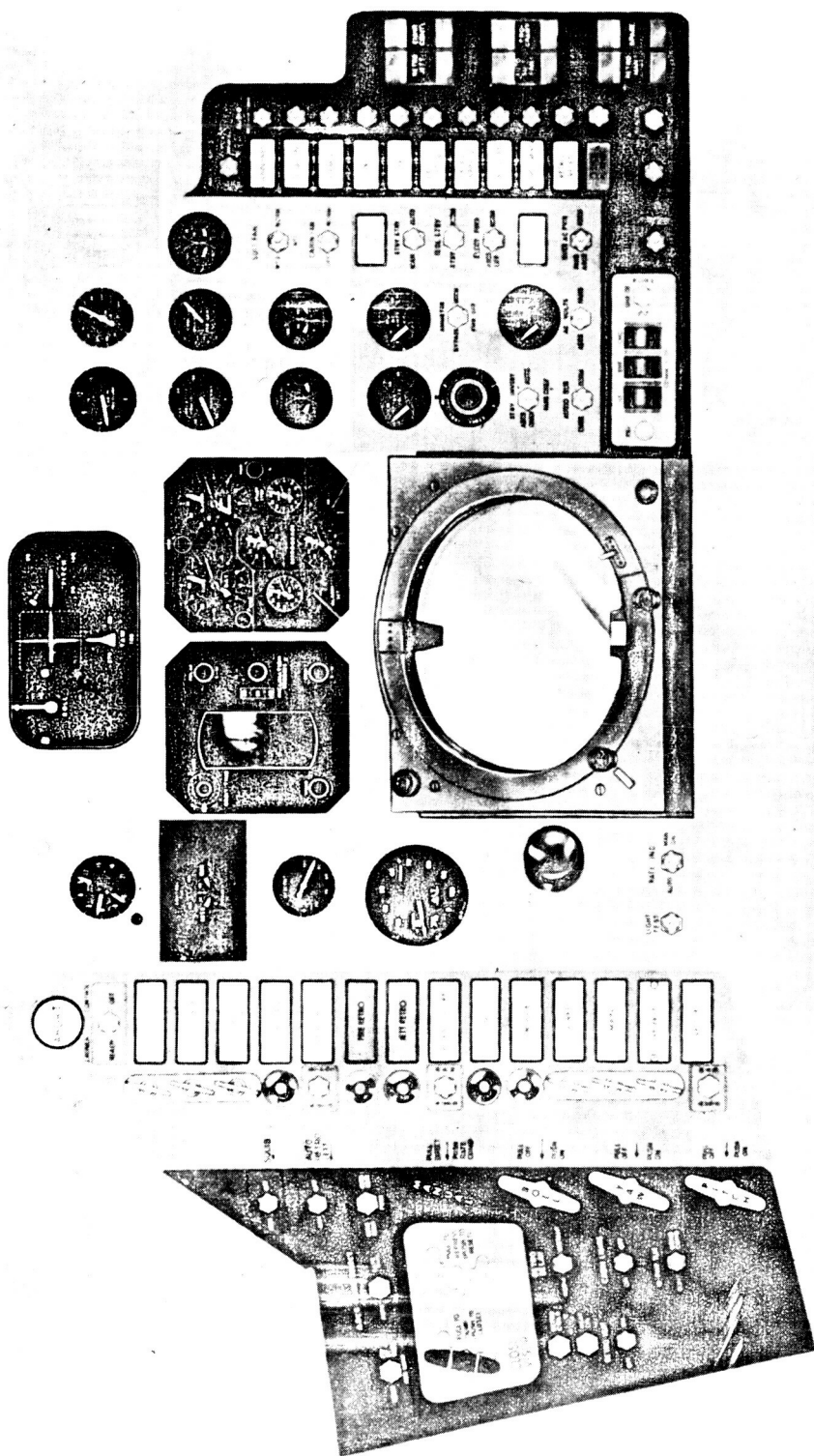


Figure 5.- Pilot's panel.

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NASA

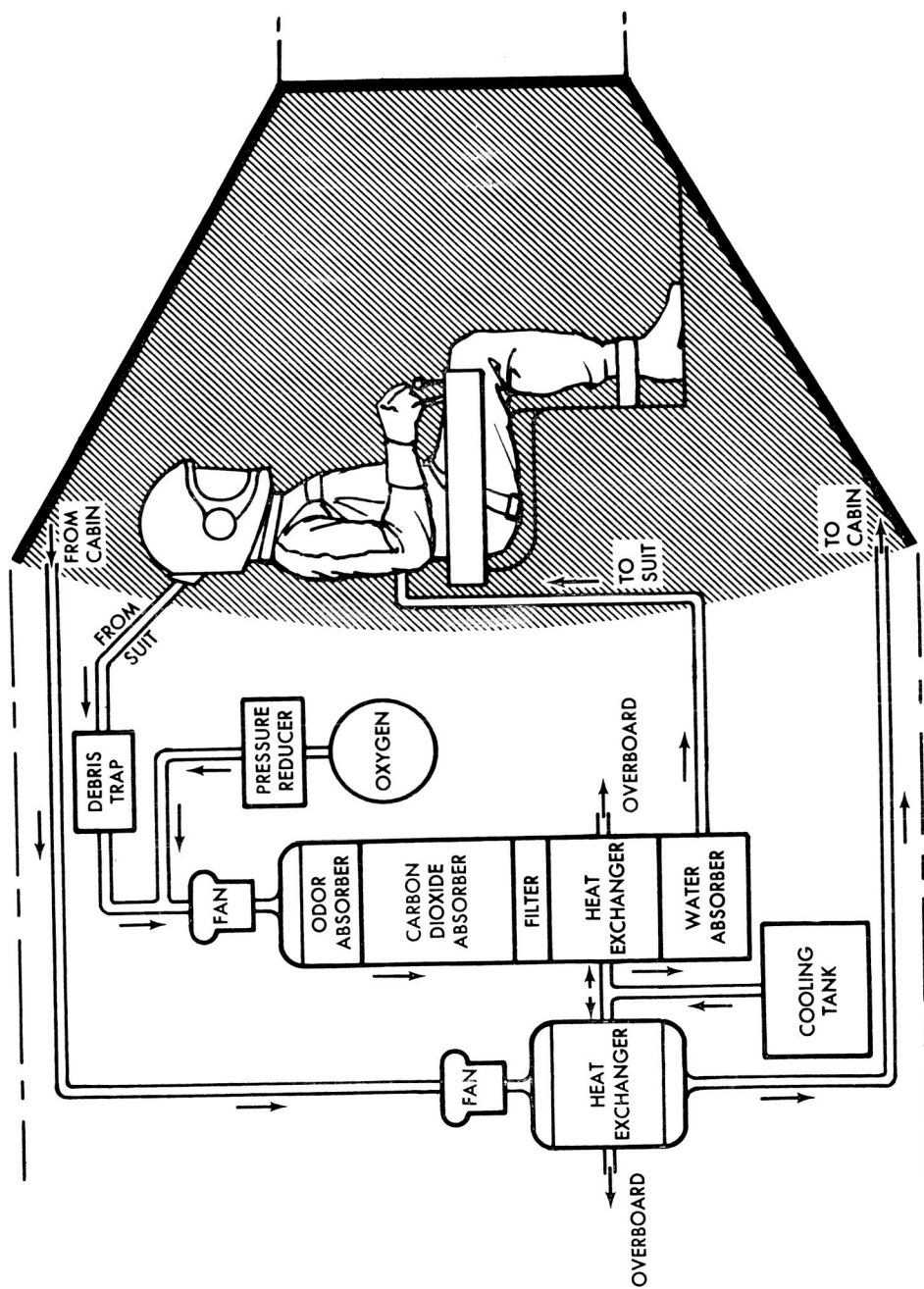


Figure 6.- The Mercury environmental control system.

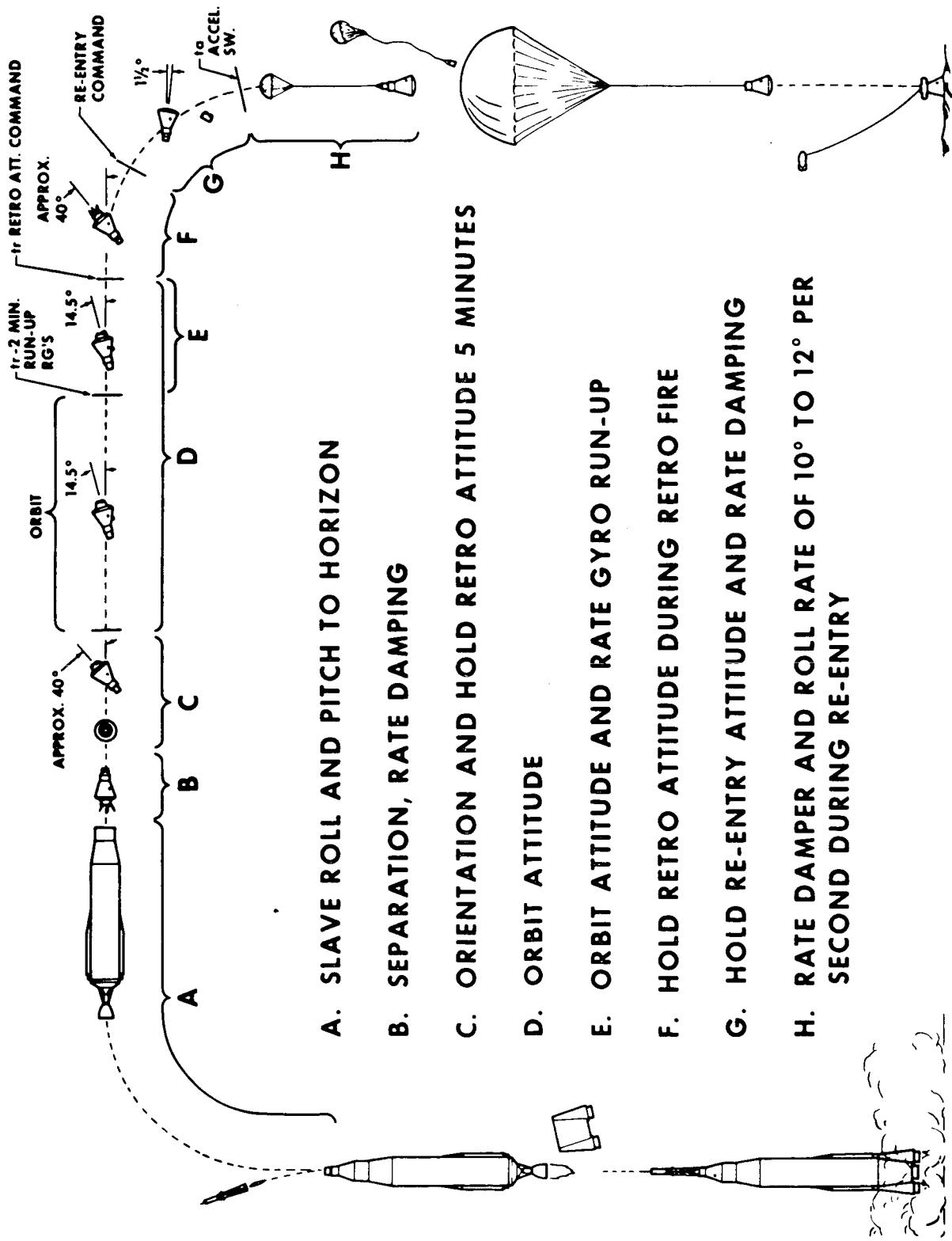


Figure 7.- Automatic stabilization control system normal operation.

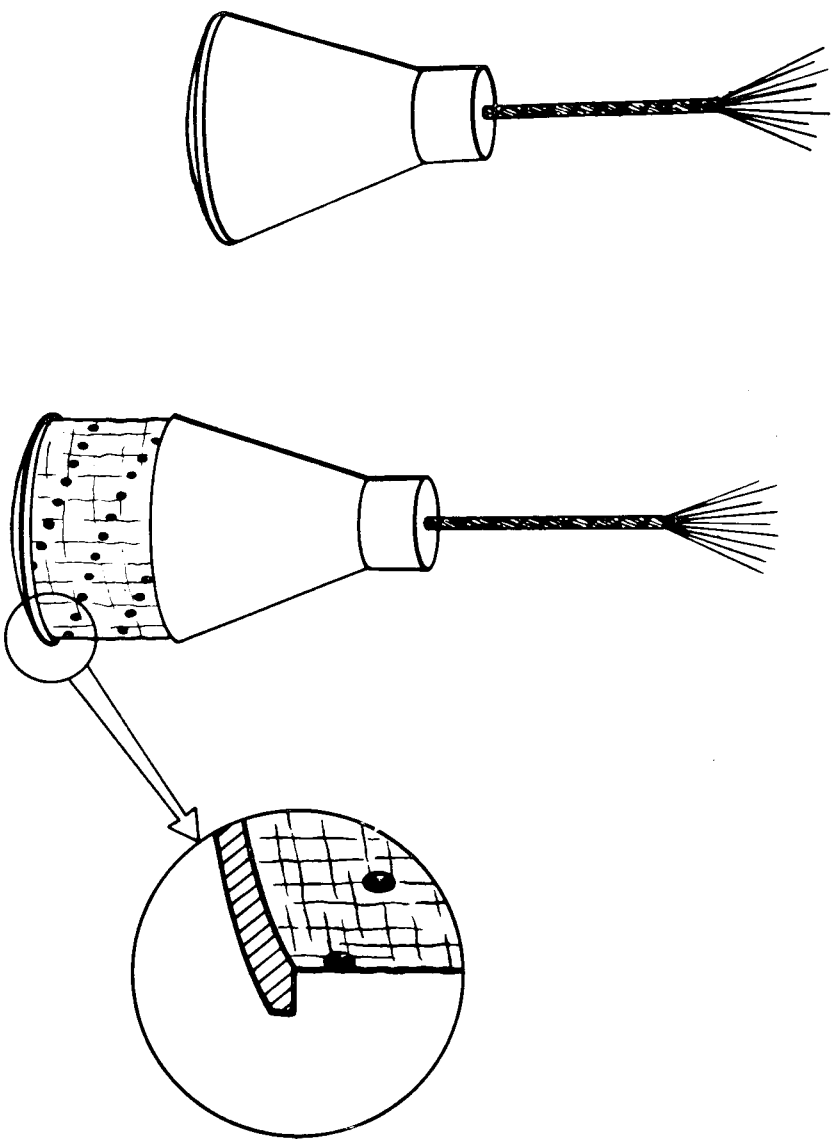
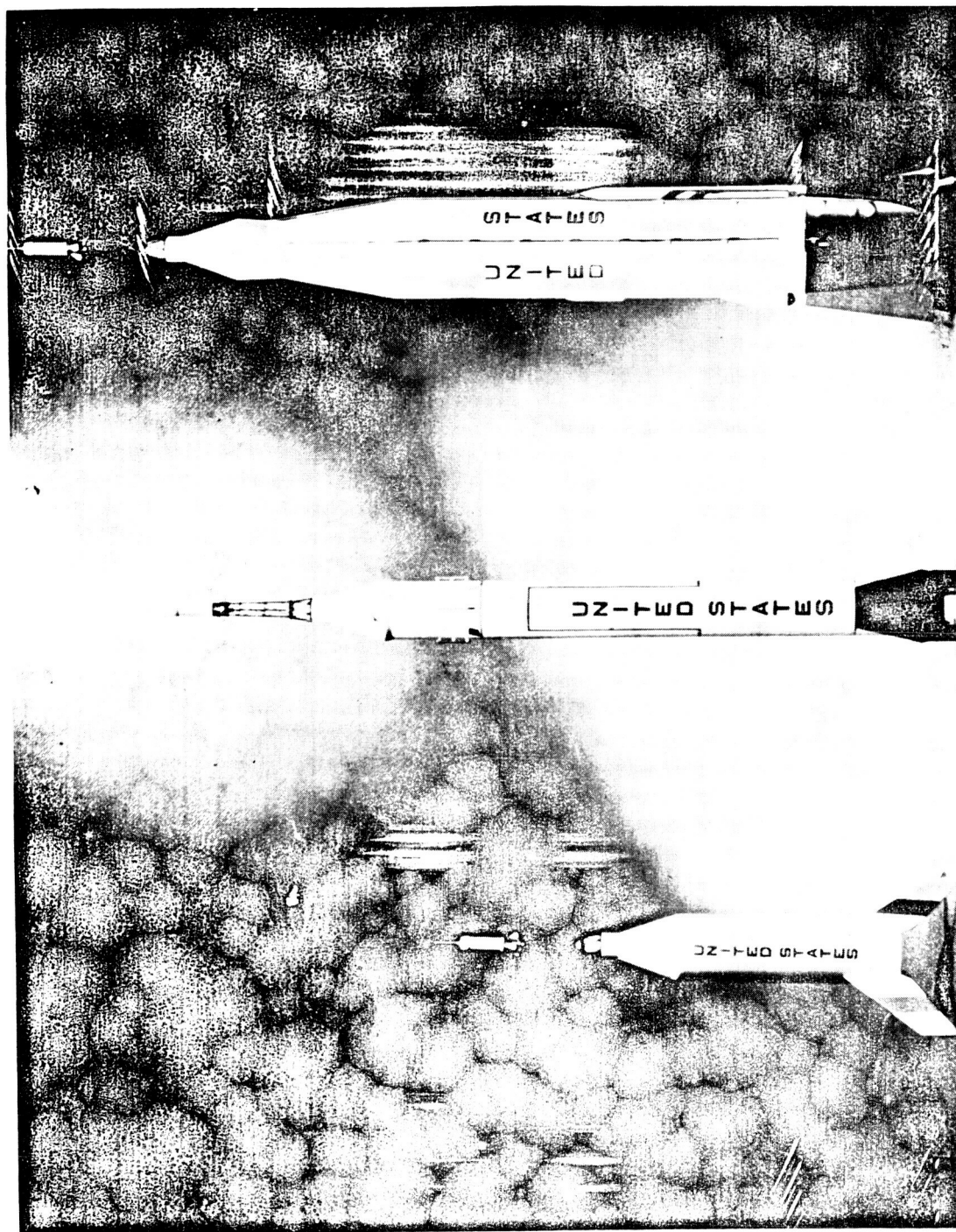


Figure 8.- Air cushion for impact attenuation.

NASA





# LITTLE JOE REDSTONE ATLAS

Figure 9.- Project Mercury flight-test vehicles.

NASA

## **STANDARD BOOSTER HARDWARE**

### **SELECTION OF INDIVIDUAL COMPONENTS AND SUBSYSTEMS**

**Nominal Performance**

**MARKING OF PARTS**

**100-PERCENT INSPECTION OF PARTS**

**COMPREHENSIVE ROLL-OUT INSPECTION**

**FLIGHT SAFETY REVIEW**

Figure 10.- Booster quality assurance.

NASA

1. USE OF EXISTING TRACKING STATIONS
2. USE OF AMR FOR NORMAL LANDING
3. REMAIN OVER CONTINENTAL U.S. FOR CONSIDERABLE  
TIME FOR TRACKING DURING ORBIT AND REENTRY
4. ORBIT OVER FRIENDLY TERRITORY AND IN TEMPERATE  
ZONE

Figure 11.- Project Mercury orbit selection criteria.

NASA

1. CONTINUOUS TRACKING FROM HAWAII TO BERMUDA
2. RETRO-TIMER RESET AND GROUND COMMAND  
CAPABILITIES DURING EACH ORBIT
3. CONTINUOUS CONTACT DURING LAUNCH AND THROUGH  
INSERTION
4. MAINTAIN FREQUENT VOICE AND TELEMETRY CONTACT
5. CONTINUOUS IMPACT PREDICTION

Figure 12.- Project Mercury ground station selection criteria.

NASA

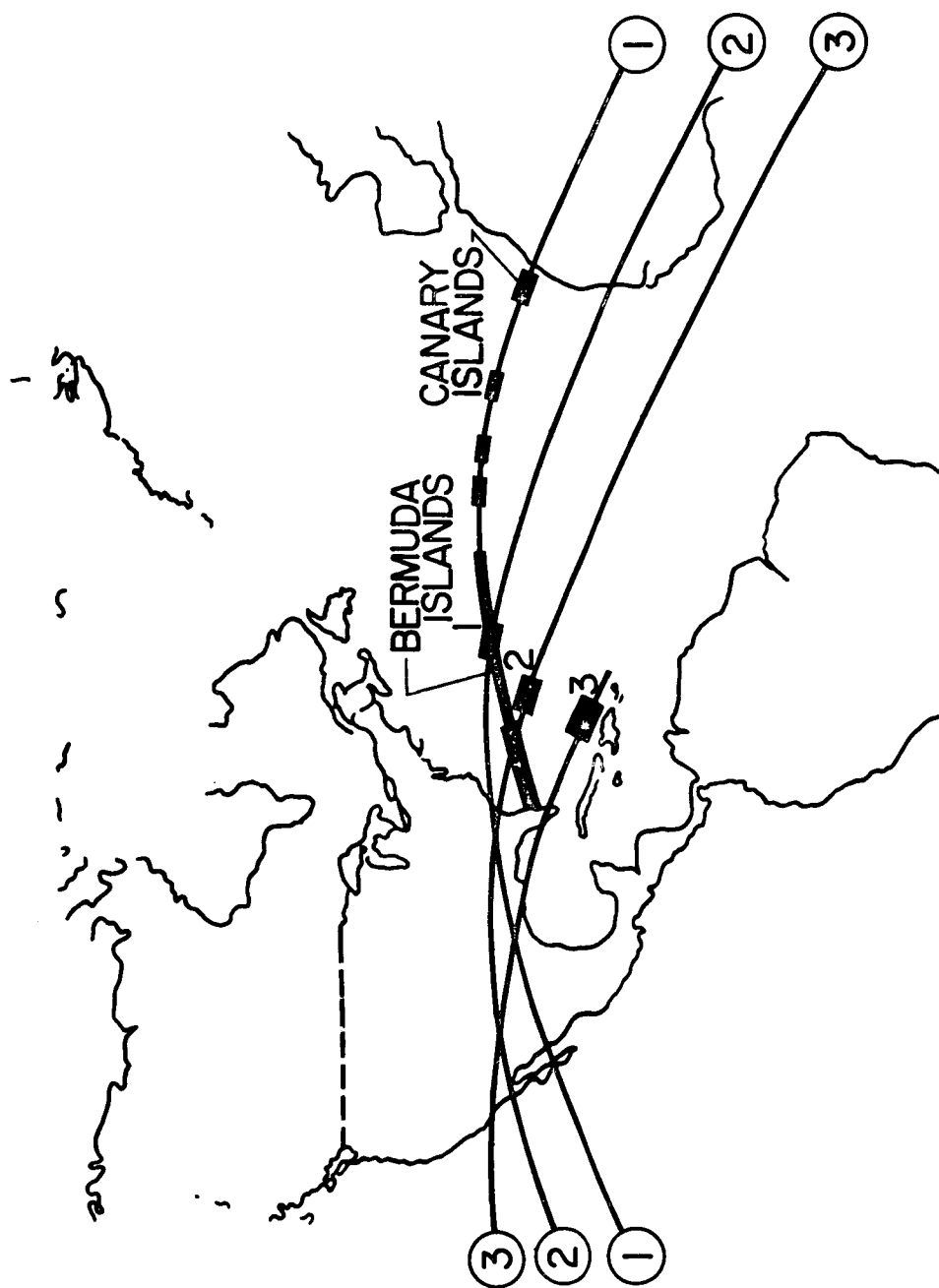


Figure 13.- Project Mercury anticipated recovery areas.

NASA

SOME EXAMPLES OF RESEARCH OBSERVATIONS  
OF THE BIOLOGICAL EFFECTS OF PROLONGED WEIGHTLESSNESS  
WHICH PROJECT MERCURY MAKES POSSIBLE

| <u>PHYSIOLOGICAL</u>                          | <u>PSYCHOLOGICAL</u>                 | <u>TASK PERFORMANCE</u>    |
|-----------------------------------------------|--------------------------------------|----------------------------|
| Function of the circulatory system            | Orientation                          | Attitude control           |
| Eating and drinking                           | Time estimation                      | Capsule systems monitoring |
| Pre and post flight tests of stress reactions | Autokinesis and oculogyric illusions | Navigation                 |

Figure 14.- Examples of Mercury aeromedical research. NASA

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GUIDELINES

FOR

ADVANCED MANNED SPACE VEHICLE PROGRAM

CLASSIFICATION CHANGED

*Unclassified Dec 9-29-71*

*by policy of Gordon E. Mercer*

CLASSIFIED DOCUMENT

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PREPARED BY

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

SPACE TASK GROUP

Langley Field, Va.

June 1960

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FOREWORD

This document is a compilation of the papers presented by NASA Space Task Group personnel at a series of meetings with personnel of NASA Headquarters and various NASA field installations during April and May 1960. The papers explain the current thoughts of the Space Task Group with regard to an advanced manned space vehicle program. This document is not an official NASA publication and is intended only for internal NASA use. It should not be referenced in publications prepared for release outside the NASA.

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INTRODUCTION TO  
ADVANCED MANNED SPACE VEHICLE PROGRAM  
As of April 1960

by Robert R. Gilruth

INTRODUCTION

The Space Task Group of the Goddard Space Flight Center has been assigned the task of implementing an advanced program of manned space flight. The tremendous amount of applied research required for such a program can only be done with the assistance of all of the NASA Centers. The following discussions were designed for use in enlisting the aid of the various centers in formulating the vehicle and mission design.

A very broad mission objective has been established; i.e., manned circumlunar flight and return to earth. It is implicit that the Saturn will be the primary propulsion system for this mission.

The following papers attempt to:

- (a) Define the objective so as to achieve as much capability in the vehicle as possible.
- (b) Draw on Mercury experience to provide broad guidelines for vehicle performance and safety.
- (c) Provide flexibility in the vehicle capability in the event that the manned lunar mission is proved to be subject to unacceptable risk in the target time period.
- (d) Indicate problem areas where work appears to be particularly needed.

Guidelines

The following papers outline and discuss guidelines for the mission. These guidelines are summarized in figures 1 and 2 and are presented more completely in Appendix I. Most of the guidelines are fairly obvious ones. A few of the guidelines may appear somewhat arbitrary and these require some introduction.

1. It is suggested that the vehicle be capable of lunar reconnaissance, i.e., it must pass close to the moon rather than simply make a broad lunar pass. This step is to insure that the problems of space navigation which ultimately must be solved for manned landings are attacked at an early date.

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2. It is shown that, as in Mercury, the escape or abort maneuver will strongly influence the vehicle design, particularly for onboard propulsion. However, having provided sufficient onboard propulsion for aborts the vehicle should be quite versatile for mission-oriented space maneuvers - either in lunar or earth orbits.

3. In the human-factors area the crew size and environment will be discussed. However, it is felt that the greatest unknowns lie in this area, particularly in the field of radiation. It is this lack of certainty about radiation which raises questions about the feasibility of the lunar mission in the target time period. This is an area that is most difficult to explore and which will require the utmost support from the Life Sciences organization.

4. In the control and guidance phase of the mission guidelines, a definite stand is taken for onboard command of the mission. One major finding of the Mercury project to date is that great simplification is possible through the use of onboard command. This will be proven more important in this more ambitious space flight where the duration is relatively great, where communications with earth may become tenuous, and where unusual and unpredictable situations may require direct assessment and action.

### Modular Concept

The various papers make frequent reference to "modules" or the "modular concept." This design concept has emerged from studies made by several groups and may have considerable merit in the present program. In this concept, it is suggested that a reentry vehicle together with its escape package might be Part I of the system. This vehicle would have attached to it a lunar module or caboose which would complete the system. For orbital qualification of the lunar system, these two parts would be flown in earth orbits. The reentry vehicle, however, would also be compatible with a much larger and heavier module for specific earth-orbital missions. The reentry vehicle with only the onboard propulsion would form a multiman space taxi with a considerable rendezvous capability.

### Weight

A brief discussion of weight allowances is in order - the use of the Saturn does not permit unlimited, or even generous, weight allowances for any part of the system. Figure 3 lists various components required for a lunar system along with early crude estimates of the weights that might be allowable for each. One point is clear - when the weight is parcelled out among the various components, the weight situation is fairly tight; since the total shown on figure 3 is 14,100 pounds and

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the Saturn capability for a lunar mission is presently estimated to be 15,000 pounds. Another point worth particular notice is the rather large requirement of 6,000 pounds for the auxiliary propulsion. Much work will be required in this area; and, in fact, highly ingenious systems will have to be used across the board in order to stay within weight limits.

Timing

Figure 4 presents an approximate timetable for the NASA manned space flight program. The primary point to be made at this time is that the research and development phase of the advanced vehicle program is beginning now and must be vigorously pursued in order to achieve our mission within the target time period.

Editor's note: Because of the preliminary nature of these discussions the timetable shown may not be current at any given time and some of the terms used are subject to redefinition as the program develops. The term "Lifting Mercury" in figure 4, for instance, refers to some type of continuation of the Mercury flight-test program prior to the beginning of actual flights of the advanced vehicle. This is discussed more fully on pages 49 and 50 (paper by Mr. Donlan).

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## GUIDELINES FOR ADVANCED MANNED SPACE VEHICLES

- I A CAPABLE OF MANNED LUNAR RECONNAISSANCE WITH LUNAR MISSION MODULE
- B CAPABLE OF COROLLARY EARTH-ORBITAL MISSIONS WITH LUNAR MISSION MODULE AND WITH SPACE LABORATORY
- C COMPATIBLE WITH SATURN BOOSTER  
(WEIGHT NOT TO EXCEED 15,000 LB FOR COMPLETE LUNAR VEHICLE AND 25,000 LB FOR EARTH-ORBITING VEHICLE)
- D CAPABLE OF 14-DAY FLIGHT TIME

Figure 1

## GUIDELINES FOR ADVANCED MANNED SPACE VEHICLES

- II A CAPABLE OF SAFE RECOVERY FROM ABORTS
- B CAPABLE OF GROUND AND WATER LANDING  
(ALSO CAPABLE OF AVOIDING LOCAL HAZARDS)
- C CAPABLE OF POINT (10 SQUARE-MILE) LANDING
- D DESIGNED FOR 72-HOUR POST-LANDING SURVIVAL PERIOD
- E AUXILIARY PROPULSION REQUIRED FOR MANEUVERING IN SPACE
- III A DESIGNED FOR "SHIRT SLEEVE" ENVIRONMENT
- B DESIGNED FOR THREE-MAN CREW
- C DESIGNED FOR RADIATION PROTECTION
- IV A PRIMARY COMMAND OF MISSION TO BE ON BOARD
- B COMMUNICATIONS AND TRACKING FACILITIES NEEDED

Figure 2

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## WEIGHT ESTIMATE CIRCUMLUNAR VEHICLE

|                                                |                        |       |        |
|------------------------------------------------|------------------------|-------|--------|
| COMMAND CENTER<br>AND<br>REENTRY CAPSULE       | CREW                   | 650   |        |
|                                                | STRUCTURE              | 1,800 |        |
|                                                | ECOLOGICAL             | 450   |        |
|                                                | POWER SUPPLY           | 600   |        |
|                                                | COMMUNICATIONS         | 300   |        |
|                                                | GUIDANCE               | 800   |        |
|                                                | RECOVERY               | 425   |        |
|                                                | TOUCHDOWN CONTROL      | 75    |        |
|                                                |                        |       | 5,100  |
| MISSION MODULE                                 | STRUCTURE              | 850   |        |
|                                                | ECOLOGICAL             | 575   |        |
|                                                | POWER SUPPLY           | 1,000 |        |
|                                                | COMMUNICATIONS         | 75    |        |
|                                                | SCIENTIFIC INSTRUMENTS | 500   |        |
|                                                |                        |       | 3,000  |
| AUXILIARY PROPULSION<br>(ESCAPE AND MIDCOURSE) |                        |       | 6,000  |
| TOTAL                                          |                        |       | 14,100 |

Figure 3

## MANNED SPACE FLIGHT PROGRAM

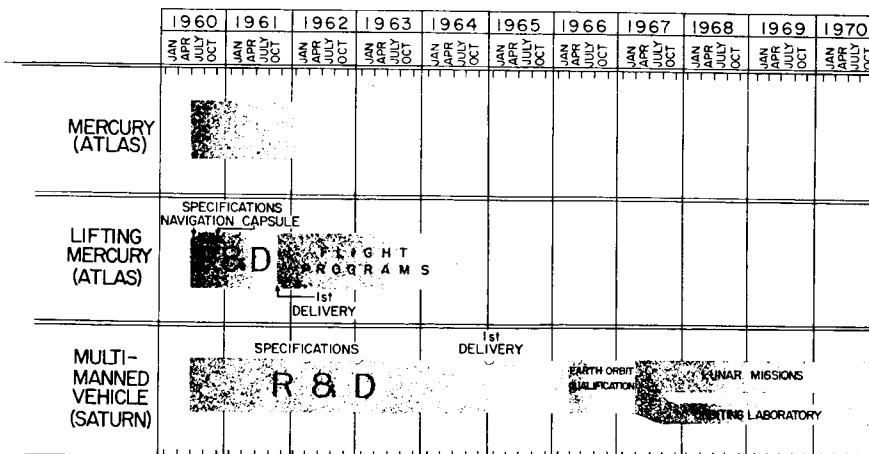


Figure 4

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MISSIONS, PROPULSION AND FLIGHT TIME

by Robert O. Piland✓

Introduction

The first four guidelines for the design of the advanced manned space vehicle deal with mission requirements, propulsion systems and design flight time capability. They are presented in figure 1.

1. The vehicle should be capable ultimately of manned lunar reconnaissance.
2. The lunar vehicle should be capable of earth-orbital missions for initial evaluation and training. The reentry component of this vehicle should be capable of earth-orbital missions in conjunction with space laboratories or space stations.
3. The multiman advanced space vehicle should be designed to be compatible with the Saturn and for the lunar mission it should weigh not more than 15,000 pounds including auxiliary propulsion and attaching structure.
4. The vehicle should be designed for a flight time capability, without resupply, of 14 days.

Lunar Mission

The first guideline states that the vehicle should be capable, ultimately, of manned, lunar reconnaissance. The justification for this step, in a technical sense, is that it is a logical intermediate step toward future goals of landing men on the moon and other planets. This mission will require solution of many of the problems associated with the manned, lunar-landing mission. This is particularly true of the earth reentry and recovery phases of the flight. In addition, it is a mission which will require a considerable amount of trajectory control, consequently imposing rather severe requirements on the navigation and control system, and thereby effectively demonstrating our ability to navigate in space. A further significant consideration in selection of the manned lunar reconnaissance is that it is the ultimate manned mission compatible with our firmly-planned booster program.

Numerous mission analyses involving trajectory studies will be required to determine the most desirable flight path for the vehicle. Figure 2 presents a typical lunar trajectory and return. It may be used to illustrate the initial approach to the mission that is being taken. This involves a flight path which would pass the moon at a distance on the order of several thousand miles. Such a flight path

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can give a condition which will allow the vehicle to return to earth without the necessity of firing large rockets. These trajectories which will give this so-called "free" or "safe" return will have to be identified by study, and their compatibility with launch insertion conditions and reentry conditions established.

To really accomplish lunar reconnaissance, prior to a manned landing, it is considered desirable to come much closer to the moon than several thousand miles. The order of 50 miles would appear to be a reasonable initial target for study purposes. To achieve this, we would propose to go into orbit about the moon. This procedure (as compared to a close pass-by) has the advantage of allowing the decision to be made in flight after establishing the integrity of the system, and also allows loitering in the neighborhood of the moon for a time period sufficient for reconnaissance requirements.

Many other detailed trajectory studies will naturally be required. Several examples might be various midcourse abort trajectories, and the effects of the nonuniformity of the moon's shape on the vehicle trajectory while in the neighborhood of the moon.

#### Earth-Orbital Mission

While the ultimate capability of the proposed vehicle is manned lunar reconnaissance, an intermediate or parallel mission requirement must be considered in the design of the vehicle. The second guideline, therefore, states that "The lunar vehicle should be capable of earth-orbital mission and that the reentry component of this vehicle should be capable of earth-orbital missions in conjunction with space laboratories or space stations."

The justification for this requirement stems from two sources:

1. Before lunar missions are attempted, earth-orbital flights will be required for lunar vehicle and crew evaluation, crew training and the development of operational techniques.
2. Concurrently, there is a need for a vehicle to be used in earth orbits with space stations for numerous research and development programs related to space flight.

Consequently, the reentry vehicle should be designed for use with both the lunar and orbiting space laboratories to avoid unnecessary duplication of vehicle development.

Figure 3 conceptually illustrates some of the possible combinations of components for use in the lunar and earth-orbital flights. It is suggested that the lunar vehicle will consist of a reentry component



can give a condition which will allow the vehicle to return to earth without the necessity of firing large rockets. These trajectories which will give this so-called "free" or "safe" return will have to be identified by study, and their compatibility with launch insertion conditions and reentry conditions established.

To really accomplish lunar reconnaissance, prior to a manned landing, it is considered desirable to come much closer to the moon than several thousand miles. The order of 50 miles would appear to be a reasonable initial target for study purposes. To achieve this, we would propose to go into orbit about the moon. This procedure (as compared to a close pass-by) has the advantage of allowing the decision to be made in flight after establishing the integrity of the system, and also allows loitering in the neighborhood of the moon for a time period sufficient for reconnaissance requirements.

Many other detailed trajectory studies will naturally be required. Several examples might be various midcourse abort trajectories, and the effects of the nonuniformity of the moon's shape on the vehicle trajectory while in the neighborhood of the moon.

#### Earth-Orbital Mission

While the ultimate capability of the proposed vehicle is manned lunar reconnaissance, an intermediate or parallel mission requirement must be considered in the design of the vehicle. The second guideline, therefore, states that "The lunar vehicle should be capable of earth-orbital mission and that the reentry component of this vehicle should be capable of earth-orbital missions in conjunction with space laboratories or space stations."

The justification for this requirement stems from two sources:

1. Before lunar missions are attempted, earth-orbital flights will be required for lunar vehicle and crew evaluation, crew training and the development of operational techniques.
2. Concurrently, there is a need for a vehicle to be used in earth orbits with space stations for numerous research and development programs related to space flight.

Consequently, the reentry vehicle should be designed for use with both the lunar and orbiting space laboratories to avoid unnecessary duplication of vehicle development.

Figure 3 conceptually illustrates some of the possible combinations of components for use in the lunar and earth-orbital flights. It is suggested that the lunar vehicle will consist of a reentry component

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and a minimum "mission component" to stay within payload limitations, as shown on the figure. The "mission component" or "caboose" allows jettisoning of this portion of the vehicle prior to reentry, thereby not requiring protection from reentry heating. In addition, this arrangement appears to have good usable space characteristics and two compartments appear possibly desirable for emergency conditions. However, complete studies have not been carried to the point where it can definitely be said that the two-component vehicle is superior from all standpoints as compared to using a single, larger vehicle, such as illustrated on the figure. Consequently, a detailed study needs to be made to determine the trade-offs between using a one- or two-component lunar vehicle. Such a study should consider the optimum distribution of systems and supplies between the two components. On the right of figure 3 the reentry component is illustrated, which may be either of the two sizes shown on the left, with an earth-orbital laboratory or space station. The large size of the space laboratory is allowable because of the increased earth-orbital payload capability as compared to lunar payload capability. It is suggested that the first of these orbiting space laboratories might be made compatible with the early version of the Saturn and, therefore, should weigh 25,000 pounds or less including the reentry component.

### Propulsion

The third guideline relates to the propulsion system and states that "The multiman advanced space vehicle should be designed to be compatible with the Saturn and for the lunar mission it should weigh not more than 15,000 pounds including auxiliary propulsion and attaching structure." The justification for this guideline is fairly straightforward in that the Saturn is the only firmly-scheduled booster capable of the manned lunar mission.

Although not required for our guideline presentation, we have prepared, for background information, several figures which give some approximate Saturn characteristics.

The Saturn booster is being developed in several versions. The two versions with which we are most concerned are referred to as C-1 and C-2. Figure 4 presents sketches and characteristics of the booster. The booster overall lengths, without payloads, are 150 feet and 180 feet for the C-1 and C-2, respectively. It should be remembered that these boosters are in a development or preliminary design phase and all numbers given here are subject to modification as studies and development continue.

Saturn C-1. - The first stage of C-1 is being designed and developed at the Marshall Space Flight Center in Huntsville, Alabama. The first booster is now on the test stand and testing is underway. This stage is

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made up of a central tank and eight smaller tanks located around the central tank. Liquid oxygen is carried in the central tank and in four of the outer tanks. RP-1 is carried in the remaining four tanks. Control of the first stage is obtained by gimbaling the four outer engines. The four central engines are fixed.

The second stage of C-1 will be developed by Douglas Aircraft Company. The stage uses liquid oxygen and hydrogen, as do all the upper stages of the Saturn vehicles. All four engines, which will be uprated Centaur engines, are gimballed for control.

The third stage is essentially the Centaur stage being developed by Convair/Astronautics.

Two-stage C-1 vehicles may be available for research and development testing as early as December 1962. Vehicles qualified for manned use are estimated to be available in 1966. Ten flights are planned in this qualification program.

The C-1 version does not have sufficient payload capability for the manned circumlunar mission; however, it does have the capability of placing up to 25,000 pounds of payload in earth orbits leading up to lunar missions.

Saturn C-2.- The first stage of C-2 is similar to the first stage of C-1, except that its tankage may be reduced consistent with optimum propellant loading.

The second stage of C-2 is not used in C-1 but is a new stage. It will use four 200,000-pound thrust hydrogen-oxygen engines. The contract for this engine is now being let. The stage contract has not been let; therefore, this is the least advanced of any of the Saturn stages.

The third stage of C-2 is the same as that used as the second stage of C-1.

C-2 vehicles will be available for research and development flight tests in 1965 and for manned tests in 1967. Four flights of this vehicle are planned for C-2 qualification, in addition to the ten planned for C-1.

#### Launch Trajectory Parameters

Figures 5 and 6 present some launch trajectory parameters for the C-1 and C-2 vehicles. These trajectories have not been optimized for the missions considered and are presented here to indicate relative performance of stages, and the loadings to be expected in the way of dynamic pressure and axial accelerations.

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Figure 5 presents the variation of dynamic pressure, acceleration and velocity as a function of time for a launch trajectory into earth orbit using the C-1 vehicle. The maximum dynamic pressure is seen to be 775 pounds per square foot and occurs about midway through first-stage burning. For reference, the maximum dynamic pressure on the Mercury-Atlas flights is about 1,000 pounds per square foot. At first-stage burnout, at 210,000 feet altitude, the dynamic pressure has dropped to near zero.

The maximum axial acceleration of 6.5g occurs at first-stage burnout, and is several g less than experienced in the Mercury-Atlas flights.

The velocity contributions of the three stages are 9,000, 8,000, and 8,500 feet per second, respectively, to give the orbital velocity of 25,500 feet per second.

Figure 6 presents the variation of dynamic pressure, acceleration, and velocity as a function of time for an escape launch trajectory for the C-2 vehicle. Maximum dynamic pressure is seen to be 600 pounds per square foot as compared to 775 pounds for C-1. The dynamic pressure at separation is on the order of 400 pounds per square foot. The implications of separation dynamic pressures of this magnitude on the design of automatic abort-sensing systems, and on vehicle loading, in case of abort, will require careful consideration.

Maximum longitudinal acceleration is about 5g and occurs at second-stage burnout.

Velocity contributions of the three stages to escape velocity are 3,000, 15,000, and 18,000 feet per second, respectively, for the three stages.

#### Flight Time Capability

The fourth guideline states that "The vehicle should be designed for a flight time capability, without resupply, of 14 days." In justification, it is to be noted that minimum time for a lunar circumnavigation mission is about 6 days. Taking into account that the total flight time may be affected by lunar perturbations, loiter times, or use of nonminimum time trajectories, considering the need for safety factors, and the desire for mission flexibility, it is believed that the vehicle should be designed for a minimum flight time of 14 days.

A particularly important study is required in regard to this capability in connection with the selection of a suitable type of power supply. It is roughly estimated that 400,000 watt-hours are required for the mission. Considerable study of storage batteries, fuel cells, auxiliary power units, and solar batteries has been made by a number of

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people. The merits of these various systems or combinations must now be considered in the light of the particular requirements of this mission, with particular emphasis being placed on the reliability required for manned flight. Some items that should be considered in such a study include:

1. What percentage of the power units is desirable to place in the "caboose?"
2. Would the use of storage batteries both for power and as radiation shielding make this type of unit preferable?
3. Should redundancy for reliability be obtained by using two different types of systems or two of the same system?

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## MISSION AND VEHICLE GUIDELINES

- I A. CAPABLE OF MANNED LUNAR RECONNAISSANCE  
WITH LUNAR MISSION MODULE
- B. CAPABLE OF COROLLARY EARTH-ORBITAL MISSIONS WITH  
LUNAR MISSION MODULE AND WITH SPACE LABORATORY
- C. COMPATIBLE WITH SATURN BOOSTER -  
(WEIGHT NOT TO EXCEED 15,000LB FOR COMPLETE  
LUNAR VEHICLE AND 25,000 LB FOR EARTH-  
ORBITING VEHICLE)
- D. CAPABLE OF 14-DAY FLIGHT TIME

Figure 1

## TRAJECTORY PLAN FOR MANNED LUNAR SPACE VEHICLES

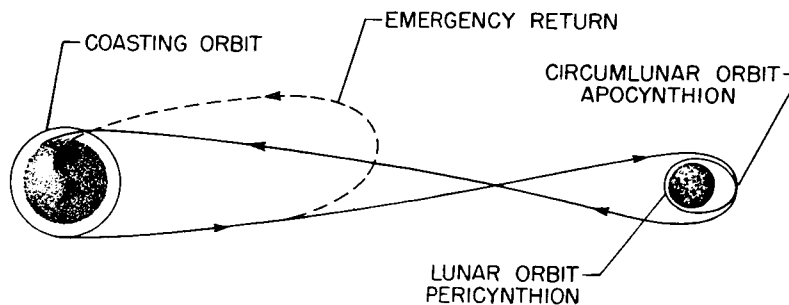


Figure 2

### CONCEPTUAL VEHICLES

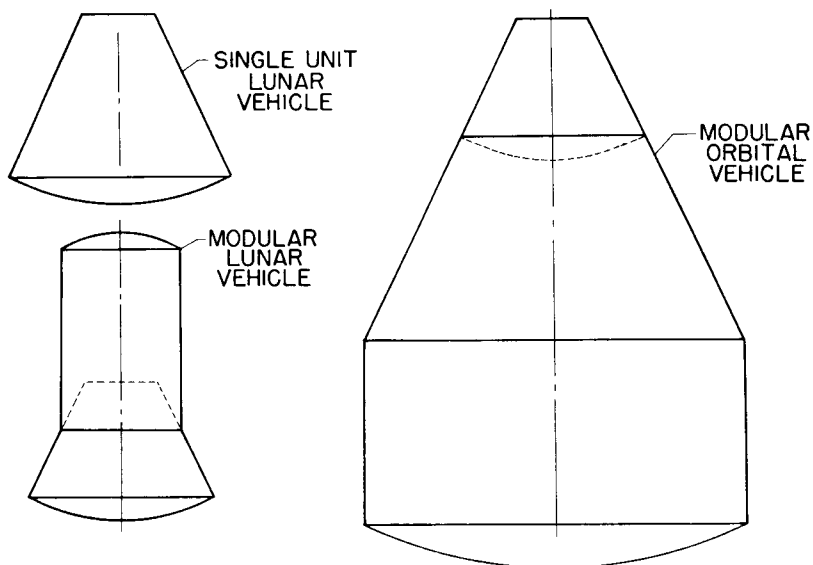


Figure 3

### SATURN VEHICLE CONFIGURATIONS

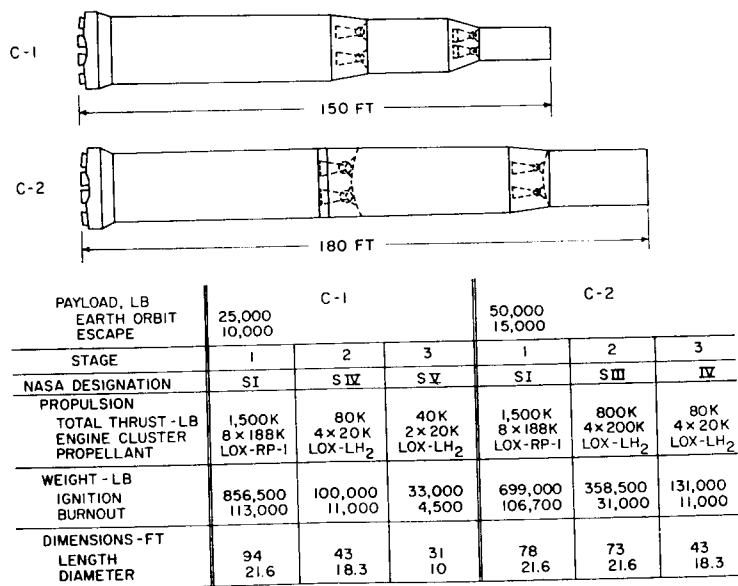


Figure 4

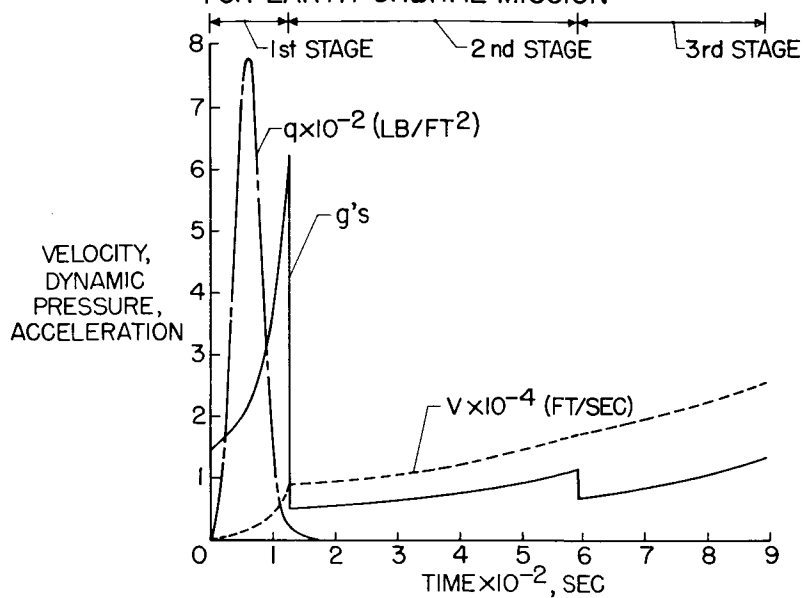
SATURN C-1, TYPICAL LAUNCH TRAJECTORY PARAMETERS  
FOR EARTH ORBITAL MISSION

Figure 5

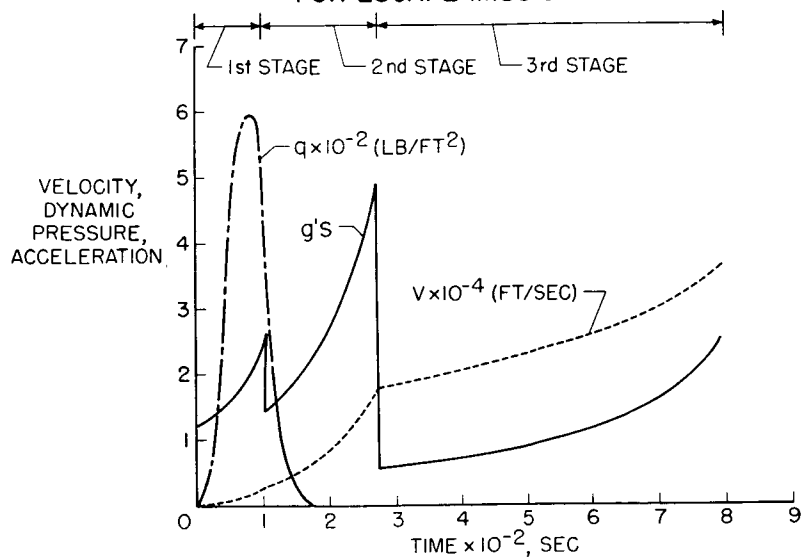
SATURN C-2, TYPICAL LAUNCH TRAJECTORY PARAMETERS  
FOR ESCAPE MISSION

Figure 6



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ABORT AND LANDING

By Maxime A. Faget ✓

Introduction

The following presentation is primarily concerned with the vehicle's return to the earth's surface at the termination of either a normal mission or an aborted mission. There are five guidelines in this area and they are shown in figure 1.

The first guideline requires that the vehicle should have the capability of safe crew recovery from aborted missions at any speed up to the maximum velocity, and that this capability be independent of the use of the launch propulsion system. It would follow that safe control of the vehicle subsequent to aborting the mission is thereby required. This control may be exerted by the auxiliary rocket propulsion (maneuvering rockets) or the aerodynamic L/D capabilities of the configuration.

The second guideline requires that the vehicle should be capable of a satisfactory landing on both land and water and should have the capability of avoiding local hazards in the landing area. This requirement was predicated upon the following considerations:

(a) Emergency conditions or navigation errors that force a landing on either land or water.

(b) Accessibility for recovery is an important consideration and the relative superiority of land-versus-water landing will depend on local conditions as well as other considerations. From our Mercury experiences, we have found that weather conditions which may exist over the earth's surface during a mission play an important part in determining the landing capability of the vehicle. Therefore, we feel that the vehicle's capability should include landing in a 30-knot wind, and that the vehicle should be water-tight and have adequate sea-keeping qualities under sea-state 4 conditions (10 - 12-foot waves).

The third guideline requires that the vehicle should have the normal capability of landing at one of several previously designated ground surface locations, each approximately 10 square miles in area. In order to meet this requirement, studies are needed to assess the value of impulse maneuvers, guidance quality, and aerodynamic L/D during return from the lunar mission. It is desired, of course, that a suitable trade-off between these items be arrived at to facilitate this performance. It might be mentioned that this requirement for point landing is far less severe for the orbital mission than for the lunar-return situation.

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In the fourth guideline, it is required that the vehicle be designed for crew survival for at least 72 hours after landing. Due to the random nature of possible emergency maneuvers, it will be impossible to provide sufficient recovery forces to cover all possible landing locations. The above requirement will permit mobilization of normally existing facilities and adequate time for a safe recovery. The vehicle, of course, should also be equipped with adequate location devices that will perform suitably in the various regions of the earth.

Finally, in the last guideline, it is required that adequate auxiliary propulsion should be provided for the guidance maneuvers required in the assigned missions and to effect a safe return in event of launch emergencies. The guidance accuracies and capabilities should be studied in order to determine the auxiliary propulsion requirements. Sufficient reserve propulsion should be included to accommodate corrections for maximum guidance errors. A single system may suffice for both the guidance maneuvers and the escape propulsion requirements.

The remainder of the presentation will provide additional discussion and material associated with the above guidelines.

#### Escape Regimes

First, let us consider the escape problem which will exist for this new mission, as shown in figure 2. This figure shows the various escape regimes that will exist for a launching to escape velocity. These regimes are presented as follows:

Launching period (including time on the pad prior to launch).- In this regime, we are primarily concerned with providing sufficient altitude after escape from the propulsion system in order to satisfactorily deploy the landing parachute or, in the case of a winged vehicle, the escape maneuver must provide sufficient velocity to attain flying speed.

Atmospheric flight.- This portion of the launch trajectory is within the atmosphere; here the problem of escape is associated with aerodynamic loads induced on the manned vehicle during and after the escape maneuver. It is important that the escape propulsion system has sufficient power to overcome the drag of the vehicle and pull it away from the booster system. It is also important that attention be paid to the stability characteristics of the vehicle during the escape maneuver.

Suborbital space flight.- This regime starts at the time when the vehicle leaves the atmosphere and extends up to the time when orbital velocity is attained. In this case, the concern is with the subsequent reentry following an escape maneuver. It is important that the launch

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trajectory and the escape maneuver be made in a manner that reentry deceleration loads will not exceed the tolerance of the crew.

We are rather familiar with the problems associated with these first three escape regimes as they all exist in the Mercury launch trajectory. It might be mentioned at this time that all of the critical heating and air loads associated with the design of the Mercury capsule were generated in analysis of the maneuvers resulting in escape, or subsequent to escape, during the reentry. Therefore, it is very important that close attention be paid to these escape regimes inasmuch as they most likely also provide the basis for all loads criteria for the advanced vehicle.

Superorbital space flight.-- The final escape regime extends from the time at which orbital velocity is reached up until the actual burn-out of the propulsion system where escape velocity is achieved. In this regime, it is necessary to provide an adequate auxiliary propulsion system which can return the capsule to the surface of the earth in a reasonably short time. In fact, it is desired that descent to the earth's surface commence immediately with the escape maneuver. This requirement results from the desire to avoid long dwell periods in either the lower or upper radiation belt which may otherwise result if a propulsion failure left the vehicle in a highly elliptical orbit.

Superorbital escape trajectories.-- The most reasonable way to dissipate the energy that exists in the space vehicle is to use atmospheric braking. This is most easily done by turning the flight path downward so that the vehicle intersects the earth's atmosphere. The performance requirements for this maneuver are illustrated in figure 3. On this figure, the velocity increment required for immediate reentry is plotted on the vertical scale, with the flight time in seconds plotted on the horizontal scale. There are three trajectory cases considered on this figure.

Case 1 is for an optimum launch trajectory as given to us by the Marshall Space Flight Center. In this case, burnout velocity is achieved at an angle of  $+4^{\circ}$  and the superorbital escape regime is entered at satellite velocity at a flight-path angle of zero degrees. It can be seen that a great deal of velocity change may be required in order to effect an immediate reentry as escape velocity is approached. In order to improve this situation, we have considered alternate trajectories in which the initial path angle has been reduced.

In Case 2, the initial path angle of  $-4^{\circ}$  changes to zero degrees in going between satellite and escape velocity.

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In Case 3, the initial path angle of  $-6^{\circ}$  at satellite velocity changes to  $-2^{\circ}$  at escape velocity. It can be seen that for these two cases the velocity increment required to effect an immediate reentry is greatly reduced. The amount of performance penalties associated with these alternate trajectories has not yet been determined.

It should also be pointed out that there may be another undesirable effect associated with these alternate trajectories. This has to do with the escape situation that would exist prior to the attaining of orbital velocity. With these alternate trajectories, the reentry angle following an abort at suborbital velocity will be much greater and, therefore, the reentry decelerations will, of course, be greater. How well the auxiliary propulsion system can cope with this situation is also yet to be determined. Thus, it is obvious that a great deal of study and analysis must be done in order to arrive at an optimum launch trajectory, taking into consideration the various abort maneuvers that may be required, as well as the performance of the launch propulsion.

#### Aerodynamic Considerations

Figure 4 outlines some of the aerodynamic considerations associated with the guidelines under discussion. As pointed out previously, the heating and loads criteria will undoubtedly be generated by consideration of the various abort and emergency maneuvers rather than the return from the moon in the normal mission. From a heating standpoint, it might be pointed out that the vehicle must be capable of reentries at escape velocity. Not only must convective heat-transfer analysis be extended to this new velocity, but also a great deal must be learned about the radiation heating associated with the bow-shock wave. There are indications that this heating may be on a level equal to that of the convective heating. It might be mentioned that the effectiveness of the ablation heat shield will be greatly reduced when coping with radiation heating as compared to the normal situation when convective heating is the criteria.

The needed aerodynamic capabilities of the vehicle will be determined by studying the trade-offs between aerodynamic L/D, guidance capabilities, and the usefulness of the maneuvering rockets in making final corrections just prior to reentry. Also of concern is the ability to maneuver at low speeds after the vehicle has come out of the blackout region during reentry when no radio communications are possible. In the postblackout period, it is expected that another fix will be available from ground radar, and corrections for drift of the inertial systems aboard the capsule will be made. These corrections may require further maneuvers in order to land in the designated area.

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The actual aerodynamic configuration will be compromised between these and other requirements spelled out throughout the guidelines. Needless to say, weight will be of primary importance in these considerations.

### Landing Considerations

Figure 5 outlines some of the landing considerations resulting from the guidelines presented at this time. Inasmuch as the landing may be made either by a parachute system or by a glider, the landing considerations will be taken up separately for these two systems. For the parachute system it appears that the continued development of reliable large parachutes must be pursued. For the Mercury capsule, we found that no suitably-developed parachute was obtainable and a development program had to be carried out in order to provide a design having adequate reliability. The advanced vehicle will be approximately three times the size of Mercury. It is desirable that some maneuvering capability be provided during parachute descent. This will be valuable in such events as a pad abort where a local hazard, such as a blockhouse, could be avoided during landing. Impact attenuation was found to be necessary for the Mercury capsule and in this new vehicle it will also probably be needed. Such devices as impact bags, collapsing struts, and honeycomb material should be investigated in order to provide sufficient latitude in choice.

In the event a glide vehicle is chosen, it will be necessary that an amphibious landing gear be provided inasmuch as both ground and water landings are a requirement. In addition, the L/D and wing-loading criteria, which will provide a satisfactory approach and landing maneuver, must be determined in the same manner as was used for the Dyna-Soar and X-15 projects. However, in this case, 30-knot winds and very rough water must be considered.

Finally, for the emergency and postlanding considerations, we must consider the effect of residual heated surfaces on the vehicle which might start local fires in the event of landing in brush or tall grass. Following water landings, the prime requirement is adequate flotation and water stability. Suitable location devices and survival support for the crew during the postlanding period will be needed following both water and ground landings in remote locations.

### Auxiliary Rockets

Figure 6 shows the requirements for the auxiliary propulsion system. Auxiliary propulsion propellants can either be solid or liquid, or possibly, a combination of these two. Inasmuch as there is a great difference between these two types of propulsion systems, the design

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criteria will be different. The auxiliary propulsion system must provide for a wide variety of escape and guidance maneuvers. It is also estimated that the number of maneuvers during a mission may be large. These requirements can be met with a solid-propellant system by use of a large number of rockets of various sizes. For a liquid system, restart capability will be required. In addition, it may be expected that more than one thrust unit will be required in order to provide sufficient thrust variation.

Reliability is a very important requirement inasmuch as the auxiliary propulsion system is the crew's "ticket" back home. In the case of the liquid rocket system, it is not only important that the liquid fuel propulsion system meet the present design criteria, such as in the X-15 where the emphasis is put on safety (the rocket will not blow up), but it is also necessary that the rocket will start and function every time it is needed.

Since this is a major system, a certain amount of redundancy is justified. In the case of a liquid-propellant system, extra thrust units and pumps will probably provide the minimum acceptable level of redundancy. For a solid-propellant system, a suitable number of spare units will be required. In either case, a failure analysis is needed to provide the redundancy criteria.

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## ABORT AND LANDING GUIDELINES

- II A. CAPABLE OF SAFE RECOVERY FROM ABORTS  
 B. CAPABLE OF GROUND AND WATER LANDING  
 (ALSO CAPABLE OF AVOIDING LOCAL HAZARDS)  
 C. CAPABLE OF POINT (10 SQUARE-MILE) LANDING  
 D. DESIGNED FOR 72-HOUR POST-LANDING  
 SURVIVAL PERIOD  
 E. AUXILIARY PROPULSION REQUIRED FOR  
 MANEUVER IN SPACE

Figure 1

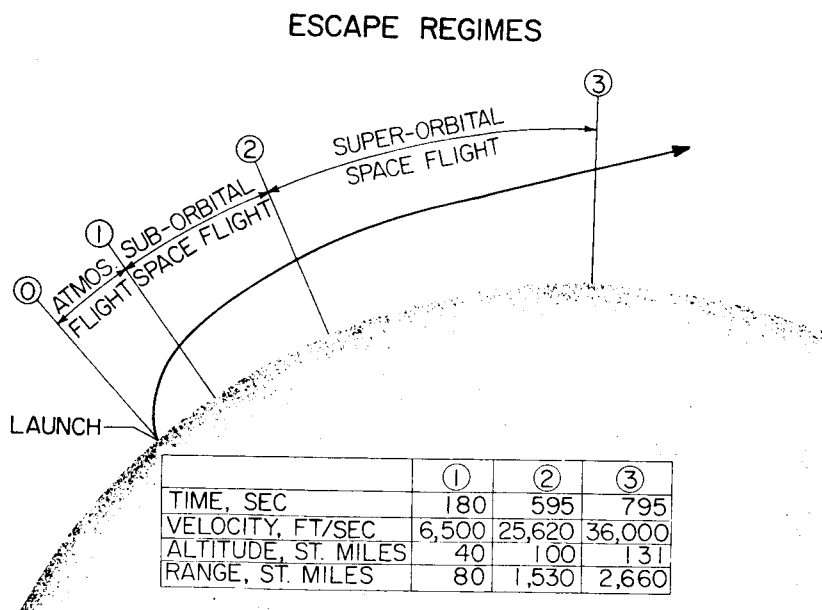


Figure 2

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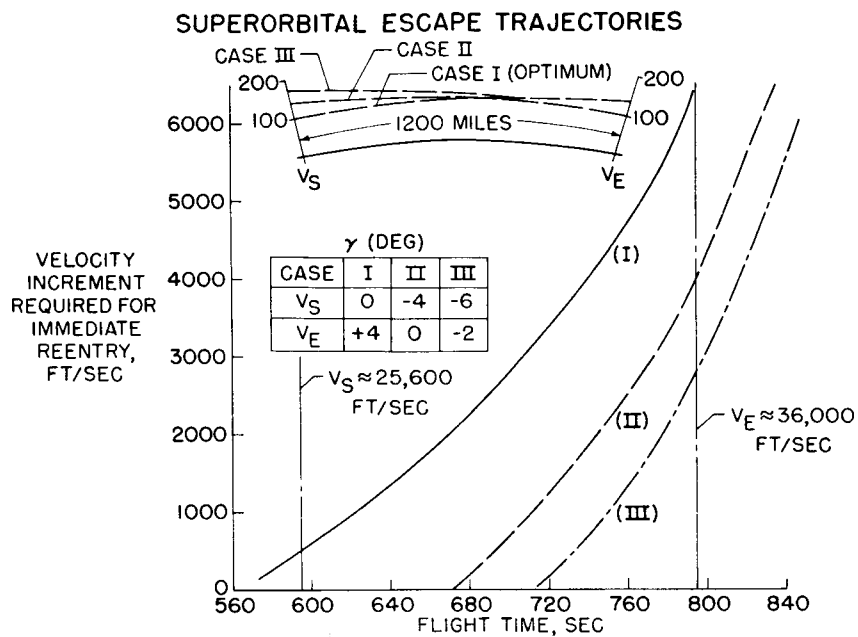


Figure 3

## AERODYNAMIC CONSIDERATIONS

### A. HEATING AND LOADS

1. BOW SHOCK RADIATION
2. THERMAL AND STRUCTURAL DESIGN CRITERIA

### B. AERODYNAMIC MANEUVERS

1. TRADE-OFF BETWEEN L/D, GUIDANCE, AND MANEUVERING ROCKET CAPABILITIES
2. POST BLACKOUT MANEUVERING REQUIREMENTS

### C. CONFIGURATION

1. COMPROMISE BETWEEN ABOVE AND OTHER GUIDELINE REQUIREMENTS
2. WEIGHT CONSIDERATIONS

Figure 4

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## LANDING CONSIDERATIONS

- A. PARACHUTE SYSTEM
  - 1. DEVELOPMENT OF RELIABLE LARGE PARACHUTES
  - 2. TRANSLATION CAPABILITY
  - 3. IMPACT ATTENUATION
- B. GLIDERS
  - 1. AMPHIBIOUS LANDING GEAR
  - 2. L/D AND WING LOADING SATISFACTORY FOR APPROACH AND LANDING MANEUVER
- C. EMERGENCY AND POST LANDING CONSIDERATIONS
  - 1. GROUND LANDING - HEATED SUFACES
  - 2. WATER LANDING - FLOTATION AND WATER STABILITY
  - 3. LOCATION AND SURVIVAL TIME

Figure 5

## AUXILIARY ROCKETS (SOLID OR LIQUID)

- A. MULTI-PURPOSE
- B. MULTIPLE FIRINGS
- C. RELIABILITY
- D. REDUNDANCY

Figure 6

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### HUMAN-FACTORS CONSIDERATIONS

by Stanley C. White

#### Introduction

The 2-week duration of the mission and the need for the occupants to rely on their internal resources, rather than continually relate to the earth, determines many of the Human-Factors requirements. This fact can be best illustrated by comparing this proposed circumlunar flight with the Mercury flights now contemplated. The Mercury capsule flight can be better compared to a high-performance aircraft program in that computation and much of the control can be effected from the ground. The Mercury Astronaut is in nearly continuous touch with the ground stations and has need for only limited provisions for habitability because he need neither eat, sleep, eliminate waste, nor find relaxation during programed missions. In contrast to this, the lunar astronauts are more in the positions of men on a long voyage. They must have provisions for eating, sleeping, waste evacuation, recreation, care of their health, as well as a more complete work space since the ground rule of the maintaining command control aboard the vehicle has been established. Furthermore, the long period of relative isolation of the group will threaten mental disturbance unless some freedom of movement and relief from boredom can be arranged by the provision of a variety of tasks. The size and dimensions of the capsule dictate that the crewman will be living under unsatisfactory small group society rules. This fact will require very thorough study and provision within the capsule in order that the problems of crew relationship may be alleviated. The duration of flight and the fact that man must face weightlessness and isolation for many consecutive days raise the question of crew selection to assure compatibility.

Many of the above-noted items are subject to solution through relatively straightforward mechanical- and human-engineering studies. One item, however, is not now considered to be in such good shape: that is the problem of radiation in space, particularly during solar flares.

Figure 1 lists the crew and environment guidelines. Briefly, these guidelines state that we should plan on having a "shirt sleeve" environment for a minimum crew of three men and that we must design the vehicle (and the mission) to provide adequate radiation protection for the crew.

#### Required Scientific Experimentation

The magnitude of the step proposed in the cislunar flight over that accomplished in Mercury and the demand for placing the control center of

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the operation in the capsule under control of the astronaut require a rather extensive biological program to be accomplished prior to the time when the cislunar flight will be undertaken. These experiments should go along with the other developmental tests and data-gathering flights which are planned as intermediate steps in the development program. This discussion will be confined to an outline of the types of experiments required within the life-sciences area, assuming that there will be an equal number of experiments required in both the physical data-gathering and the engineering test programs. The life-sciences program can be divided into roughly four categories. They are as follows:

(a) Psychophysiological (Habitation).- This area will include a study of the provisions for quarters and working area. This will include a study of the equipment that is placed onboard to assure that this equipment will allow for crew duties to be adequately performed. Also included will be the problem of crew association, namely that of small group sociological problems and the methods of the small group working together for such a venture. Here also, studies on the problems of sensory deprivation should be made. The problems of earth separation should be considered. A study of the methods of attacking boredom must be made. These studies should result in the determination of the magnitude of the problems in this area and should dictate the directions for solution such as the manner of work schedules, numbers of crew, relaxation, and entertainment provisions, and any other approaches to alleviate or ameliorate these as problem areas.

(b) Physiological.- Since this flight is to be a large extension of any experience we have had to date, it will be necessary to have studies which are primarily in the field of weightlessness and its effect over chronic exposure studies. These will include such things as studies of the respiratory reflexes, cardiovascular reflexes, cardiovascular efficiency, muscle tone, maintenance of muscle tone, metabolic aberrations, radiation estimation and the adequacy of the protection provided, and the methods for gathering such data, and should include the new instrumentation required to do this in a more efficient and less burdensome manner.

(c) Human Engineering.- This area will confine its experimentation to the actual crew performance for the job of successful flight. The study and experimentation on the analysis of the jobs to be done will be made. The crew layout, which naturally results from such an analysis, can then be better designed. The number of crew members who are required to work together in performing an operational task can be ascertained. The operational procedures, which must be established for successful flight, can be made and exercised through the development program, modified and retested to validate adequacy for leading toward successful flight. The final result of this area of study will be the

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development of mechanisms and the training devices for training crews for such a cislunar vehicle flight.

(d) Medical support for the above areas.- In this area, such studies as the natural development of micro-organisms occurring within a group living within confined quarters for a 2-week period must be made. As has been well-established in bacterial growth and their habits over earthbound existence, the natural incubation period for such bacteria falls within the 7 to 14 days planned for the maximum operation of this vehicle. This will mean, for example, that a healthy astronaut leaving on such a voyage could, by exposure to his crew members and the bacteria that each brings to the cabin, become quite ill during the flight time unless provisions are made within the environmental control system and through medications placed onboard to control such events. Another area that falls under the research program will include the establishment of the radiation tolerances, the devices for measuring the radiation occurring within the capsule during flight, and to ascertain any effects which may be detrimental to the flight during such exposure. Underlying this research program, of course, will be the desire to establish what can be handled by the crewman aboard and what must be provided, as far as basic equipment, to enable them to take care of themselves. In addition, the mechanisms for gathering such information aboard the vehicle when the problems present themselves during a flight must be established.

Many of these areas which have been listed under the research program will require both human and animal studies as a method for the establishment of validity to the answers of the questions that have been raised. The animals in this case will allow for multiple numbers of specimens to be exposed along with small crew groups in order that statistical numbers can be considered and data on tissues resulting from tests can be considered. The animal offers the opportunity to translate his effects to the human performance, while at the same time giving the bioscience advisors an opportunity to study pathologically some of the animals after flight. This permits the evaluation of the subtle changes seen in tissues if there are no changes in the crew anticipated or found during the flight, because it then establishes more firmly man's capability to withstand the rigors of such an advanced flight.

#### Environmental Considerations

The crew accommodations must provide a safe and comfortable environment for the duration of this mission. Figure 2 lists the general areas which must be considered as environmental factors. Before going into a detailed discussion of these factors, certain basic considerations should be reviewed.

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The long duration of the flight precludes the continuous use of a full-pressure suit; therefore, alternate methods for decompression protection must be devised to allow a "shirt sleeve" type environment for normal operation. The long confinement period in close quarters may require some accommodations for individual privacy, recreation and means of physical exercise. The cabin arrangement must be functional and still provide a satisfactory liveable space environment.

Let us now consider the areas listed in figure 2.

(a) Atmospheric control. - The general requirements for atmospheric control are as follows:

- Hermetically sealed cabin
- Pressurization control
- Breathing oxygen
- Carbon dioxide removal
- Temperature control
- Humidity control
- Control of toxic gases
- Instrumentation

(1) Sealed cabin - The reentry vehicle and space laboratory cabins should be hermetically sealed with provisions for repair or self-sealing of leaks. Each cabin must be considered as a self-contained independent pressure vessel. Repairs to the pressure vessels should be made from the capsule interior and not require the occupants to leave the cabin.

(2) Breathing oxygen and pressurization - A system must be developed which can supply oxygen at a rate of 2.0 pounds/man/day and remove carbon dioxide produced at a rate of 2.5 pounds/man/day. These metabolic data should be confirmed under actual flight conditions as part of a general capsule habitability study. The oxygen partial pressure must be maintained at a minimum of 160-mm Hg to maintain normal blood saturation levels. To determine the total cabin pressure and gaseous composition, investigations into the physiological requirements, structural weight, and system weight and complexity must be made. From such a study, it can be determined if a two-gas system, i.e., oxygen-nitrogen, at 14.7 psi is feasible or required. The selection of oxygen sources must be a compromise between storability, reliability and weight. To achieve these objectives and to provide the required redundancy, two types of oxygen sources may be required. For example, a liquid-oxygen source could be used as the primary supply with a chemical superoxide system as the backup or emergency supply.

(3) Carbon dioxide removal - The carbon dioxide partial pressure must be maintained below 4-mm Hg at all times. This control and removal of carbon dioxide may require the use of chemical absorption or the utilization of materials to collect carbon dioxide from the cabin environment and subsequently dump the CO<sub>2</sub> overboard. With the duration of this flight and the current state of technology, it does not appear that a regeneration system for breaking CO<sub>2</sub> into O<sub>2</sub> and carbon is feasible or required.

(4) Temperature control - The system must be capable of controlling and maintaining the cabin temperature between a selectable range of 65°-75° F. This will require removal of the metabolic heat loads of 300 to 700 Btu/hr/man with peak rates as high as 1,400 Btu/hr/man. In addition, the electrical heat load must be removed by this system. The use of a refrigerant cycle or coolant liquid appears impracticable due to weight and power requirements. It therefore seems probable that a radiant-type heat removal system will be required.

(5) Humidity control - A method must be devised for removing respiration and perspiration water vapors in the cabin. This removal system may be developed in conjunction with the temperature control system in which the vapors could be condensed and collected.

(6) Control of toxic gases and micro-organisms - Provisions for removal of toxic materials such as carbon monoxide, methane, etc., must be made. Consideration must be given in the selection of capsule materials and in capsule operation to prevent inclusion of micro-organisms which might have nurtured and multiplied in space flight to cause illness or systems interference.

(7) Instrumentation - Basic instrumentation must be provided to monitor the capsule environment. This must include temperature-pressure measurements and partial pressures of oxygen and carbon dioxide. If possible, a measure of toxic materials such as hydrocarbons, C<sub>0</sub>, etc., should be made.

(b) Decompression protection.- This space vehicle must have a highly-reliable pressurized cabin and laboratory to preclude the use of a pressure suit in normal flight. There are several approaches to the problem of decompression protection. The first approach would be a quick-donning pressure suit together with subdivision into two compartments with an air lock in-between. Pressure suits would only be used in critical times such as launch or reentry. The second approach would be to provide a small rigid module for the crewmen in which they can be pressurized above the bends level and have emergency controls to effect reentry and landing. This approach, however, does have the disadvantage of not allowing the occupants to move from a pressurized compartment into a decompressed compartment in case of an emergency.

(c) Acceleration and restraint.- The resultant vector accelerations caused by linear, oscillatory and rotational motion and applied to the astronaut support couch or restraining harness during either normal or aborted missions shall not exceed accepted limits. Figure 3 summarizes the known data for sternumward forces for pure accelerations in directions perpendicular to the spine and for lateral loads and the resultant vector loads. From this plot, it can be seen that 20g in the perpendicular and 10g in the lateral are the presently accepted tolerance levels. Current studies at Wright Air Development Division (WADD) indicate that these levels may be exceeded without injury; however, data from these tests are not complete. Figure 4 summarizes the accelerations that human beings have endured for the time duration shown. Tolerances may be greater but these are the best data available at this time. The duration of sustained load is an important variable and only data having significant durations were plotted. The gaps from 0.01 to 0.04 second exist since this is in the region of impact-type decelerations in which onset rates may affect tolerance. Another gap exists in the data from  $1-\frac{1}{2}$  to 8 seconds.

The accelerations applied must be limited to these values because tolerance data justifying larger accelerations are not available for vector directions other than parallel to the principal body axis. Furthermore, intelligent vehicle control efforts do not appear possible when the accelerations imposed exceed these values. If these limits are exceeded, additional tolerance and control studies must be conducted.

If tumbling, spinning, or rotation occurs during the mission, the rate should not exceed one revolution per second and the duration should not exceed 1 minute. These limits have been demonstrated as tolerable limits (Ref.-Journal of Aviation Medicine, Feb. 1954, pp. 5-222).

The design of the restraint couch and harness system should place the occupant in a supine position during launch and reentry. This position provides maximum tolerance protection to linear accelerations.

Several approaches to couch design should be investigated - for example, the net couch or a universal formed couch which utilizes particulate matter that becomes rigid when evacuated but remains pliable and readily contoured in the absence of a pressure differential. The couch must be designed to serve as a bed and, if possible, the couches should be designed to fold away to give maximum floor space in flight. The integration of decompression protection into the couch must also be investigated.

The restraint harness must be designed to restrain the crewman in all directions in which accelerations will be encountered. The harness must be easy to attach, simple to adjust and free of loose straps, etc.

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to provide a clean harness installation. The use of an integrated restraint garment must be investigated to achieve these goals.

If emergency accelerations exceed the tolerance limits, attenuation devices must be provided as backups to the capsule-landing attenuation system.

(d) Noise and vibration. - The noise level within the capsule must not exceed those limits shown by figure 5. During normal flight the vehicle noise level should be maintained below 40 decibels. Vibrations during launch and reentry must be maintained below the "painful" level indicated by figure 6 and below the "unpleasant" level during normal flight.

(e) Nutrition. - A balanced diet of approximately 3,000 kilocalories/day/man must be provided. The foods must be palatable and provisions for warming the foods must be considered. If possible, low-residue-type foods should be provided to ease the waste-handling problem. The types of food and the dispensing equipment must lend themselves to weightless operation.

(f) Waste disposal. - With the relatively short duration of this mission, it does not seem practical to attempt to recycle and utilize the fecal wastes. Therefore, provisions for storing and decontaminating the fecal waste must be made.

The recycling of urine, however, appears to be worthy of investigation. Figure 7 indicates that 168 pounds of urine would be available for reclamation; this coupled with water from perspiration and respiration would greatly reduce the onboard water requirements provided a highly reliable purification system is developed.

(g) Interior arrangement and displays. - Since the space available will be limited, every effort should be made to make what space there is usable for several purposes (use of fold-away devices, etc.).

In general, provisions for privacy and a sleeping, general living and eating area apart from the laboratory would be desirable. Sleeping may best be accomplished in a zippered sleeping bag that would restrain the crewman and provide decompression protection.

Provisions for recreation, personal hygiene, and exercise must be made. The use of radio and reading materials should provide adequate outlets for the recreation requirement. Use of microfilms may prove useful in this problem area. Dry-wash techniques for personal hygiene and the use of ultrasonic cleaners for maintaining clothing should be investigated. Some type of exercise equipment such as a treadmill or

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rowing machine must be provided to maintain body tone. The exercise equipment may be developed to transform the mechanical energy into electrical energy to recharge onboard electrical supplies.

Figure 7 summarizes the metabolic requirements for this mission. This data is presented only as "ball park" data to give a feel for basic requirements.

(h) Bioinstrumentation. - The amount and types of instrumentation in the three-man capsule should not differ materially from the Mercury capsule instrumentation. Practically the same parameters will need to be measured and the increased size of the capsule will not need to increase the size of the instrumentation package to any extent. With less instrumentation per man and a division of responsibility among the crew members, the monitoring of the capsule should not require a large portion of the men's time. An automatic alarm and warning system for dangerous conditions and malfunctions should be provided so the men will not have to monitor continuously and be available for other useful functions.

The bulk of the instrumentation should be placed in the reentry vehicle so most of the control can be carried out in this portion and duplication in the laboratory portion will be unnecessary. In any case the control will be needed during launch and reentry in this portion.

#### Radiation

The area of radiation and its measurement (also its protection) is probably one of the most controversial ones that exist in the entire system. The levels of radiation which the man will accept during routine operation should be kept, through design, to an absolute minimum. By this technique, the man will have every opportunity to use his total maximum dosage limit during the times of solar activity. The maximum allowable dosage permissible for an astronaut has been discussed with many experts within the United States, and the general consensus of opinion is that 25 REM is the upper limit. To demonstrate the problems of radiation effect, the best guesses of these experts and what they predict would occur in the astronaut when compared to the total exposure, can be seen in figure 8.

It seems obvious that the significance of the maximum dosage of 25 REM, as mentioned above, is that we are in an area, as predicted by the experts, where a rapid increase in both the chronic as well as the acute changes, is beginning to occur. It is probably of equal significance to note that this is a guess of percent of effect rather than tested data in this area. This, we believe, summarizes very well the problem and need of gathering hard figures which can be used for design and prediction of effect of radiation on the astronaut. For example, there is a small group in the United States which feels the dosage

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techniques presently being used may not be really applicable to space exposure. If this is true, then the individuals receiving 25 REM in the present measurement technique would not receive the total dosage effects predicted. In addition to this, it should be noted that an individual receiving the 25 REM maximum dosage over small increments over a period of time would not get a completely additive result. For example, if the man received 10 REM on the way out through the radiation belts and 15 REM 2 weeks later coming back through the radiation belts, the total effect would not be 25 REM, but some increment less than this. The 25 REM dosage mentioned above is considered as the exposure of a man on a single dosage. The radiation exposure can be reduced through the proper utilization of trajectory planning and also through the use of acceleration planning to schedule shorter times of flight through the belts. The best estimates at this time would indicate that through the use of the equipment aboard the vehicle as shielding, the use of trajectory planning, and the use of time in radiation-belt exposure as methods of protection, the crew will provide sufficient latitude for design and flight planning to permit the flight of such a cislunar vehicle within the state-of-the-art today. In the event a solar flare occurs during the flight program of either the orbital or of later cislunar flight, serious consideration as to what dosage limits will be, under these conditions, should be made through experimental studies; and the techniques which would be advisable for the crew should be established. Presuming that the further studies in the radiation areas do not develop any additional new problems for our planning and designing people, there still remain to be solved several problem areas in the vehicle design. An actual establishment of a radiation dose should be made. This includes the validation of either the present measurement technique of dosage or the validation of the volumetric dose technique. After the dosage limits have been established, then the development and testing of the equipment which should be placed aboard the capsule for the measurement of the accumulating dosage for the crewman must be made. This instrumentation will have to reflect a capability of reading the accumulated dosage for the crewman as well as a warning system for the men when a certain level of tolerance has been exceeded. It should be such that the men can easily ascertain their day-to-day situation.

Crew Size

The number of astronauts required for the crew of a cislunar flight varies with the following factors:

- (a) The number of crew jobs required for successful flight. These tasks must be further divided as to the relative priority of each task, the number of tasks which must be done concurrently, etc. As an example, the tasks can be divided into those of primary importance (vital to flight) and those of secondary importance (essential but of lesser

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magnitude). For example, the tasks of pilot, communicator, navigator, engineer, and flight conductor may be considered as primary tasks. Such areas as health monitor (radiation count, medical care for injury, etc.), commissary supervisor, waste material supervisor may be considered secondary. It is possible for one man to be cross-trained in more than one area. However, the desire to have one man primarily responsible for more than one area must be modified by such events as how many men will be required to perform simultaneously to make a successful flight (i.e., a midcourse guidance will require both pilot and navigator to operate simultaneously). In addition, the complexity of each of the primary tasks dictates that one man cannot be counted upon to do a reliable job in more than two of these areas. He can act as a backup on some of the other areas through cross-training. Actually, the job areas are analogous to the present-day B-47, B-52, and B-58. A man on the crew can do one task or make one decision at a time; therefore, more men are requested. The first study indicates a minimum crew size of three men to handle the total task area.

(b) The duration of the orbital and cislunar flight will cause the diminution of the capability of a given crewman to perform under an abnormal on-duty-off-duty schedule. Man can have his normal 8-hour work cycle interrupted for a series of 2 or 3 days with a small but increasing loss in efficiency; even here, breaks must be provided. However, when this interruption is maintained for more than 2 to 3 days, a standard 8-hour shift system must be considered in the flight as maximum. This dictates more than one crewman will be needed to cover the flight duration of 2 weeks, if the desire to operate continuous surveillance and communication capability is met. Again, the minimum crew size will be three for a 24-hour period of operation with each working 8-hour primary shifts with extra short periods of specific help on problems.

(c) The level of alertness required by the crewman will cause further modification of the 8-hour work shift. If monitoring is the task, then shorter work cycles will be required. This will increase the need for multiple men. For example, a pure monitoring task such as surveillance of vehicle operation will set a 4-hour maximum single period performance without rest or severe loss of alertness.

(d) Redundancy of man's system to prevent abort of the flight due to injury or illness of a crewman must be considered. Presuming a second crewman can support the injured or ill astronaut, thus increased reliability of the flight can be obtained by multiple crew. This will simplify the problems of inadvertent recovery at unprepared sites. Serious consideration of flying the total mission rather than aborting should be made due to the complexity of recovery from an unscheduled earth impact.

(e) The utilization of this vehicle as an intermediate orbital laboratory dictates space and provision for carrying aloft experiments

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and experimenters necessary to demonstrate man's capability to operate under these advanced flight precepts. In these tests, it would appear essential that personnel other than normal flight crewmen will be required to complete the tests with a minimum of flights. This becomes essential if the stated desire to reduce bioinstrumentation monitoring on the astronaut is held to problem areas in astronaut provision and habitability is eliminated before the cislunar flight. The significance of this type of approach becomes apparent where the need for a reliable crew is considered as vital to the cislunar flight, and the automatic recovery aids are reduced or eliminated.

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## CREW AND ENVIRONMENT GUIDELINES

- III A. DESIGNED FOR "SHIRT-SLEEVE" ENVIRONMENT
- B. DESIGNED FOR THREE-MAN CREW
- C. DESIGNED FOR RADIATION PROTECTION

Figure 1

## PHYSICAL ENVIRONMENT FOR CREW

ATMOSPHERE CONTROL  
DECOMPRESSION PROTECTION  
ACCELERATION PROTECTION  
NOISE AND VIBRATION  
NUTRITION  
WASTE DISPOSAL  
INTERIOR ARRANGEMENT AND DISPLAYS  
BIOINSTRUMENTATION

Figure 2

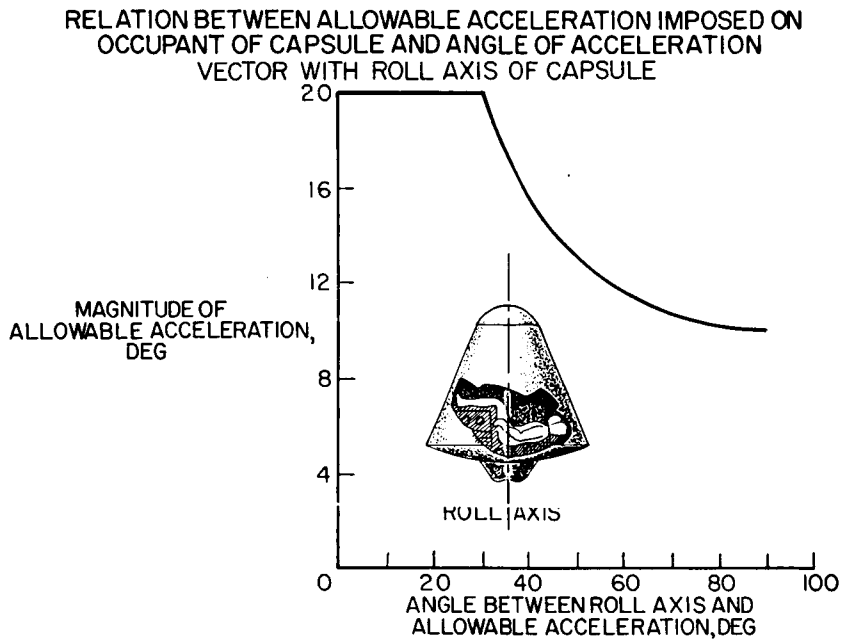


Figure 3.

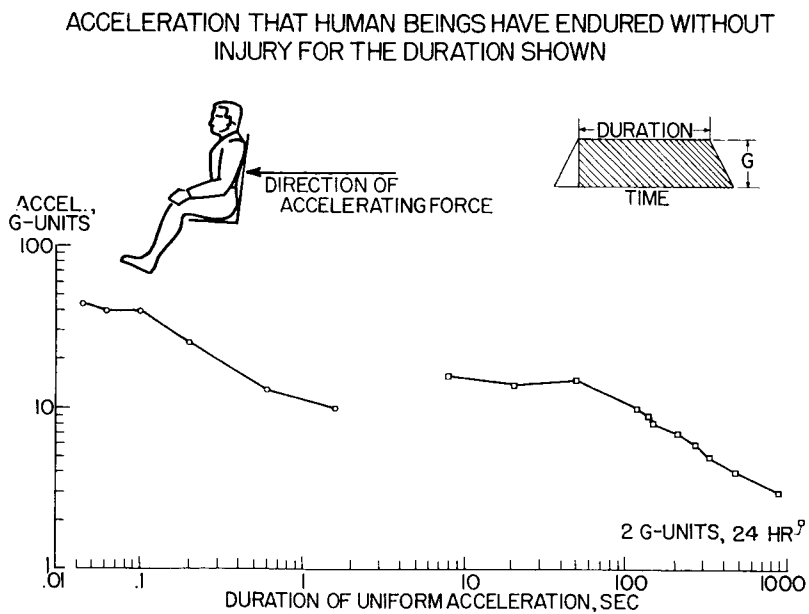


Figure 4

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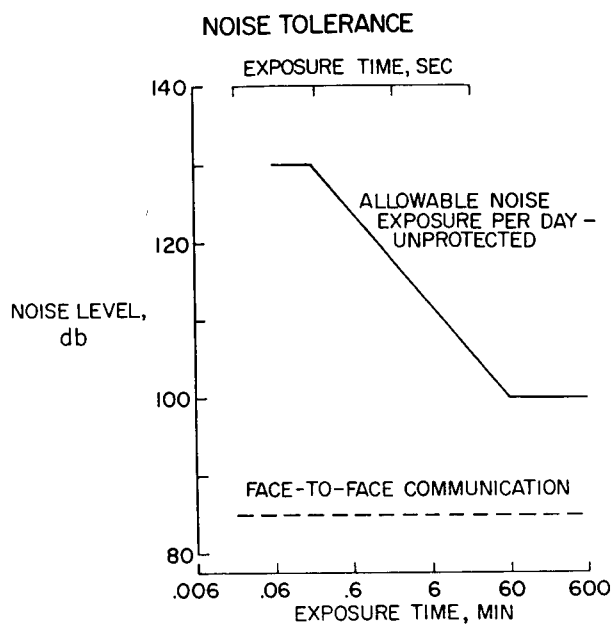


Figure 5

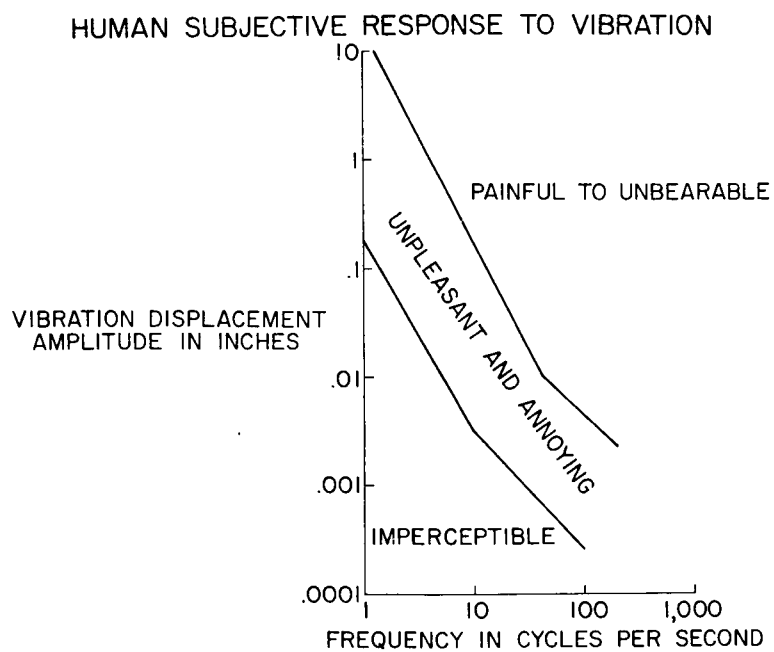


Figure 6

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### THREE-MAN-14 DAY MISSION

#### ESTIMATED METABOLIC REQUIREMENTS

DRY FOOD - 84 POUNDS  
(126,000 k CALORIES)  
H<sub>2</sub>O - 239 POUNDS  
\*OXYGEN - 84 POUNDS

#### BY-PRODUCTS

|                   |              |
|-------------------|--------------|
| CARBON DIOXIDE    | 105.0 POUNDS |
| WATER             |              |
| RESPIRATION       | 37.8 POUNDS  |
| PERSPIRATION      | 54.6 POUNDS  |
| URINE             | 168.0 POUNDS |
| FECES             | 16.8 POUNDS  |
| TOTAL             | 277.2        |
| RECOVERABLE WATER | 260.4 POUNDS |

BASED ON MAN USING 3,000 k CAL/ DAY FOODS  
R. Q. - 0.8

\*NO ALLOWANCE FOR CABIN LEAKAGE, ETC.

Figure 7

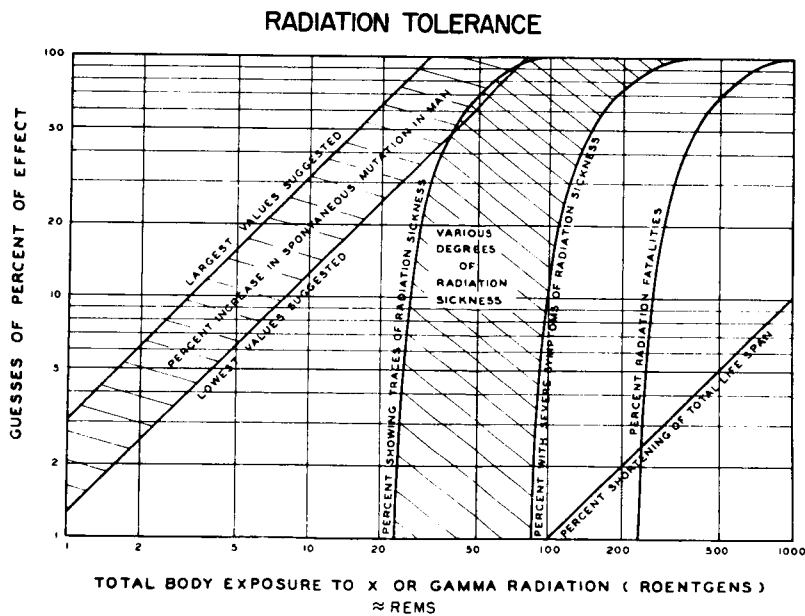


Figure 8



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### COMMAND AND COMMUNICATIONS

by Robert C. Chilton

#### INTRODUCTION

The manned circumlunar vehicle is an important developmental step toward the goal of establishing a manned scientific outpost on the moon. Man's role on the moon or in space can be fulfilled only if he can achieve the means of controlling his environment and this includes the command and control of the flight of his spacecraft. The next two guidelines, figure 1, concern this aspect of the program: (a) Onboard command of the mission and (b) Communications and tracking necessary to support this concept of manned operations in space.

#### Onboard Command

The first guideline states that primary command of the mission should be onboard. The vehicle should be designed solely for manned operations with no systems requirements for carrying out unmanned qualification missions. A primary objective of the circumlunar program is the development of operational techniques for manned vehicles. This development has its beginning in the Mercury program. It will be expanded in the orbital phase of the circumlunar program, and by the conclusion of the circumlunar program, the full scope of manned space operations will have been explored except for lunar landings and take-offs.

The payload cost of supporting a man in space is very large in comparison with unmanned space operations. There is also a great cost in system complexity for providing the mission abort capability necessary for manned space flight. Therefore, when manned space flight is the basis for design, the maximum advantage must be taken of the command decision and operational capabilities of the crewman. It is impractical to provide a completely automatic mission operated by ground command. Furthermore, it is our opinion that by concentrating on the manned operational concept we can achieve an earlier and greater probability for success toward the ultimate goal of accomplishing reliable flights to and from the surface of the moon.

#### Crew Responsibilities

It should be emphasized at this point that the manned operational concept does not imply that all operations are manually performed. There is a definite requirement for such automated functions as attitude control and abort-sensing to obtain high accuracy, short response time, and pilot relief. The correct implication of manned operations is that the crew is given the command control of the vehicle and its systems by

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judiciously assigning tasks to the crew, by cross-training the crew in the various tasks, and by providing means for checking and correcting systems performance.

Figure 2 indicates the system responsibilities of the crew. The primary task is to make command decisions and monitor systems performance. In particular, this includes supervision of navigation and control. The secondary task is to maintain systems performance by accomplishing necessary maintenance and repair or by engaging the appropriate backup systems. The emergency task is to take over by manual modes certain critical functions. At this time, these functions cannot be described in detail. A study is required to determine the extent of the capabilities of the crew to perform these tasks and to specify what is reasonable to expect for routine, urgent, and extreme emergency conditions of operation.

#### Hardware Requirements

The flight hardware should be designed not only for intrinsic reliability, but also, when practical, it should be designed to incorporate special features for operational reliability such as replacement modules for easy maintenance, self-checking, and manual correction modes. Some of the hardware requirements for onboard command are shown in figure 3. This does not represent a configuration for any particular mode of control, but rather a grouping of subsystems showing their relationship to one another. The primary inputs to the crew are from the display console and from the ground complex. Not shown are the pilot inputs to the subsystems. It is anticipated that he will have inputs to every subsystem. A brief description of the subsystems and some of the problems they suggest will be given in the following sections.

Inertial Systems.- Applications of gyros and accelerometers supply short-term integration for position and velocity determination and short-term attitude reference. Both stabilized and body-fixed applications may be employed in the best configuration for overall operational reliability. Inertial platforms will be employed for monitoring insertion guidance, abort command, and for abort-reentry navigation. The exact role of the inertial platforms in midcourse and normal-reentry navigation will be determined by careful study of trade-offs between system accuracy, power requirements, weight, and techniques for correcting and updating the system by intermediate reference. Body-fixed systems may be attractive for backup modes.

Celestial-Optical.- These systems include telescopes, star trackers, horizon scanners, and optical ranging devices. They supply inertial attitude and position reference, and local reference information in the vicinity of earth and moon. Optical devices may operate in a variety of modes and careful study will reveal the extent to which they may be

employed to obtain maximum operational reliability for manned space flight to the moon.

Onboard Computer.- The computer represents a capability to perform a number of digital calculations, some of them special purpose and some general purpose. The entire concept of onboard control depends upon the development of versatile, reliable, long-lived computers. The computer must handle all the navigation computations whether insertion, midcourse, reentry, abort, or orbital operations around the moon. Midcourse and reentry guidance concepts must be evolved which will simplify the computer requirements and which will allow relatively simple schemes using redundant hardware for obtaining the capability of checking and correcting system performance.

Onboard Display.- The display provides all the system performance indices and mission status parameters necessary for the pilot to carry out his assigned role. It is integrally related to the computer and inertial systems. Display requirements for reentry control have received quite a lot of attention in the study phases of the Dyna-Soar program. These must be continued for the circumlunar vehicle with added emphasis on the aspect of control of reentries from aborted missions. Display requirements for insertion guidance monitoring have received less attention, and the other phases of the mission are as yet unexplored from the standpoint of developing displays which will put the pilot squarely in the middle of the onboard systems with means for making real inputs for achieving operational reliability.

Attitude Control.- Attitude control employs aerodynamic surfaces, reaction jets, reaction wheels, etc., for control and utilizes the inertial and celestial-optical systems for reference. The accuracy of the reference systems is determined by the navigation requirements which may or may not be reflected as attitude-control requirements. Vehicle orientation for midcourse corrective impulse is not expected to require precise control. Other requirements for attitude control are for antenna look angles, solar orientation, control during thrust application for abort and lunar retrograde and escape, rapid reorientation for some abort cases, and control during reentry. Reentry control and low-altitude abort control must stress reliable stability augmentation modes.

A multimode system will be necessary to satisfy all the requirements. Aerodynamic control is attractive for the low-altitude abort and the atmospheric reentry. Combinations of jets and wheels will likely provide the long-term stabilization requirements. Techniques for obtaining reliable long-period limit cycles will be very important for economical use of fuel. Jet propellant performance characteristics must be compatible with operational and system requirements. Either high thrust jets or gimballed engines will be required for control during

the high thrust application required for aborts or lunar retrograde. Careful attention to the propulsion system design can simplify the attitude-control requirements.

Propulsion.- The propulsion systems provide the small corrective impulse for midcourse guidance and the large impulse for emergency abort, lunar retrograde and escape, and retrograde for reentry control from high-altitude aborts. The system should be designed for maximum reliability and flexibility, and for minimum demand on attitude control due to thrust misalignment.

Communications and Tracking.- Tracking data provide position and velocity data for the pilot or, if necessary, it can be read directly into the computer. This brings us to the next guideline.

#### Communications and Tracking

Communications and ground tracking should be provided throughout the lunar mission except when the vehicle is blanketed by the moon. Communications and tracking are important to keep the ground crew informed of the status of the mission and to obtain and relay navigation data to the vehicle. Naturally, for the lunar mission it will be desirable to maintain continuous contact for as long as possible. The orbital mission, however, will be similar to the Mercury program except for the altitude and length of time in orbit. At present, it seems unlikely that the cost in the number of ground stations required to maintain continuous tracking during earth orbits can be justified. Contact once each orbit appears to be a reasonable and adequate requirement.

Figure 4 describes the operational requirements for communications and tracking for the lunar and orbital missions. The methods employed for transmitting messages should provide the greatest practical coverage for both the lunar and orbital missions. Voice contact once per orbit is considered sufficient for orbital missions; however, other means, such as CW transmission should be provided for essentially continuous contact. For the lunar mission, telemetry will be required only for backup data. The crew will be capable of relaying periodic status reports on men, systems, and vehicle; therefore, the strongest requirement for telemetry will be for providing data in case of disaster. Telemetry for the orbital mission may be as complete as necessary to satisfy the requirements for the development program. Television for the lunar mission may be desirable and practical in the time period involved. There is no apparent requirement for television for the orbital mission. A mission and operations analysis should be made to determine the number and types of communications systems and their duty cycles which will be required for efficient and reliable communications.

Ground tracking will provide trajectory data continuously for the

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lunar mission to the extent possible and once per orbit for the orbital mission. These data in the form of position and velocity fixes will be relayed to the vehicle for midcourse and orbital navigation. Ground tracking in some form may appear in the guidance loop for approach and reentry control.

Figure 5 illustrates the requirements for ground tracking. It must be emphasized that this is purely speculative. There are shown at least two different systems. One system provides tracking data during the atmospheric reentry phase and terminal-point control; the other system has the greater range required for midcourse tracking. The Mercury range may offer the best means of satisfying the close-in requirement. A study should be made to determine whether the Mercury range can be modified and relocated to meet the lunar operational requirements taking into account the reentry control capability of the vehicle and the ability to control the time of approach to the corridor.

The midcourse tracking requirement may be satisfied by the presently planned deep space net with facilities at Goldstone, and in Australia and Africa. This system should be studied for applicability to the circumlunar mission taking into account tracking accuracy and requirements for data processing. Radio interferometer systems should also be studied. Finally, it will be important to investigate existing or proposed facilities to insure that the frequencies for all systems can be made compatible in order that a single beacon may be used for midcourse and reentry tracking.

The last figure, figure 6, summarizes the problem areas which have been mentioned in connection with this guideline:

(a) Determine the detailed requirements for communications systems with particular attention to range, power, and duty cycle.

(b) Study the tracking requirements for reentry control and determine the applicability of the Mercury range.

(c) Study the requirements for midcourse tracking and determine the applicability of the deep space net.

(d) Determine the feasibility of providing compatible frequencies for midcourse and reentry beacon requirements.

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## COMMAND AND COMMUNICATIONS GUIDELINES

IV A. PRIMARY COMMAND OF MISSION  
TO BE ON BOARD

B. COMMUNICATIONS AND TRACKING  
FACILITIES NEEDED

Figure 1

## CREW RESPONSIBILITIES

PRIMARY: COMMAND DECISION, MONITORING,  
NAVIGATION, AND CONTROL

SECONDARY: SYSTEM REPAIR AND MAINTENANCE

EMERGENCY: MANUAL MODES

Figure 2

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## HARDWARE REQUIREMENTS FOR ONBOARD CONTROL

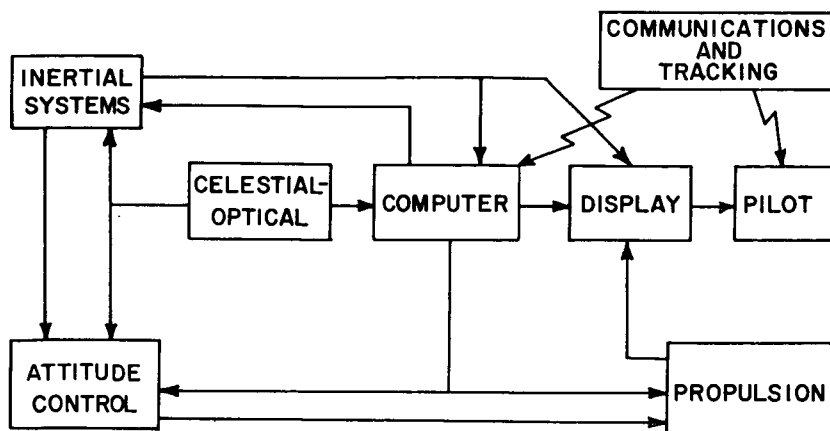


Figure 3

COMMUNICATIONS AND TRACKING,  
OPERATIONAL REQUIREMENTS

| COMMUNICATIONS:      | <u>LUNAR MISSION</u> | <u>ORBITAL MISSION</u> |
|----------------------|----------------------|------------------------|
| (a) VOICE, CW, OTHER | NEAR CONTINUOUS      | CONTINUOUS             |
| (b) TELEMETRY        | SPECIAL REQUIREMENTS | COMPLETE AS REQUIRED   |
| (c) TELEVISION       | POSSIBLE             | NO REQUIREMENT         |
| TRACKING:            | <u>LUNAR MISSION</u> | <u>ORBITAL MISSION</u> |
| (a) TRAJECTORY DATA  | NEAR CONTINUOUS      | ONCE PER ORBIT         |
| (b) NAVIGATION FIX   | MIDCOURSE            | PRE-RETROGRADE         |
| (c) GUIDANCE LOOP    | REENTRY              | REENTRY                |

Figure 4

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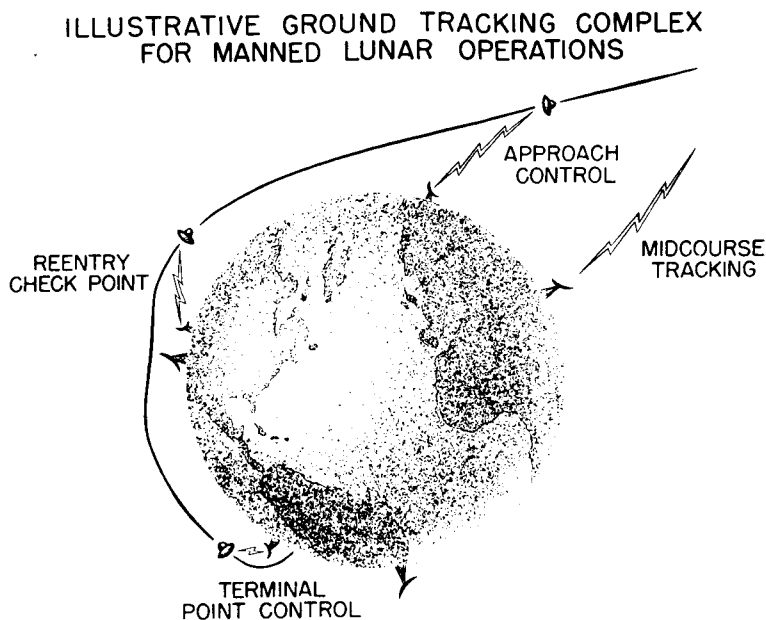


Figure 5

## COMMUNICATIONS AND TRACKING, PROBLEM AREA

- (a) NUMBER AND TYPES OF COMMUNICATIONS SYSTEMS  
(RANGE, POWER, DUTY CYCLE)
- (b) TRACKING REQUIREMENTS FOR REENTRY CONTROL  
(MODIFY AND RELOCATE MERCURY TYPE FACILITIES)
- (c) TRACKING REQUIREMENTS FOR MIDCOURSE CONTROL  
(DEEP SPACE NET, RADIO INTERFEROMETER - ACCURACY,  
DATA PROCESSING)
- (d) COMPATIBLE FREQUENCIES FOR REENTRY AND MIDCOURSE  
SYSTEMS

Figure 6

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SUMMARY AND SCHEDULING

by Charles J. Donlan ✓

Introduction

This paper attempts to recapitulate some of the problem areas pertinent to manned space flight that have been discussed in the preceding papers and indicates some of the phasing problems which must be faced in incorporating information resulting from research investigations into an active program for the procurement of hardware. In this regard, the time table that is currently being used in planning the work and the manner in which we propose to coordinate activities will also be discussed.

Summary of Problem Areas

The problems that have been discussed by the previous speakers have covered a wide range of subjects. For the purposes of discussion, figure 1 is a chart that groups these problems of multimanned space flight into three categories - the mission, the vehicle and the operation. Appendices II and III list other problem areas in slightly greater detail. It is not the intention to single out or define the problem areas best suited for study by any particular Research Center. Matters of this kind, of course, are best determined by the Centers themselves. It is anticipated, however, that the Research Centers, in orienting their thinking toward the objectives outlined in this presentation, will develop a comprehensive problem list that would serve themselves and the Space Task Group as a guide to activity in this area. This list of problem areas, of course, is not complete, but it contains many items of problem areas that Mercury experience has shown must be considered and thoroughly evaluated before one would be able to prepare specifications for a mission such as this. Once the extent of the NASA in-house effort on these problems is known, we will be in a position to issue complementary study contracts with industry and other groups to cover specialty areas not otherwise provided for.

In the mission area, considerable study is needed relative to the mission itself. For example, what part should the pilot really play? Should he be an active element in the system? What use can be made of the existing ground networks available in the time period? It may well turn out that emergency conditions dictate much of the essential elements of the design in the same manner that the escape and abort considerations for Project Mercury had a great influence on the design of that vehicle. The vehicle itself will have many of the problems reminiscent of those encountered in the Mercury program. But these will be more difficult to solve than in the case of Mercury. The problem of heating will, of course, be a serious one although it is one about which more is known perhaps than some of the other problems. Even in Project Mercury, we

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find that we are pushing the technology of materials to the limit. The radiation hazard and the required protection will involve a tremendous research effort if, indeed, flights of the kind envisioned can be considered at all. In the other areas listed in the vehicle column, one will recognize regions requiring extensive investigations before the completion of any design can take place.

The items under operations reflect Mercury experience. Booster reliability and safety problems become an order of magnitude greater in difficulty because of the superorbital flight regime. We know from Mercury experience that the abort-sensing problems and the launching techniques must be carefully programed. We are facing, perhaps for the first time, the problem of training people for real space flight. Mercury, we hope, will supply us with some insight to fit these training requirements. This is a whole new area of operations for NASA and some plans must be forthcoming for providing answers to human factors problems.

A proposed timetable of events for initiating the procurement of hardware is outlined on figure 2. Following the disclosure of plans to Research Centers, it is expected to disclose in a similar way the plans to industry representatives. After studying the guidelines, it is hoped the Centers will be able to outline what problem areas they will undertake, thus permitting us to initiate industry studies for problem areas not otherwise covered. Some procedure for the exchange of information will be required. The Space Task Group will have a team or group following these programs continually. However, we see no substitute here for a direct and frequent communication between Research Center personnel and Space Task Group people. Frequent communication between working groups is essential. Also, there are times when progress as a whole should be reviewed. For this reason, we have indicated the desirability of two briefings for the exchange of technical information to allow for such an overall assessment of progress. One such briefing is suggested for November 1960, at which time a preliminary research progress report would be arranged. Such a meeting would assist in focusing attention to the pertinent problem areas and might well reveal additional problem areas that have not yet come to light. It is believed desirable also, perhaps in the Fall of 1961, to arrange for a NASA-Industry conference similar to the one recently held at Langley relative to Dyna-Soar problems. The purpose of this conference would be to lay the foundation for the preparation of final specifications for the multi-manned vehicle prior to extending an invitation for industry to bid on such a vehicle. The presence of industry at such a conference would also assure having them incorporate the latest thinking of NASA in their proposals which we would hope to be in a position to evaluate by July 1962. Contractor go-ahead under this plan may be feasible as early as August 1962.

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The bar chart in figure 3 is intended to give a general idea of how we anticipate the various programs following Mercury to fit into the overall planning. Mercury itself will probably continue through 1962. The phasing out of the dark areas of the chart are intended to indicate the successful completion of the basic Mercury mission. If we can skip for the moment the lifting Mercury bar and focus our attention on the third line identified as the multimanned vehicle, we will observe that the research and development activity extends from the present time to approximately the next 5 years. The ticks for the bidder's conference correspond to those previously discussed. If everything goes according to schedule, however, it may be possible to receive some delivery on initial hardware for the multimanned vehicle as early as the beginning of 1965. We are of the opinion at this time that earth orbit qualifications for multimanned vehicle is essential. The period of this activity, of course, will be dependent upon the problems we encounter; but in full recognition of the fact that the radiation hazard may prohibit early completion of the lunar mission, we have allowed for the possibility of rather extensive earth-orbiting flights to supply the experience and information that may well be essential to the formulation of more definite plans for the lunar mission. About all one can say at this time, is that the reentry vehicle or component of the multimanned system should be designed with the full capability of reentering with the hypervelocity required for the lunar mission.

In the time period between the phasing out of the Mercury and the introduction of flight tests of a multimanned vehicle, it is attractive to consider the possibility of a flight-test program involving the reentry unit of the multimanned vehicle which at times we have thought of as a lifting Mercury. We, frankly, do not know what form this reentry vehicle should have. Only further studies will crystallize concepts in this regard. We feel rather certain it should be a vehicle capable of some lift in order to make use of the maneuvering capabilities that lift affords. We have envisioned the use of a Mercury capsule designed to provide lift capability as a full-scale flight vehicle allowing us to gain considerable experience in guidance problems in this interim time period. The adoption of an interim program of this type, however, is a major undertaking for NASA, and we feel that we are not yet ready to finally determine whether we should attempt this step until more information is available upon which to base a decision.

The general concept of the lifting reentry vehicle, be it a Mercury capsule or a basically new reentry module of the multimanned vehicle, is to equip the capsule with the reentry control navigation system that would permit the vehicle to maneuver at least 500 miles from the impact point that would correspond to a ballistic trajectory. As noted in figure 4, two or three ground stations would provide command inputs to update the guidance system. The capsule would receive intelligence just

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before the retrofiring period and at least in the experimental stage prior to the onset of high acceleration forces during the reentry plus some terminal guidance during the subsonic portion of the flight. At the present time, there is some hope that this kind of experiment could be performed with a Mercury capsule modified in such a manner as to provide the capability of providing a trimmed  $L/D$  other than zero. However, none of the problems have really been investigated to the degree required for a rational decision. We have proposed, therefore, that the problems presented by the lifting Mercury be studied in the wind tunnels in order to provide information on the problem areas described in figure 5.

The type of flap suitable for a lifting reentry vehicle is in itself something that will be established only from research programs. The type of flap shown in figure 5 is for illustrative purposes only and does not represent any preconceived notion as to the ultimate geometry that a lifting flap might assume. Whatever flap develops, however, must operate over a substantial Mach number range, and careful coordination of the interesting flap arrangements must be made in order to assure adequate coverage. The method of control is still under study but could very well be aerodynamic in nature. On figure 6, there are indicated some of the NASA facilities that could well be utilized in investigating the adequacy of a controlled flap for a lifting Mercury or a lifting reentry vehicle for the multimanned lunar program. Most of the basic wind-tunnel models required for the tests that are shown in figure 6 are available from the Mercury program except for the incorporation of an aerodynamic flap. Many of the models are at the Space Task Group and can be made available at any time.

Having described our thoughts on the multimanned vehicular mission, we earnestly solicit your suggestions and proposals as to how best this effort can be carried out in the best interest of the NASA organization with its many centers and somewhat overlapping interests. We would hope in the immediate future to obtain your views as to the problems that each Center may concentrate on so that the whole NASA effort can be integrated as soon as possible.

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## MULTIMANNED SPACE FLIGHT PROBLEM AREAS

| MISSION             | VEHICLE              | OPERATION            |
|---------------------|----------------------|----------------------|
| SYSTEM CONCEPT      | ESCAPE SYSTEM        | BOOSTER REQUIREMENTS |
| NAVIGATION-GUID.    | HEATING PROTECTION   | ABORT SENSING        |
| GROUND REQUIREMENTS | RADIATION PROTECTION | LAUNCH REQUIREMENTS  |
| REENTRY TECHNIQUE   | NOISE                | COMMUNICATIONS       |
| ABORT PATHS         | STRUCTURE            | TRACKING             |
| HUMAN FACTORS       | AUXILIARY POWER      | RECOVERY             |
|                     | LANDING SYSTEM       | TRAINING             |
|                     | INSTRUMENTATION      |                      |

Figure 1

## ADVANCED MANNED SPACE PROGRAM

|          |      |                                                                     |
|----------|------|---------------------------------------------------------------------|
| APRIL    | 1960 | BRIEF NASA CENTERS                                                  |
| JULY     | 1960 | BRIEF INDUSTRY                                                      |
| NOVEMBER | 1960 | NASA INTERNAL BRIEFING<br>(PRELIMINARY RESEARCH<br>PROGRESS REPORT) |
| OCTOBER  | 1961 | NASA-INDUSTRY CONFERENCE                                            |
| MARCH    | 1962 | SPECIFICATIONS COMPLETE-<br>INDUSTRY BRIEFING                       |
| JULY     | 1962 | INDUSTRY PROPOSALS RECEIVED                                         |
| AUGUST   | 1962 | CONTRACTOR GO AHEAD                                                 |

Figure 2

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## MANNED SPACE FLIGHT PROGRAM

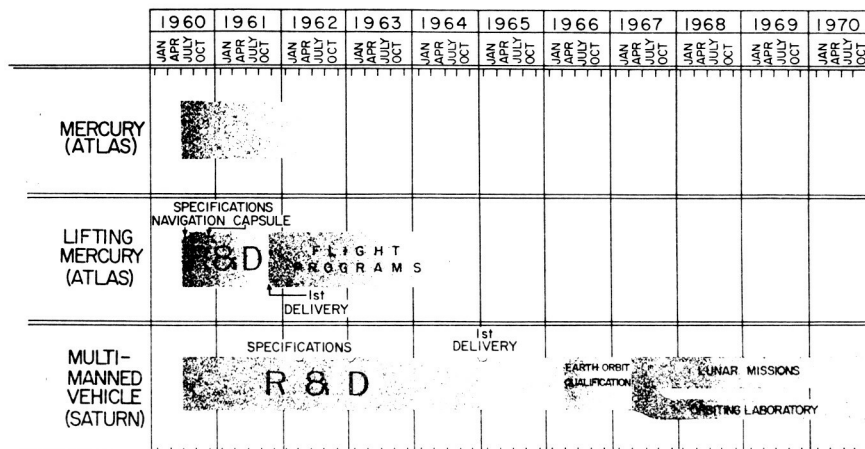


Figure 3

## LIFTING MERCURY

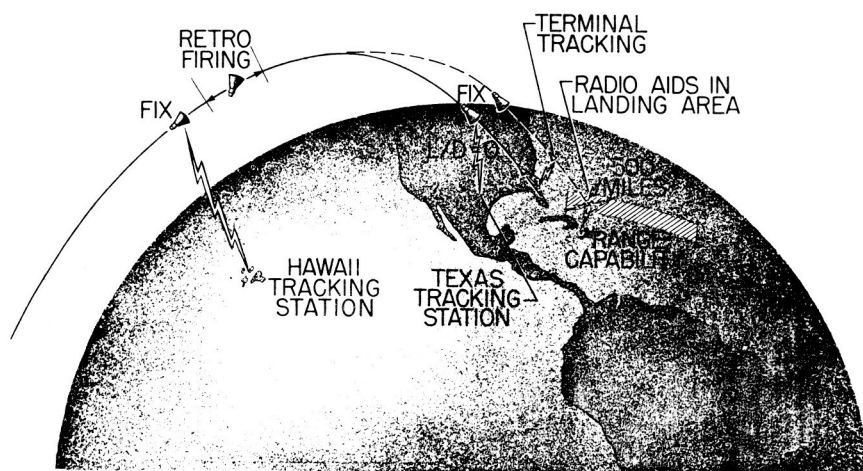


Figure 4

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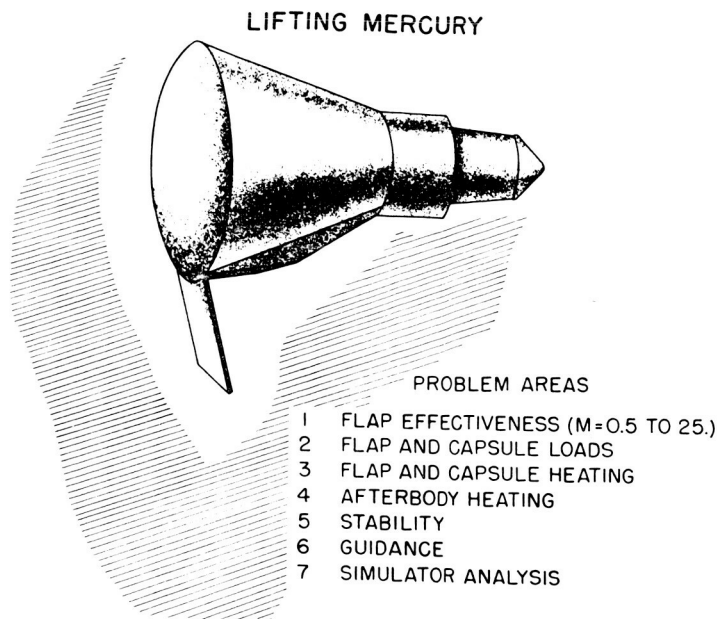


Figure 5.

## WIND TUNNEL STUDIES

| FACILITY                   | MACH NUMBER RANGE | STATIC STABILITY | DYNAMIC STABILITY | PRESSURE | HEAT TRANSFER |
|----------------------------|-------------------|------------------|-------------------|----------|---------------|
| LANGLEY FREE-FLIGHT TUNNEL | 0                 | x                | x                 |          |               |
| LANGLEY 8-FOOT TUNNEL      | 0.50 TO 1.14      | x                | x                 | x        |               |
| AEDC 16-FOOT PWT           | 0.50 TO 1.60      | x                | x                 | x        |               |
| AMES UNITARY TUNNELS       | 0.50 TO 3.50      | x                |                   | x        | x             |
| LANGLEY UNITARY TUNNELS    | 1.6 TO 4          | x                | x                 | x        | x             |
| AMES FREE-FLIGHT TUNNEL    | 3.00; 9.00; 14.00 |                  | x                 |          |               |
| AEDC B-2 TUNNEL            | 8.00              | x                |                   | x        | x             |
| LANGLEY 11-INCH TUNNEL     | 9.60              | x                |                   | x        | x             |
| ROCKET MODELS              | 15.00             | x                |                   | x        |               |
| 1-FT SHOCK TUNNEL          | 10                |                  |                   |          | x             |
| 1-FT HELIUM TUNNEL         | 8.0 TO 26.0       |                  |                   | x        |               |

Figure 6.

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APPENDIX I - GUIDELINES

I-A Lunar Mission.-

1. Guideline - The vehicle should be capable ultimately of manned lunar reconnaissance.
2. Justification - The mission is a step toward landing man on the moon and other planets.
3. Discussion -
  - (a) The trajectory of the vehicle must be such that reconnaissance of the moon can be effected in sufficient detail to acquire or verify information for selection of lunar-landing sites and design of lunar-landing vehicles.
  - (b) Close lunar orbit may be required for satisfactory reconnaissance.
  - (c) Instrumentation capabilities will dictate what is required in the way of "closeness" to the moon and "time" near the moon.

I-B Orbital Mission.-

1. Guideline - The lunar vehicle should be capable of earth-orbital missions for initial evaluation and training. The reentry component of this vehicle should be capable of earth-orbital missions in conjunction with space laboratories or space stations.
2. Justification - The reentry vehicle should be designed for use with both the lunar vehicle and the orbiting laboratory to avoid unnecessary duplication of vehicle development.
3. Discussion - It is assumed that the vehicle will consist of a "reentry component" and a "mission component" without reentry capability. The mission component for the basic vehicle should be designed for the circumlunar mission and the total vehicle should not weigh more than 15,000 pounds. However, the vehicle should be adaptable to an alternative mission component which will allow it to function as an earth-orbiting laboratory. For this corollary earth-orbiting mission, the total weight of the reentry vehicle plus mission component can be increased to the order of 25,000 pounds.

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I-C Propulsion and Weight.-

1. Guideline - The multiman advanced space vehicle should be designed to be compatible with the Saturn and for the lunar mission it should weigh not more than 15,000 pounds including auxiliary propulsion and attaching structure.

2. Justification -

(a) Saturn will be the only propulsion system with sufficient payload capability available in the near future.

(b) The 3-stage Saturn C-2 payload capability for the lunar mission is estimated to be not more than 15,000 pounds.

3. Discussion -

None

I-D Flight Time Capability.-

1. Guideline - The vehicle should be designed for a flight time capability without resupply of 14 days.

2. Justification -

(a) Minimum time for lunar circumnavigation is on the order of 6 days.

(b) Total flight time may be appreciably affected by lunar perturbations, ensuing midcourse corrective actions, loiter time, or choice of nonminimum trajectories.

3. Discussion -

None

I-E Maneuvering in Space.-

1. Guideline - Adequate auxiliary propulsion should be provided for guidance maneuvers required to carry out the assigned missions and for safe return in event of launch emergencies.

2. Justification - Self-evident from mission requirements.

3. Discussion -

(a) Guidance accuracies and capabilities should be studied in order to determine auxiliary propulsion requirements.

(b) Sufficient reserve propulsion should be included to accommodate corrections for maximum guidance errors.

(c) A single system may suffice for both midcourse and escape propulsion requirements.

II-A Mission Abort Capability.-

1. Guideline - The vehicle should have the capability of safe crew recovery from aborted missions at any speed up to maximum velocity, independent of use of the launch propulsion system.

2. Justification - The above requirement is basic to manned operations.

3. Discussion -

(a) The vehicle should be provided with some manner of abort capability throughout the launch phase.

(b) Safe control of the vehicle subsequent to aborting the mission is required, i.e.: auxiliary rocket power (maneuvering rockets) and the aerodynamic L/D capabilities of the configuration must be adequate for this condition as well as the normal mission.

II-B Ground and Water Landing Capability.-

1. Guideline - The vehicle should be capable of a satisfactory landing on both land and water and should have the capability of avoiding local hazards.

2. Justification -

(a) Emergency abort conditions may force landing on either land or water.

(b) Reliability of guidance, control and propulsion systems on lengthy missions is unknown and may cause unexpected error in earth reentry coordinates.

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(c) Accessibility for recovery is an important consideration. Relative superiority of land-versus-water landing will vary with locale.

3. Discussion -

(a) Vehicle structure should be capable of adequate attenuation of ground- and water-landing loads in a 30-knot wind.

(b) Vehicle should be watertight and have adequate sea-keeping qualities under sea-state four conditions (10 - 12-foot waves).

II-C Point Return.-

1. Guideline - The vehicle should have the normal capability of landing at one of several previously-designated ground surface locations on the earth approximately 10 square miles in area.

2. Justification - This capability is needed to provide for convenient return of vehicle and crew.

3. Discussion -

(a) Studies are needed to assess the value of impulse maneuvers, guidance quality, and aerodynamic L/D for the return from lunar mission.

(b) This requirement is far less severe for the orbital mission.

II-D Postlanding Survival Period.-

1. Guideline - The vehicle should be designed for crew survival for at least 72 hours after landing.

2. Justification - Due to the random nature of possible emergency maneuvers, it will be impossible to provide sufficient recovery forces to cover all possible landing locations. The above requirement would permit mobilization of normally existing facilities in adequate time for a safe recovery.

3. Discussion - The vehicle should be equipped with adequate location devices.

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III-A Crew Environmental Factors.-

1. Guideline - The crew accommodations must provide for a safe and comfortable "shirt sleeve" crew environment for the entire mission.

2. Justification - The mission is predicated upon the efficient operation and safe return of the crew.

3. Discussion -

(a) A sealed cabin with decompression protection should be provided.

(b) The long duration of the flight precludes the continuous use of a pressure suit.

(c) An environmental control system capable of meeting the crew's load and flight duration must be developed.

(d) Accelerations imposed on the crew should not exceed the physiological limits in keeping with the support and restraint systems provided.

(e) Noise and vibration levels should be maintained within the tolerable limits.

(f) Provision for recreation and exercise may be required to maintain crew efficiency.

(g) The long confinement period may require provisions for individual privacy.

(h) Provisions for nutrition, personal hygiene, and waste disposal, should be provided.

III-B Crew Size.-

1. Guideline - The minimum crew should consist of at least three men.

2. Justification -

(a) Continuous duties and communication make it desirable that at least one man be on duty at all times.

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(b) On duty - off duty cycle tests indicate that three men may be the necessary minimum.

(c) Redundancy in capabilities for backup purposes is essential to insure a degree of success in the event of disablement of personnel.

3. Discussion -

None.

III-C Adequate Protection from Radiation.-

1. Guideline - The mission should be accomplished without subjecting the crew to more than a permissably safe radiation dose.

2. Justification -

(a) The possible hazards to the health and well-being of the crew must be commensurate with or less than the other risks inherent in an adventure of this nature.

(b) High dosage could incapacitate the crew for the proper performance of their duties.

3. Discussion -

(a) Present knowledge indicates that 25 REM is probably an upper limit.

(b) Studies of minimum radiation trajectories are needed.

(c) Maximum utilization of structural materials and on-board equipment should be studied for minimization of radiation dosage to crew.

(d) Detailed information regarding the radiation environment around the earth and in cislunar space is needed.

(e) Present knowledge of solar radiation, coupled with the payload capability of chemical propulsion systems currently under development, indicates that sufficient shielding cannot be provided for the crew in the advent of a solar flare.

(f) A solar-flare prediction method is needed.

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IV-A Onboard Command.-

1. Guideline - The primary command of the mission should be onboard.
2. Justification - System simplicity and mission reliability will result from the maximum utilization of the command decision capabilities of the crewman.
3. Discussion -
  - (a) Automatic control will be employed to obtain precision or speed of response otherwise unattainable, but monitoring by the crew of each control function with provision for override will be required.
  - (b) The navigation computer must not only solve the problems of insertion, midcourse, and terminal guidance, but also be capable of performing calculations to provide the crew the information that is required to assess or respond to emergency situations.
  - (c) Systems and simulation studies will be required to evolve computer requirements, navigation and guidance techniques, attitude control requirements, and display requirements.

IV-B Communication and Tracking Facilities.-

1. Guideline - Communication and ground tracking should be provided throughout the lunar mission except when the vehicle is blanketed by the moon.
2. Justification -
  - (a) Communications and tracking are the means of keeping the ground crew informed of the mission status.
  - (b) Information for trajectory control will be obtained by ground tracking and relayed to the vehicle.
3. Discussion -
  - (a) Detailed studies are required to determine the most efficient and reliable means of maintaining communications and tracking.
  - (b) Contact at least once during each orbit will probably be adequate for the orbital mission.

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APPENDIX II - PROBLEM AREAS FOR ABORT, REENTRY, AND LANDING

1. A study of launch trajectories is needed to determine:
  - (a) Velocity changes necessary to safely abort the mission during launch.
  - (b) Design criteria for heating and air loads associated with various emergency reentry situations.
2. A study of the reentry following return from the lunar mission is needed to:
  - (a) Establish navigation accuracy required during the terminal adjustment and reentry maneuvers.
  - (b) Determine auxiliary propulsion requirements during this phase.
  - (c) Establish L/D requirements for corridor maneuvers.
  - (d) Determine design criteria for heating and air loads associated with the corridor boundary conditions.
  - (e) Determine the desired amount of aerodynamic L/D needed during the postblackout period.
3. Additional research must be done to extend aerodynamic heating theory to escape velocity and to establish the magnitude of shock-wave radiation.
4. Continue development of high-temperature structures and materials. Establish ablation effectiveness of suitable materials for heating situations to be encountered.
5. Continue study of promising configurations until best reentry vehicle can be established. Study should include among other things:
  - (a) Parametric wind-tunnel studies where practical.
  - (b) Single-unit and caboose-type vehicles.
  - (c) Practical method of escape during boost within atmosphere.
6. Extend evaluation of subsonic L/D and wing-loading criteria for glider landing to include rough water conditions.

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7. Investigate and establish design criteria for large parachutes and parachute clusters. Note that parachute systems with translation capability may be desired.

8. Continue investigation of impact bags and other impact attenuation devices.

9. Establish design criteria for auxiliary propulsion systems. Note -this will probably be different for solid- and liquid-rocket systems. Institute the development of suitable rocket systems in accordance with established requirements.

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APPENDIX III - PROBLEM AREAS FOR MISSIONS, PROPULSION, AND FLIGHT TIMES

1. A study of "free return" and, therefore, relatively "safe" trajectories which are also acceptable from the standpoint of earth reentry conditions.

2. A study of the effect of the moon's shape on:

(a) Return from a circumlunar pass

(b) Lunar orbits

(c) Return from a lunar orbit.

3. A study of the stability of lunar orbits at various altitudes above the moon's surface. Reconnaissance objectives will probably require close lunar orbit, at least on later missions prior to a landing mission.

4. A study of the use of the available auxiliary propulsion to return to earth from an emergency situation during the midcourse phase of the flight.

5. A detailed study needs to be made to determine the trade-offs between using a one- or two-component lunar vehicle. Such a study should consider the optimum distribution of systems and supplies between the two components.

6. A detailed study is required to determine the most desirable type of system for providing the required electrical power.

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## PROPORTIONING THE AIRPLANE FOR LATERAL STABILITY

By Charles J. Donlan

In proportioning the airplane for lateral stability, the two characteristics of most importance are the directional stability factor  $C_{n\beta}$  defined as the slope of the curve of yawing-moment coefficient against sideslip, and the effective dihedral factor  $C_{l\beta}$  defined as the slope of the curve of rolling-moment coefficient against sideslip. The advantages of combining high values of  $C_{n\beta}$  and low values of  $C_{l\beta}$  have been amply demonstrated. The design problem is to provide the optimum ratio of  $C_{n\beta}$  to  $C_{l\beta}$  and to determine how both of these parameters and their ratio are apt to change for the various flight conditions encountered. The purpose of this brief paper is to summarize the design factors that influence these parameters and to indicate how reliably they can be estimated.

The basic forces influencing the directional stability of an airplane are indicated in figure 1. The forces and moments contributing to the yawing moment about the center of gravity are: (1) the propeller side force  $N$ , (2) the basic fuselage yawing moment  $N_F$ , and (3) the side force on the vertical tail  $L_T$ . We have found that side force developed by the propeller becomes important for large propellers located at some distance from the center of gravity. In any event, however, the propeller side force presents no serious design problem inasmuch as the variation of the propeller side force with angle of sideslip can be predicted very reliably and rapidly from design charts such as those presented in a paper by Ribner. (See reference 1.)

The unstable yawing moment of the fuselage also can be estimated fairly reliably either from experimental results on similar shaped bodies or by theories such as Multhopp's. (See reference 2.)

The side force developed by the vertical tail and its variation with angle of sideslip however cannot be estimated as reliably or as easily. The contribution of the vertical tail to directional stability is proportional to the slope of the tail-normal-force curve, which, in turn, is primarily a function of the aspect ratio of the tail. It must be remembered, however, that the horizontal tail and fuselage act as an endplate which may cause the effective aspect ratio of the vertical tail to differ from its geometric value by 50 percent or more. Consequently, it is rather important to estimate how large the endplate effect may be. Methods for doing this are discussed extensively in papers by Pass (reference 3) and Murray (reference 4).

The effectiveness of the vertical tail is also influenced by the sidewash associated with the flow about the fuselage-wing combination. We have found that the sidewash is generally favorable for low-wing airplanes and adverse for high-wing airplanes. The sidewash effect is important and should be taken into account in estimating the directional stability. Methods of estimating this effect together with the complications introduced by flaps and power are discussed extensively in several of the papers now available to you.

In addition to supplying adequate stability at small angles of sideslip, we have frequently found it necessary to increase the effectiveness of the vertical tail at large angles of sideslip. One device commonly employed for this purpose is the dorsal fin. The typical effects produced by a dorsal fin are illustrated in figure 2. In this figure the yawing-moment coefficient  $C_n$  is plotted as a function of the sideslip angle  $\beta$  for an airplane with a vertical tail used in conjunction with a dorsal fin. You will notice that whereas the vertical tail without the dorsal fin becomes ineffective at an angle of sideslip of about  $15^\circ$ , the vertical tail in combination with the dorsal fin retains its effectiveness out to  $30^\circ$  without any detrimental effects at small angles of sideslip. The asymmetry of the curves is caused by the propeller slipstream and is typical of the effects introduced by power.

The dihedral effect exhibited by airplanes is, of course, closely associated with wing position, but fairly reliable estimates can be made of the effect. In general, high-wing arrangements exhibit greater dihedral effect than low-wing arrangements. The magnitude of the interference effect is illustrated in figure 3. These results were taken from reference 5 and are for an aspect-ratio 6 wing with no geometric dihedral. The ordinate on the left is the effective dihedral parameter  $C_{lp}$  and the equivalent geometric dihedral angle is given on the right. The relative wing position is represented along the horizontal axis. It will be noted that the high-wing position is equivalent to about  $5^\circ$  of effective dihedral and the low-wing position about  $-5^\circ$ . The fuselage cross-sectional shape appears to be of minor importance.

Any dihedral effect introduced by setting the wing at a given geometric dihedral angle can be added to the basic arrangement. We have found that with low-wing arrangements that the dihedral effect is reduced when power is applied, but the effect depends to a considerable extent on the amount of wing immersed in the slipstream. For high-powered airplanes, the power effect on dihedral can be extremely pronounced when flaps are immersed

in the slipstream and many service types exhibit negative dihedral effect in the full-power, flap down condition. For low-powered airplanes, the effect is probably unimportant.

### SYMBOLS

|              |                                                                                                 |
|--------------|-------------------------------------------------------------------------------------------------|
| $C_{n\beta}$ | slope of curve of yawing-moment coefficient against sideslip                                    |
| $C_{l\beta}$ | slope of curve of rolling-moment coefficient against sideslip                                   |
| $C_n$        | yawing-moment coefficient ( $N/qbS$ )                                                           |
| $C_l$        | rolling-moment coefficient ( $L/qbS$ )                                                          |
| $N$          | propeller side force, pounds: also, yawing moment with respect to stability Z axis, foot-pounds |
| $L$          | rolling-moment coefficient with respect to stability X axis, foot-pounds                        |
| $N_F$        | basic fuselage yawing moment, foot-pounds                                                       |
| $L_T$        | side force on vertical tail, pounds                                                             |
| $Y$          | total side force, pounds                                                                        |
| $T$          | thrust (See fig. 1.)                                                                            |
| $L_W$        | wing lift, pounds                                                                               |
| $b$          | wing span, feet                                                                                 |
| $S$          | wing area, square feet                                                                          |
| $q$          | dynamic pressure, pounds per square foot                                                        |
| $\beta$      | sideslip angle, degrees                                                                         |
| $\phi$       | angle of bank, degrees                                                                          |

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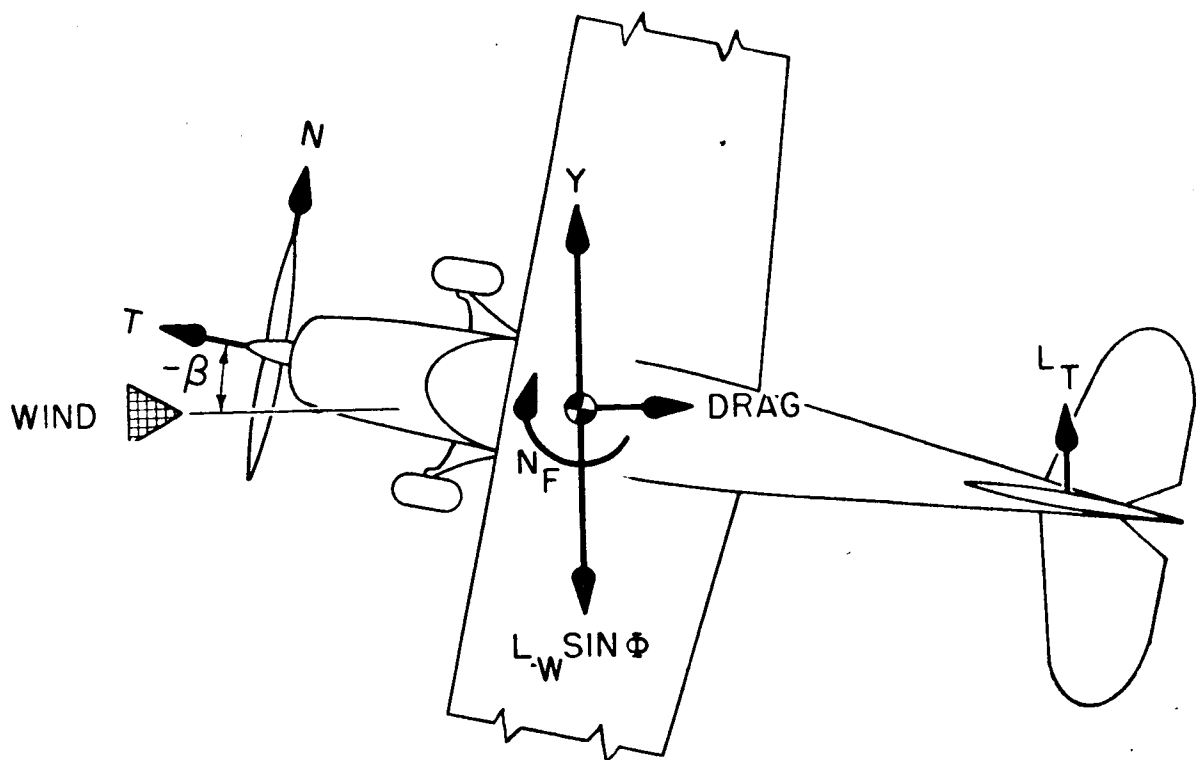


Figure 1.- Basic forces on an airplane in a sideslip.

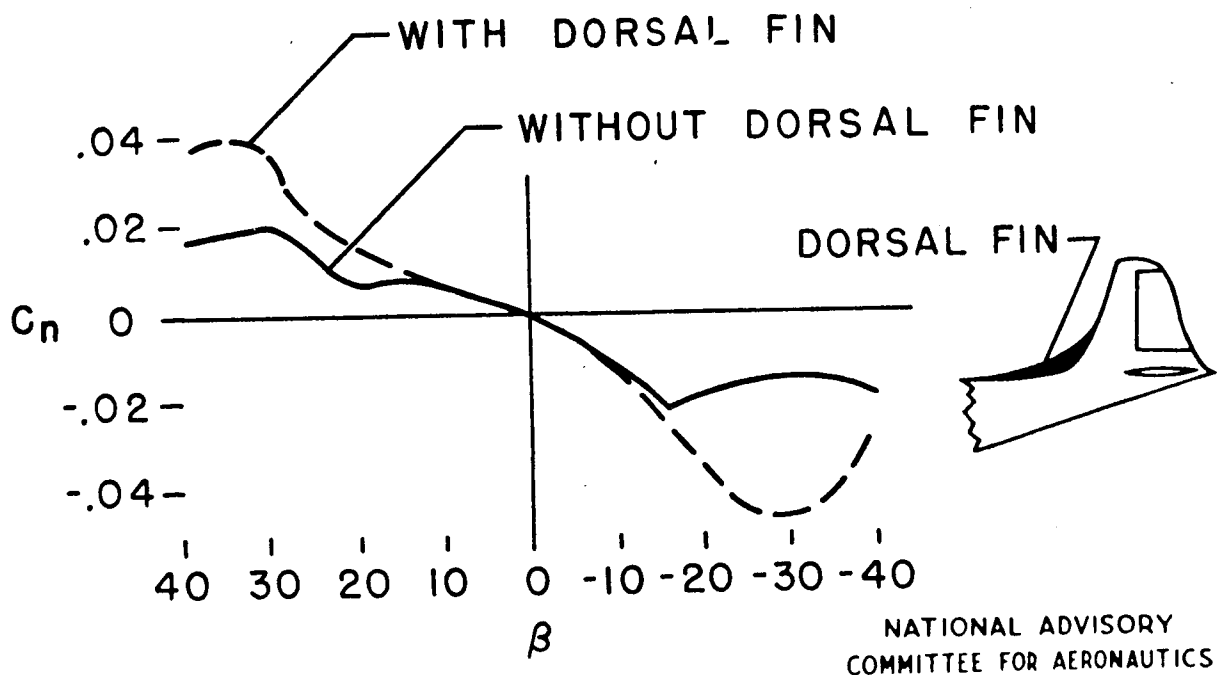


Figure 2.- Effect of dorsal fin on directional stability.

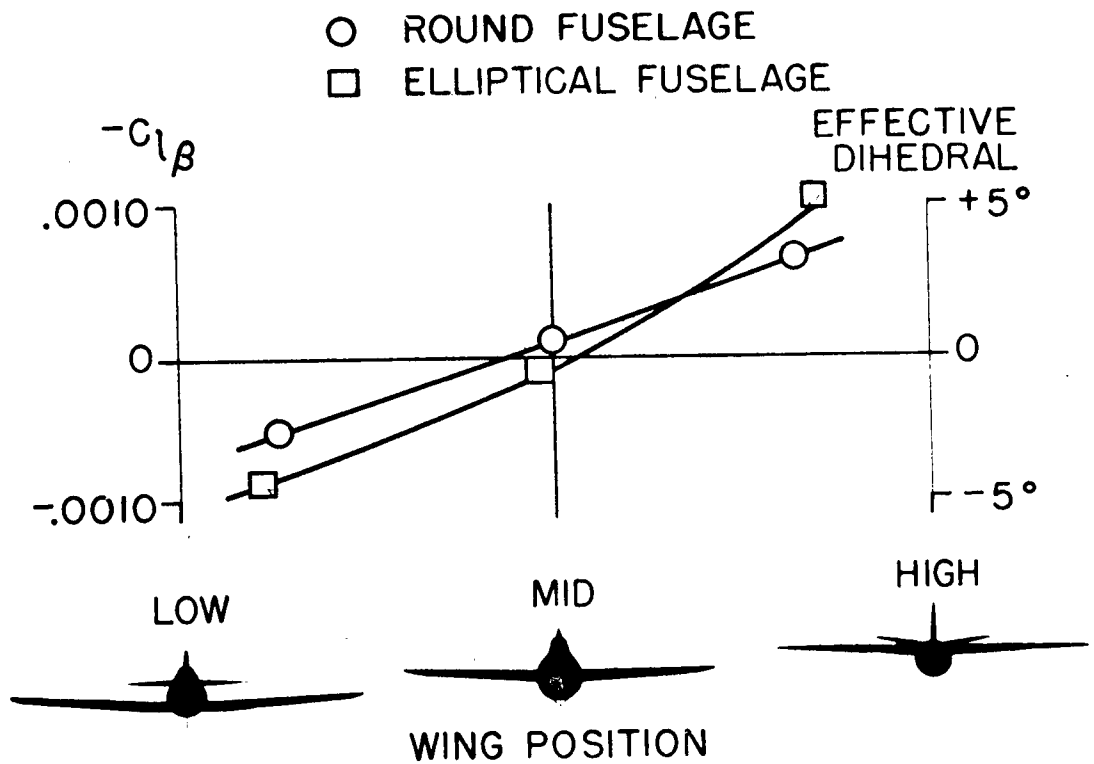


Figure 3.- Effect of wing position on dihedral effect.

NATIONAL ADVISORY  
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# Progress report on Project Mercury

Reviewed in the light of two-years' experience, the nation's man-in-space project shows both stability of aim and approach and a maturing engineering form

By Charles J. Donlan and Jack C. Heberlig

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Donlan

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Jack C. Heberlig is technical assistant, Office of Project Director, NASA Space Task Group, and was previously a Mercury-Redstone project engineer in the Space Task Group Flight Systems Div. Before being assigned to the Space Task Group when it was formed in October 1958, Heberlig was with the NASA Langley Research Center's Pilotless Aircraft Research Div., where he was active in basic research for manned space vehicles during 1957 and 1958.

Presented at the Psychological Aspects of Space Flight Symposium, sponsored by the AF School of Aviation Medicine and arranged by the Southwest Research Institute, this paper will also appear in a Proceedings to be published by the Columbia Univ. Press.

**P**ROJECT Mercury, our country's program for manned orbital space flight, enters the more dramatic stages of flight testing this fall. With major flights in the offing, we would like to review Mercury concepts and hardware in the light of our development experience thus far.

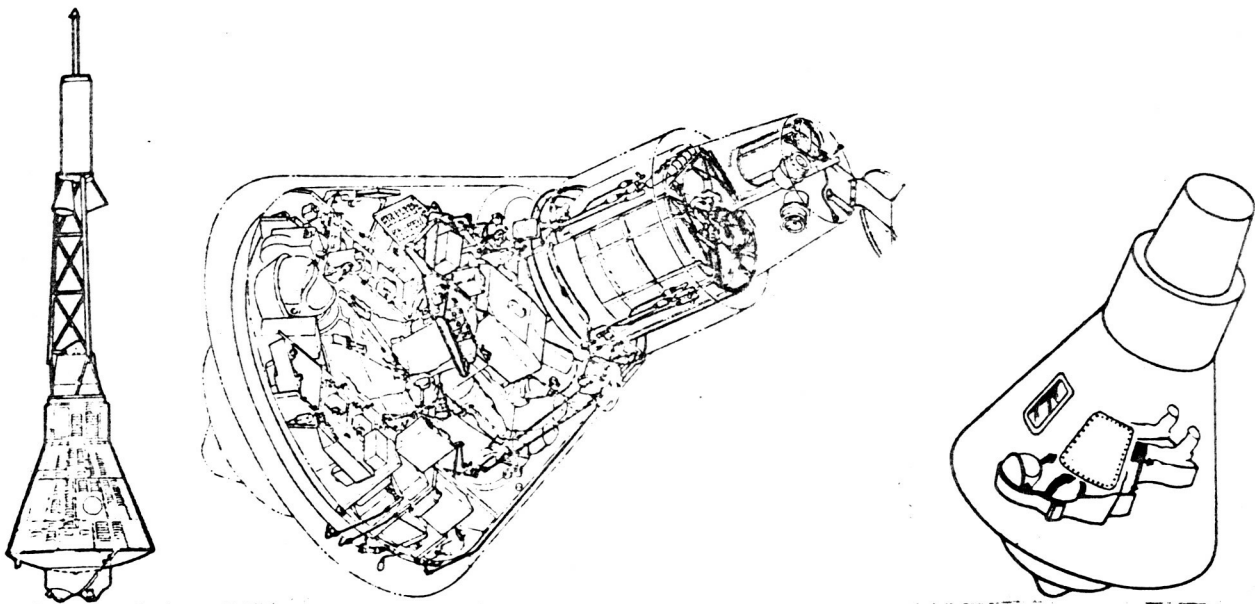
The original aims and concepts for Project Mercury, announced when it received official status Oct. 5, 1958, still guide the program. The main aims of Project Mercury are to orbit and recover a man in a space capsule safely and to study man's capabilities in space. The Mercury concept calls for simplicity and reliability, a minimum of new developments, and a progressive build-up of tests to demonstrate reliability. The basic engineering approach has been a drag vehicle boosted by an available ICBM propulsion unit; retrorocket start of descent from orbit; parachute braking of the final phase of re-entry; and an escape system for the launching and early-flight phases. The present state of space-vehicle technology indicates that the decision to base work on a high-drag shape for ballistic re-entry was a wise one.

The illustration on page 33 shows the Project Mercury capsule with its escape system. The capsule has a maximum diameter of approximately 74 in. With escape rig, it is about 25 ft high. The heat shield is an ablating phenolic resin. There are three rockets in the retro unit, mounted on the heat shield. The capsule's afterbody has a sheath of cobalt-alloy shingles that can expand in all directions to minimize stress from high heating rates.

The cylinder atop the capsule proper contains the parachute recovery system. This system continues to be tested extensively to insure high reliability. The capsule, moreover, will have a reserve chute to back up the primary one.

The escape rig, tripod-mounted above the capsule during launch, can remove the capsule from the booster at any time. Tests of this system simulating an off-the-pad abort, an abort at maximum dynamic pressure on exit, and abort at very high altitude and velocity have all been successful. There have as yet been no booster malfunctions that would have prevented a safe





The illustration at left shows the capsule complete with retrorocket unit on heat shield and escape rig tripoded on top. Center, an illustration of the capsule's internal arrangement, with the Astronaut in flight position. The drawing at the right shows the position of the window and hatch added to the capsule at the suggestion of the Astronauts.

escape of the capsule from the scene of immediate danger.

The illustrations above depict the Astronaut in the capsule. The window and egress hatch are two modifications that have been made to the capsule since its initial design. The window, directly above the Astronaut's head, allows him to make observations independent of a periscope. The hatch employs primacord to cause it to fail and fall free when its handle is turned.

### Safety Built In

The capsule is complicated by the redundant systems which have been added for increased safety. From right to left, one can distinguish the antenna can, two horizon scanners, parachute package, pitch and yaw jets and associated plumbing, periscope housing, instrument panel, side-arm controller, and sundry electronic and power packages. The environmental-control system is mainly below the Astronaut's couch. The capsule is completely automatic, but provisions have also been made for piloted operation. Each system has a backup; and in the area of communications, there are several backups. There are two telemetry systems for sending back capsule and pilot data. Also, there are on-board records which collect the same information sent by telemetry.

The Astronaut may take over completely and perform all the functions of the automatic control

system. Because of the parallel manual system, he will also be able to fly any mode that might fail on the automatic system.

Let us look at a few of the capsule systems of special interest, beginning with the instrument panel, shown on page 73. The controls and displays on this panel are grouped as to function. The group on the left side has various pilot controls, such as those concerned with the attitude-control system and the retrorockets. The two large handles are for decompression and repressurization. Decompression would be used to extinguish a fire.

The next group, a sequencing display, consists of a series of lights—red and green—to indicate whether various functions did or did not occur at the proper time. The handle just to the left of each light is the pilot's control to override and correct the failure of any particular function. The light at the very top of this group is the abort light, which comes on if an abort is initiated. If the abort does not occur, the pilot can abort by using his left-hand grip. The switch immediately under the abort light is the pilot's "ready" switch. This switch is used prior to launch to light up a "ready" light on the test conductor's panel in the blockhouse.

The next series of dials are flight instrumentation. The dials indicate acceleration, rate of descent, altitude, and the fuel supply in the hydrogen-peroxide tanks. The top center of the panel is the attitude display, which shows rate and attitude in pitch, roll, and yaw. The Astronaut's control of attitude will be aided by (CONTINUED ON PAGE 73)

## Project Mercury Report

(CONTINUED FROM PAGE 33)

a periscope, which is centered below the instrument panel, and the window directly above the panel. The Astronaut will be able to see the horizon in back of the capsule through this window in the normal orbital attitude of 14 deg, as well as in the retrofiring attitude of 34 deg. The periscope provides the Astronaut with a view of the earth beneath the capsule. He can control the attitude by location of the earth's image in the periscope.

The instrument in the left center is a dead-reckoning device based on a small model of the earth, clock-driven to rotate in time with the orbit. In the right center is the main capsule clock, which, in addition to firing the retrorockets, indicates time of day, the lapsed time since launch, and time-to-go for retrorocket firing. The Astronaut also has an additional time-to-go dial, which he can set for any desired event, and a separate stop watch.

The instruments grouped in the upper right-hand corner of the panel follow the environmental-control system: Cabin pressure, temperature, relative humidity, and oxygen partial pressure; primary and emergency oxygen-supply pressure; and carbon-dioxide partial pressure downstream of the lithium hydroxide canister in the pressure-suit control system. This por-

tion of the panel also has controls for the cabin-pressure-system fans.

A light panel, adjacent to the environmental-control-system instruments, gives the Astronaut visual warning of major system failures. At the same time, an alarm sounds when a failure occurs. The Astronaut turns off the auditory warning signal by actuating a switch located by an associated light and then takes corrective action. Warning lights are provided for loss in cabin pressurization, depletion of primary oxygen supply, emergency-rate mode of operation, decrease in cabin oxygen partial pressure below 3 psi, increase of carbon-dioxide partial pressure to 3 per cent in the pressure-suit circuit, and excessive cooling water to the suit and cabin heat exchangers.

### Manual Backup Controls

A telelight panel on the left console gives the Astronaut indication of light-event sequential operation. The launching oxygen supply and snorkel operation are also shown on this panel. Manual backup controls are installed adjacent to the lights. Heat-exchanger water-flow controls and the emergency-rate valve control are located on the right console, which is not shown in the illustration below.

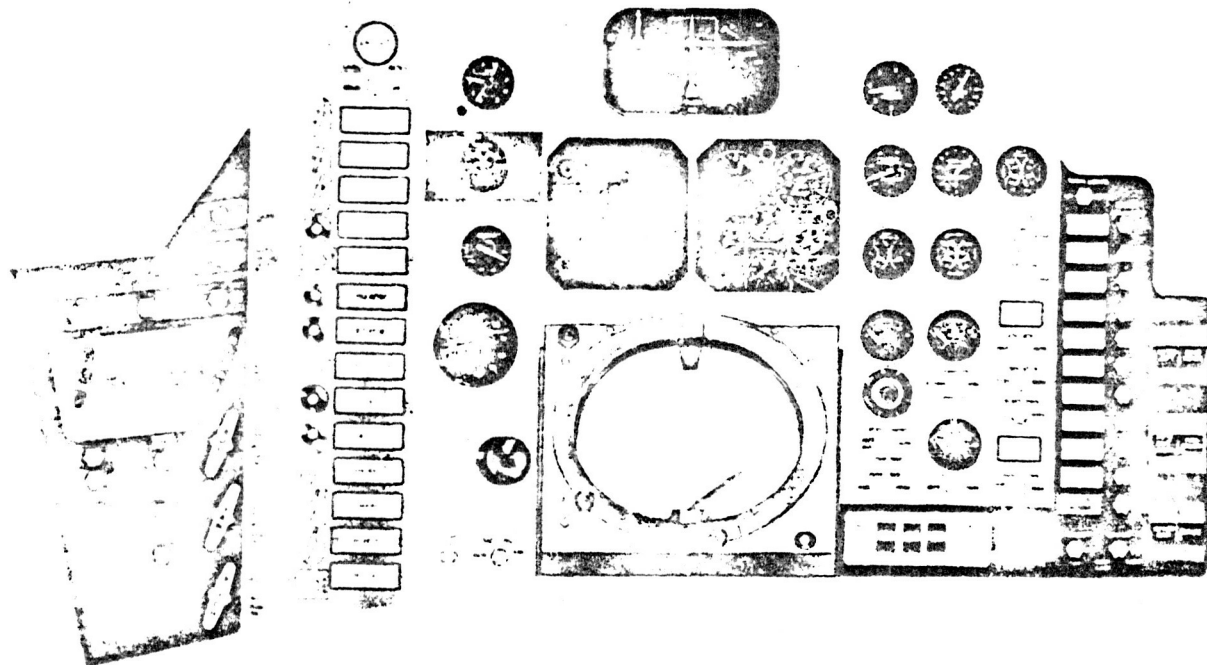
The instrument panel and Astronaut will be photographed in flight. In addition, instrument data will be telemetered and will be recorded on board. The Astronaut's electrocardio-

gram, body temperature, and respiration rate and depth will also be taken and recorded on board and telemetered continuously.

The environmental-control system, diagrammed on page 74, maintains the environment for the Astronaut at a temperature between 50 and 90 F. The Astronaut can manually select the temperature in this range that makes him most comfortable. During re-entry, when the cabin air temperature may approach 80 F, his suit will still maintain its air within the desired temperature range. The system is capable of operating for about 32 hr. Body odor is removed by activated charcoal, and carbon dioxide is removed by lithium hydroxide. Fresh oxygen enters the suit at the torso and exits at the helmet. As oxygen leaves the helmet, it enters a debris trap, where small particles are removed, and then goes through the odor and carbon-dioxide absorbers. A filter downstream of the carbon dioxide absorber prevents lithium-hydroxide dust from contaminating the oxygen. Pressure drop is counteracted by an increased supply of gas from the oxygen bottle. Ten cubic feet per minute is the flow rate through the suit at approximately 5 psi. Considering the fact that this system only weighs about 130 lb, it represents indeed a noteworthy engineering accomplishment.

Among the more complex systems in the capsule is the one for automatic stabilization and control. The illustra-

### Astronaut's Instrument Panel



tion below shows the normal operation of this system. When the Atlas booster engines drop, an event known as staging, the capsule ejects the escape rig by firing a special jettison rocket. The Atlas booster then follows a tilt program which yields an orbital altitude of approximately 105 n.mi. at the time of insertion. At booster cutoff, the capsule's "posigrade" rockets fire, separating the capsule from the booster. The stabilization system maintains capsule rate damping for 5 sec, until all oscillations of the capsule have been stopped. The capsule orients to the retrorocket firing attitude for 5 min. Within this period, ground tracking stations will determine if the velocity and insertion angle attained will allow the capsule to maintain itself in orbit.

### Emergencies Anticipated

The capsule then orients into the normal orbital attitude —  $14\frac{1}{2}$ -deg. blunt-end-up position. This permits the Astronaut, with the aid of the periscope, to see past the heat-shield end. Also, the retrorockets point in the right position for re-entering immediately, in case of emergency.

The automatic stabilization and control system next orients the capsule into the retrofiring attitude. After retrofiring, the capsule orients to the re-entry attitude, and the retopackage is jettisoned. As the capsule enters the atmosphere, the control system introduces a roll rate of 10 to 12 deg/sec and maintains rate damping, which prevents large capsule oscillations from

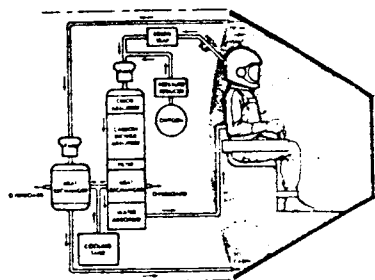
building up. At about 42,000-ft altitude, the drogue parachute deploys, stabilizing the capsule prior to deploying the main parachute at 10,000 ft. At water impact, there is released a small balloon carrying an antenna aloft to aid in capsule recovery.

One of the major concerns has been to protect the pilot from excessive accelerations during flight and landing. The escape rocket, if fired in an off-the-pad abort or at high altitude where the drag forces are low, would place upon the Astronaut an acceleration level of 20 g for a period of almost 1 sec. Impact on the water would give accelerations as high as 40 g for a few milliseconds.

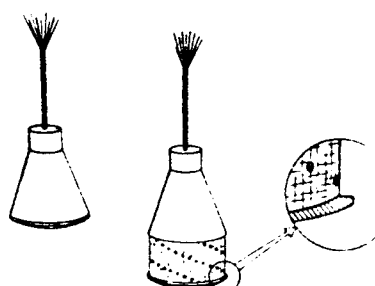
The contoured couch developed for the Astronaut evenly distributes his weight. Normal exit g-profiles run with the couch demonstrated its efficacy in helping the Astronaut do tasks which are necessary for normal capsule functions. Tests in which accelerations reached values as high as 25 g also demonstrated the validity of this support system in the case of escape-rocket firing.

A major problem still existed, however, in land impact or impact on the water with a high horizontal-wind profile. It is believed that a satisfactory solution to this problem has been accomplished by the accordion-like air cushion shown at right. The air cushion consists of a 4-ft skirt made of rubberized Fiberglas that connects the heat shield to the rest of the capsule. After the main parachute deploys, the heat shield releases from the capsule and the bag fills with air. Upon im-

### Project Mercury Environmental Control System



### Air Cushion for Impact Attenuation



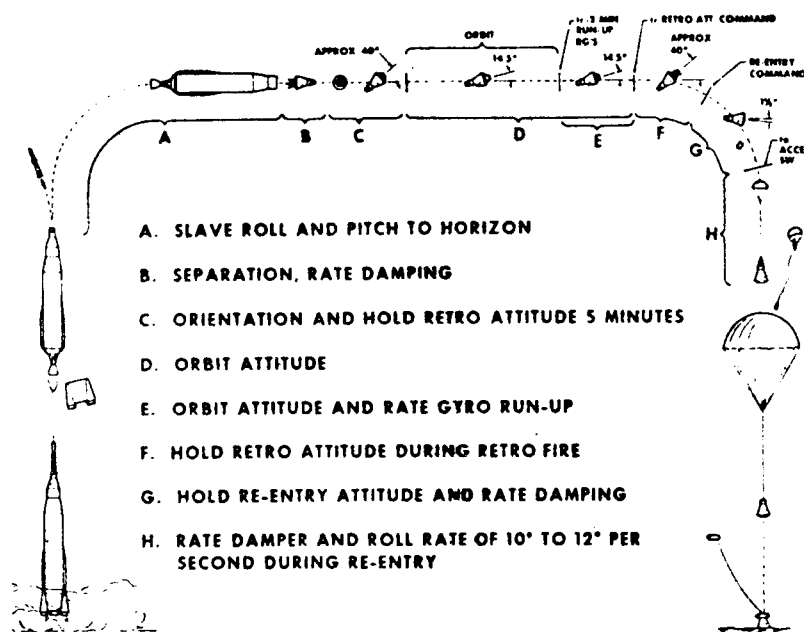
pact, the air trap between heat shield and capsule vents through holes in the skirt, as well as through portions of the capsule not completely airtight, giving a cushioning effect.

Now as to the booster for the Mercury capsule and the steps taken to assure its reliability.

The models pictured on page 75 show the three flight-test vehicles used in Project Mercury. The Little Joe is a cluster of solid-propellant rocket motors used in the research and development program. No manned flights are contemplated with this booster, although successful flights with small primates aboard the capsule have been made. The Redstone booster will be used for Astronaut training flights. Before this program begins, an animal will be flown to demonstrate the system. The Mercury capsule-Atlas combination will also carry animals before it puts a man in orbit.

The quality-control program being followed with these boosters takes advantage of all the experience gained in the weapon-system programs involving them. The reliability of each system is based upon the nominal performance of each individual component and subsystem. Also, as these systems are assembled in the various industrial plants, each part is being marked or tagged with a Project Mercury label. Although an assembly-line technique has been established for

### Normal Operation of the Automatic-Stabilization Control System

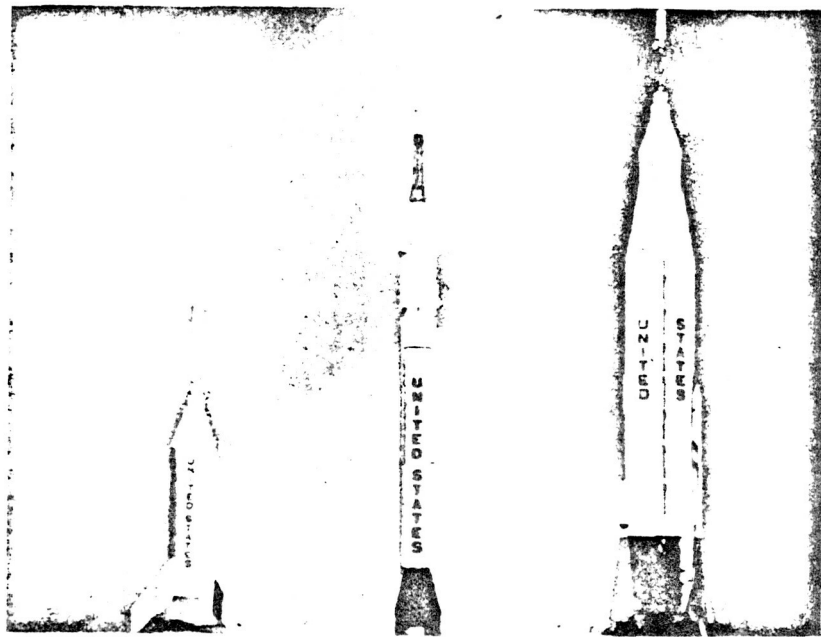


these boosters, 100-per cent inspection of all parts will take place to assure the highest possible quality control. Upon assembly of a booster system, a comprehensive rollout inspection will be made. The history of all parts going into a particular booster will be carefully documented and reviewed. In this manner, the operational proficiency of all systems and individual components should be made known.

Moreover, after a capsule has been mated with a booster and the mechanical and electrical integration completed, a flight-safety review board will be held. This review board will report on the over-all expected reliability of the capsule and booster together. Only after the capsule-booster combination meets the requirements of the flight-safety review board will approval for flight be given.

As to tracking, a total of 17 stations, including the control center at Cape Canaveral, will be active while the Mercury capsule is in orbit. Of these 17, two will be located on-board ship in the mid-Atlantic and Indian Oceans. At each tracking station there will be telemetry readout giving the personnel real-time indication of the pilot's and the capsule system's condition. Prior to each flight, aeromedical and capsule-system specialists will be sent to each of the monitoring stations; and after each flight these specialists will aid in evaluating performance. The down-range facilities of the Atlantic Missile Range will be used for the normal re-entry tracking.

## Models of Project Mercury Flight-Test Vehicles

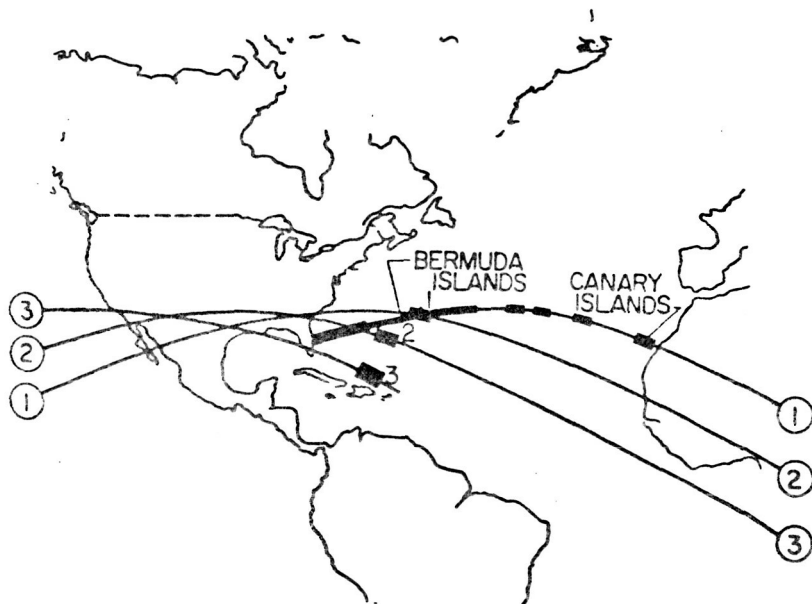


The capsule can be followed on radar from the time it begins its re-entry until impact. Constant voice communication should also be possible during this time. The tracking facilities in the continental U.S. will be used extensively for determining the correct time of retrofiring, and thus predicting impact area. In the U.S.,

there are six such tracking installations.

Certain areas, shown by the chart below left, have been designated as primary recovery areas. A long recovery area, from Cape Canaveral toward the Bermuda Islands, includes all impact points in case of an early abort. Between the Bermuda Islands and the Canary Islands are other primary recovery areas. These areas are so located that if velocity or insertion angle is not correct firing of the retro-rockets should cause the capsule to impact in them. If the initial orbit is not correct and conditions prevent the capsule from making three orbits, a second impact area will be aimed at toward the end of the second orbit. The impact area from a nominal flight, however, occurs at the end of the third orbit down the Atlantic Missile Range. As mentioned, this would mean continuous tracking of the capsule during re-entry from Hawaii until impact.

## Main Recovery Areas Planned for Mercury Capsule



No. 1—Impact areas possible in an early abort; No. 2—Areas possible with an imperfect attempt at orbit; and No. 3—Area for a normal three-orbit flight.

## Milestones

The Astronaut in Project Mercury will furnish other milestones in man's conquest of space.

The Mercury orbital missions will permit the study of the effects of prolonged weightlessness on the physiological reactions, the subjective psychological reactions, and the performance of the Astronaut. Of particular interest will be the effects of weightlessness on respiratory and circulatory systems. Telemetered electrocardio-

gram and breathing rates will make it possible to determine not only the effects of prolonged weightlessness itself, but the effects of transition from high accelerations to zero-g and back to the high accelerations associated with launch and re-entry. Problems associated with eating and drinking in prolonged weightless flight can also be studied. Stress hormone studies, together with pre- and post-physical examinations, will allow assessment of the Astronaut's physiological reaction to the spectrum of flight stresses.

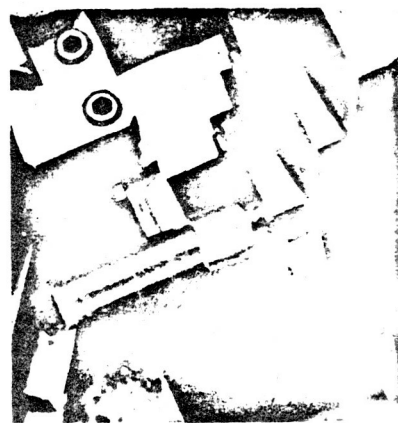
Subjective psychological phenomena likely to be affected by the weightless environment can be studied through voice reports and capsule instrumentation. The ability of the Astronaut to orient himself in space can be determined. The effects of the reduction in proprioceptive cues in time estimation can be studied. And attention will be given to the extent of autokinesis and oculogyric illusions possibly arising from weightlessness and other factors.

Of utmost practical importance to future manned space flights will be the quality of the Astronaut's performance in space. Accuracy in manual attitude control should provide a good test of his psychomotor performance. His monitoring of the capsule's orbital

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The availability of the Linde crystals should facilitate research on high-temperature materials for electronics and astronautics. George Hoffman of the Rand Corp. discusses the strength of pure crystal filaments in the Aug. 1958 *Astronautics*.



A large crystal of pure tungsten grown by Linde's arc-fusion process is shown being turned on a lathe at room temperature.

systems should provide an indication of his vigilance and perceptual accuracy. Navigation will test his reasoning and visual discrimination of earth terrain and heavenly bodies.

This is to say, we believe that Project Mercury will provide the base-lines by which future space vehicles can be built around the most valuable and useful machine—man himself. ♦♦

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EXPERIMENT PLANNING FOR MANNED PLANETARY MISSIONS

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## EXPERIMENT PLANNING FOR MANNED PLANETARY MISSIONS

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### Abstract

Planetary exploration provides an opportunity for the acquisition of extensive new knowledge in the physical and life sciences. In order to exploit this opportunity, scientific objectives must be identified and clearly understood. An experiment program leading to the attainment of these objectives must then be planned to take advantage of the various missions, systems, and modes of operation which may be available. This is accomplished by examination of the scientific objectives of planetary exploration, formulation of a planetary exploration rationale, definition of the roles and interdependence of manned and automated experiment systems, and definition of the necessary technology development requirements.

The preparation for manned planetary exploration will comprise automated missions and extensive research and development programs in earth orbit. The automated systems, while capable of significant independent contributions in the physical and life sciences, also contribute to a better definition of the role to be undertaken by manned systems. Initially, these contributions would comprise precursory definition of the interplanetary and planetary space environments to the extent necessary to assure the adequacy of the manned spacecraft design, and definition of the experiment requirements for the manned program to assure optimal scientific data collection by the scientist astronauts. During actual manned missions, unmanned systems would augment and expand the scientific return and would permit gathering of synoptic data and data from regions not readily accessible to the experimenters.

The earth orbit programs must be carried out to define biomedical and behavioral criteria for long duration manned space flight, to advance subsystems and sensor technology, and to establish onboard operational and experimentation procedures. In addition, techniques must be developed which might permit an intelligent trained experimenter to perform previously impracticable solar, planetary, and galactic observations from a position relatively remote from planetary influences.

In order to demonstrate the complex interrelationships between a manned planetary experiment program and the precursory earth orbit program, the definition of a broad and systematic manned planetary experiment program has been attempted and the major experiment systems and operational modes have been identified. The function of the manned

spacecraft as a versatile research facility is developed and similarities in onboard experiment programs for systems operating in earth orbit and on planetary missions are noted. A typical planetary mission is suggested with emphasis on the requirements for a continuous, dynamic experiment plane.

### Introduction

The primary objective of any manned mission to the planets will be to obtain scientific information contributing to our knowledge of the origin and evolution of the universe, of the galaxies which comprise it, and of the solar systems and life within it. Astronomy continues to make significant contributions and automated orbiters and landers can expand this knowledge even more. Clearly a manned planetary mission should not be undertaken solely to perform functions or execute programs which could be done more effectively or economically from earth-based or by automated systems. On the other hand, it would be proper for man to undertake special tasks that would be excessively demanding in cost and time, if undertaken by automated systems. Furthermore, automated systems are constrained to perform those functions for which the technology exists and for which the tasks can be defined in complete detail and are not so complex as to make the probability of mission success unattractive.

What then is the role of man in the planetary mission? The most important advantage of man will be found in his capability to act upon unforeseen phenomena and events. In addition, the ability to conduct experiments and to correlate results from many observations and measurements should afford the onboard experimenter an opportunity to alter mission plans, modify experiment procedures, and reorient measurement requirements to obtain particularly significant data on a timely basis. Thus man's major role in a planetary mission is similar to his role in a research laboratory on earth - it permits dynamic experiment programs in several scientific disciplines simultaneously.

It now becomes necessary to define the means by which man's unique capability as an experimenter in space can be exploited. One approach is to provide him a safe, versatile spacecraft which will support him on his long-duration, non-resuppliable mission and which will contain the proper onboard research facilities required by the experiment program. This suggests a logical sequence, which is to plan the experiment program; define the onboard research

facilities necessary to accomplish the program; integrate the requirements of the research facilities with the requirements to support man safely; and provide the necessary mission technology.

This paper will attempt to demonstrate this logic utilizing a Mars landing mission.

### Mission Analysis

A Mars opposition mission with a flight crew of six has been selected as the basis for this discussion simply because it contains all of the critical operational factors which have significant impact on experiment program planning.

A simplified mission events sequence is shown in Figure 1 and is as follows. The space vehicle which includes the propulsion modules, mission module, Mars excursion module, and Earth entry module, is assembled in earth orbit. Upon completion of the space vehicle assembly and final checkout, it is injected (1) into a heliocentric orbit targeted to Mars by firing the first propulsion modules. These modules are then staged (2).

Three midcourse corrections (3) are assumed on the outbound leg, the first occurring 5 days after launch from orbit, the second 20 days later, and the third approximately 20 days prior to arrival at Mars. Meteorite shielding and insulation for the second propulsion module is staged and the module fired (4) to insert the spacecraft into a high-altitude elliptical orbit about Mars. The spent propulsion module is separated (5) in this orbit and the spacecraft transfers (6) into an operational orbit using a trim propulsion system.

Two to five days are then spent for final landing site selection, orbital experimentation, including probe deployment, and preparing the Mars excursion module for operation. Three of the six man crew then descend (7) to the planet surface. After a 30-day stay on Mars, the ascent module of the Mars excursion module returns (8) the flight crew and scientific payload to the mission module. During planetary surface operations the flight crew remaining in the mission module continues orbital experimentation and operations and monitors surface operations. After rendezvous and transfer of flight crew and samples to the mission module, the ascent module is abandoned in the planetary orbit.

Preparations for planet departure include staging (9) of the orbit trim propulsion system and shielding and insulation for the third propulsion module. Departure from Mars orbit is accomplished by firing (10) the third module. The propulsion module is then staged (11).

Following midcourse corrections (12), the flight crew and scientific payload are transferred to the earth entry module and approximately 1 day prior to earth entry, separation from the mission module (13) is accomplished. The trajectory is then adjusted (14) for entry and landing at the desired location.

This is not to say that the relatively short-duration, high-energy Mars opposition mission should be undertaken at any particular time, or that six is the optimum flight crew complement; however, it does serve as a basis for estimating that portion

of the total man hours which are available for experimentation.

### Experiment Activity

The profile for the Mars opposition mission has been defined in terms of the variation of flight crew activity with mission elapsed time and is presented in Figure 2. It is immediately obvious that approximately three-fourths of the total man hours available are generally required for recreation, sleep, personal hygiene, and spacecraft operations, while the remaining one-fourth of the time is generally available for experimentation. During the 30 days allotted in this particular mission for experimentation in Mars orbit and on the surface when a mission module and a surface module are employed simultaneously, the total spacecraft operation time requirement is seen to double and the established level of experiment activity is maintained at the expense of recreation. Mission critical events, such as earth departure, planetary acquisition, planetary departure, and midcourse corrections, when a "general quarters" situation will prevail, are shown as gross perturbations in all onboard activities. Although all of the time during these events is shown as allotted for spacecraft operations, that portion of any biomedical/behavioral experiment program which utilizes the flight crew as the subject will obviously continue.

Other relatively high spacecraft operations time allotments occur during the periods of final landing site selection, and planetary departure preparation. Although data obtained at this time will be of an operational nature and utilized to make "real-time," onboard mission control decisions, it will be recorded for subsequent analysis as experimental data if desired.

The most significant point to be made here is that the time available for experimentation is relatively small and that the information obtained from experimentation can be increased only by increasing the flight crew complement or by maximizing the effectiveness of the experiment program. Although the remainder of this paper will be concerned with maximizing experiment program effectiveness and the attendant complexities, results of a recently completed study (NASA Contract NAS1-6774, The Boeing Co.) have shown that a moderate increase in flight crew above six is probably desirable because of the separation of personnel during the planetary orbit and surface phases of the mission.

### Outbound In-Transit Experimentation

The largest blocks of experiment activity were shown in Figure 2 to accrue to the in-transit phases of the mission. Further examination of an experiment program directed toward understanding the solar system indicates that the character of the in-transit activity outbound may be significantly different from that inbound.

Figure 3 shows a typical distribution of experimental activity outbound. The initial activity includes experiment setup and calibration with emphasis on observation of earth. After the second midcourse correction, increasing interest in the destination is noted and the associated experiment activity is seen to be responsive to this interest as the destination is approached. In particular,



observations will be made of the destination planet to establish its albedo, figure, period of rotation, axial tilt, and cloud cover and to investigate visible areas of interest on the planet surface. Also of concern, naturally, is the environment surrounding the spacecraft and the condition of the flight crew. For example, measurements would be made to determine the structure of the interplanetary magnetic field, the nature of interplanetary particulate matter, and the flux of emitted particles and photons from the sun. Some activity is directed toward selected observations of the solar system from a position relatively remote from planetary influences. This activity would include observations of the sun (for correlation with simultaneous observations from earth), Jupiter, and the asteroids such as Icarus. A relatively constant level of biological and radiation monitoring of test plants and animals is also noted.

#### Experimentation in the Vicinity of the Planet

The distribution of activity shown in Figure 4 corresponds to simultaneous experimentation on the surface of the planet and in planetary orbit.

The initial phase of the surface experiment program is primarily directed toward the selection of features for subsequent measurement, photography, mapping, and analysis. The actual scope of this particular effort will, of course, be dependent on the degree of surface mobility provided the experimenters. Later activity is directed more toward the selection of samples which can ultimately contribute detailed information about the physical characteristics of the planet and the modifying forces which may account for their contemporary status. All analyses of data and samples accomplished during this period will be of a preliminary nature directed toward assurance that the samples which are packaged and returned to the mission module for further analysis or ultimately for return to earth are those of most scientific significance. The surface program will also include the performance of various radiation measurement and atmospheric studies.

The experiment program which progresses simultaneously aboard the mission module is directed toward synoptic measurements which increase the understanding of the planet as a whole and toward observations which corroborate the results of the surface experimentation. An important part of this program will be the production of a detailed planning map which is not only essential to the establishment of the shape of the planet and the fixing of prominent features with respect to one another, but also to the proper analysis of data gathered both in orbit and on the surface. Radar and IR mapping would also be performed as well as UV, visible, and IR spectral measurements and IR and RF radiometric measurements. Biomedical and radiation monitoring is, of course, continuing.

#### Inbound In-Transit Experimentation

The inbound experiment program shown in Figure 5 includes some astronomical observations and environment measurements similar to those of the outbound program, but at a relatively low continuous level except near the end of the mission when observations of the earth and moon are begun. It is noted that in the inbound program, the emphasis is on the

selection and detailed qualitative and quantitative analysis of samples acquired from the Martian surface. This includes exposure of test plants and animals and culturing and analysis of life samples both of which are initiated at a relatively modest level. However, both are time dependent and are characterized by a varying level of activity as the mission progresses, and are continued as long as possible. The geophysical sample analysis, on the other hand, is characterized by a high level of initial effort which is reduced as the more promising samples are identified and analyzed or packaged. Later, the program becomes more operationally oriented, in that activity is directed toward increased consultation with earth concerning sample selection for earth return and transmission of experiment data.

#### Experiment Planning Process

A planning process for planetary orbit and surface experimentation is shown in Figure 6 and was selected to demonstrate a type of logic which should be applied in order to obtain an early definition of the experiment program and the associated advanced technology requirements.

The process begins with the basic definition of one of the general scientific objectives of planetary exploration, which may be summarized as that of gaining more knowledge about the origin and evolution of the solar system. This general objective may be further defined as shown - to gain knowledge about the origin and evolution of specific planets - in this instance, Mars.

The general objective comprises several scientific subobjectives, some of which are shown at the next level. Of these subobjectives, planetology, the study of the gross features of the planet, has been selected for further analysis. Some of the information requirements which contribute to this analysis are shown at the next level.

Mineralization serves to demonstrate a relatively broad information requirement which generates various measurement requirements, each of which may be investigated by several experiment techniques. In particular, mineralization may be derived from measurements such as those shown on the next level. The chemical composition measurements may be performed with a combination of imaging and spectral techniques. The imaging instrumentation is shown on the next level and consists of multispectral camera, radar imager, and the astronaut himself. The instrumentation then imposes detailed requirements for spacecraft and mission design. Typical of these are data management, power, and stability and control.

While it is recognized that the final approval of the experiment approach and instrumentation is the prerogative of the principal investigator, the experiment planning process does allow the spacecraft and mission designer to make rather detailed engineering decisions concerning the requirements imposed on the spacecraft and mission by the experiment program which can in turn result in an early definition of advanced technology requirements.

#### Advanced Mission Technology Requirements

The preceding analysis of the typical experiment program associated with a manned planetary mission

clearly demonstrates some of the complex requirements which must be considered in the planning of the planetary mission. It should be noted, however, that these are just the complexities which reflect a dynamic, onboard experiment program and serve to compound those associated with any long-duration, resupply-independent mission.

Advanced mission technology must now be directed to consider some approaches to satisfying these complex requirements within the constraint that little if any real-time support from earth can be expected during the mission because of the sheer distances and associated times involved.

One obvious solution is that the system be capable of autonomous operation which can be approached by optimizing the role of man and providing maximum systems operations flexibility (fig. 7).

#### Optimization of the Role of Man

The ability to conduct experiments and to correlate results from many observations will afford the opportunity for an onboard experimenter to adapt experimental procedures and immediately reorient measurement requirements based on an early evaluation of the most significant data. The capability to reevaluate the mission status and to reschedule onboard activity in near real-time in order to be responsive to unforeseen events may be provided through the onboard use of detailed mathematical models and their attendant computer programs.

This concept is presented in Figure 8 with resources, models, and a computer shown at man's disposal.

The resources comprise the nominal experiment and mission plans and associated space vehicle, supplies, and equipment. Some of the more obvious resources are flight crew number and skills, environmental control and life support system, power system, stabilization and control systems, crew systems, and reliability.

These models and computer programs must provide near real-time data for systems status monitoring to aid in onboard resources management and, as the mission progresses, to provide a finer range of event scheduling based on continual status updates from day-by-day or hour-by-hour resource changes. The models must have the simulation capability to "look ahead" in the mission and perform detailed analyses to optimize remaining resource utilization in the event of changing mission priorities, and to provide a dynamic rescheduling capability which is responsive to requirements imposed by a dynamic experiment program. However, because of the long mission and transmission delay times associated with planetary distances, the spacecraft cannot enjoy the luxury of real-time, ground-based computer support to aid the flight crew during the many planned mission modes or contingencies which may arise. Thus, the need arises for the development of onboard computers and ancillary systems capable of extending the flight crew capability in the management of spacecraft systems and the efficient implementation of the experiment program.

Mission simulation mathematical model technology presently being developed as an analytical, managerial tool (NASA Contract No. NAS1-7105, General Dynamics) to investigate the complex requirements

of several concepts of manned orbital research facilities is leading to the capability for such support.

Utilization of the model in deterministic planning and probabilistic simulation modes permits studies which include experiments analysis, flight crew skill and duty cycles, reliability analysis, schedules of onboard activities, vehicle maintenance, expendable requirements, environmental and stowage problems, subsystems trade-offs, operations analysis, evaluation of emergency provisions, and logistics support. It should be noted, however, that large computer complexes are currently required.

An example of results of a typical computer run from the simulation model is shown in Figure 9. A printout of a current systems status and a station efficiency check is shown, where systems description input data are extracted from several libraries as required. This check was performed following an unrepaired systems failure, simulated by Monte Carlo techniques, and consists of a current and predicted systems status analysis.

CURRENT SYSTEMS STATUS, given on this chart, includes a description of each type of component, its redundancy, number of components currently in the failed status, current spares level, and probabilities of failure, losing function, and losing alternate mode. These probabilities are combined into the PROBABILITY OF MISSION ABORT charged to every major system. This current systems-status check allows pinpointing of degrading components, and corresponding remedial action with respect to spares package, reliability, or redundancy. The system reliabilities are ultimately combined into the efficiency index.

A similar analysis is performed to determine systems status after a hypothetical special logistics launch, and the results of each analysis are combined into PROBABILITY OF CREW SURVIVAL and PROBABILITY OF MISSION SUCCESS, which are in turn combined with PERCENT EXPERIMENTAL RETURN into an EFFICIENCY INDEX. The incremental efficiency is computed, and prorated with respect to the number of days until the next regularly scheduled logistics launch, giving SPECIAL LAUNCH VALUE. This is compared to an input cutoff point which reflects a predefined mission rule. If it is greater, a special logistics launch is requested to resupply spares. For this particular run, the special launch value was less than the cutoff point, so a special launch was not requested.

Although emphasis here is on the value of a special launch to assure flight crew safety and mission success, such an option would not be available to the flight commander during a planetary mission. In addition, such extensive data must be displayed in a simplified format. The experience gained in building such large simulation models will prove invaluable in the eventual onboard utilization of these methods. As computer technology advances, so will the techniques which will allow easier manipulation of the input-output data. This will include advanced visual display techniques and printed formats along with simplified model routines and software which will alleviate the storage problems associated with large, complex programs.

## Spacecraft Systems Operations Flexibility

The second broad area of advanced mission technology reflects the requirement of the experiment program for a spacecraft which can operate effectively in several modes - manned, automated, modular, and in integrated combinations of all three. Consideration of the mission and attendant experiment program evolution from earth departure to earth return demonstrates the dynamic functional requirements imposed on system design as is shown in Figure 10.

During the outbound portion of the mission, the primary mode of experimentation is that peculiar to the observatory. The nature of these observations, with repetitive measurements and extreme stabilization and control requirements, suggests automation and modularization for part-time operation at a point relatively remote from the influence of the mission module and the presence of man. The capability for this mode of operation is particularly important if it is established that artificial gravity with the attendant motion and complexity is a requirement for flight crew survival or habitation. In addition, since this portion of the experiment program may be relatively static, the role of man tends initially toward one of technical support to the instruments. As the mission progresses, however, the scope of man's involvement increases as he becomes more concerned with observations which contribute directly to landing site selection.

The experiment programs to be performed in planetary orbit and on the planetary surface must be complementary. The surface experimenter will be concerned with selection of specific experimentally significant samples and obtaining measurements in a relatively small area. The program being conducted simultaneously aboard the orbiting mission module is directed toward the collection of synoptic data and is enhanced by automated satellites and probes under control of the mission module. Man's primary role here is twofold - on the surface - sample selection; in orbit - data handling.

During the early inbound portion of the mission, the primary task of man is that of qualitative and quantitative analysis in the biological and geophysical sciences. The mission module must accommodate separate laboratory facilities for bioscience and geophysics as well as supporting facilities for common usage such as optics, electronics, and chemistry. In addition, the mission module must provide the facilities for data selection, storage, and transmission. Thus, the requirement for an integrated manned experimental research laboratory in space is established.

The overall requirement is for laboratory and spacecraft systems that can be operated in several modes depending on the nature of the experiment, the optimum degree of man's participation at the particular time, and the mode of mission module operation.

Concepts of advanced optical equipment, for example, the 120-inch manned orbital telescope, have been studied with considerable emphasis on operating in conjunction with a manned orbital research laboratory (Contract No. NAS 1-3968, The Boeing Co.). Eight modes of operation were evaluated and some of the results are presented in Figure 11.

The best data quality, which in this effort reflected technical risk, is seen to be obtained when the optical system is completely separated from the laboratory system; however, man's ability and safety, which are measures of data quantity, are somewhat degraded because of the necessity for extravehicular activity or operation of a shuttle system. On the other hand, when man's ability to conduct these experiments is best, the data quality is relatively poor and the laboratory interface requirements which in turn impact other onboard experimentation requirements are severe.

Further study is obviously needed in this and related onboard operative areas. Experiment modules should be capable of more than one operational mode and the experiment program should be planned to exploit each mode to provide adequate data of the quality desired in every phase of experimentation.

## R & D Programs

A broad program of research and development which is responsive to the advanced mission technology requirements imposed by a manned planetary mission and an experiment program derived from the scientific objectives is presented in Figure 12.

It begins with the basic assessment of man's capability as an experimenter in space utilizing earth orbital programs of increasing complexity and duration. Man's requirements and limitations must be determined and supplemented when necessary.

The space rating of man will require detailed clinical studies on man and basic studies on man and animals at the cellular level. Of particular interest will be man's performance in earth orbital programs such as earth resources and astronomy where he will function with automated or semi-automated equipment to make repetitive measurements as well as have the opportunity to redirect the experiment program as unforeseen phenomena occur.

Data acquisition techniques and equipment will require extensive investigation. Sensor technology must be extended and include development of common components where possible, and data handling techniques which permit preliminary onboard analyses must be defined in order to assure return of the most significant planetary information to earth.

Engineering design data on the planetary environment and surface must be obtained from precursor, automated probes. Analyses of the long-range requirements of the flight crew and experiment program on environmental control and life support, power, stabilization and control, and crew systems must be completed, and highly reliable systems which can meet the requirements must be developed.

Applied programs derived from basic technology will be directed toward the optimization of the role of man and will include simulation of most or all of the planetary mission in earth orbit in order to demonstrate complete systems reliability. Automated systems under remote control of man in orbit must be used to develop techniques for the effective utilization of the telescopes and planetary orbiters and probes which will ultimately provide synoptic coverage of the planet. Procedures for the handling of samples in a sterile or closely controlled environment must be established.

The feasibility of a versatile experimental facility for research in space has been defined in extensive detail in several previous studies. (For example, see NASA Contract No. NAS1-3612, The Douglas Aircraft Co.) In each instance, these concepts were operational in earth orbit and were dependent on logistics support and near real-time communication with earth. The planetary mission module must provide the versatility of these earth orbital systems without support from earth.

#### Concluding Remarks

Possession of the capability for manned planetary exploration depends on possession of three complex, interdependent elements; a scientifically significant experiment program, a space vehicle system, and a flight crew. Examples of the impact of a broad experiment program on spacecraft design and operation and on the role of man have been shown to demonstrate the significance of experiment planning on advanced mission technology. It is seen

that a comprehensive experiment program imposes the requirement for a highly versatile research facility capable of autonomous space operation. It may be described as integrated, manned, automated, and modular and it must allow man to utilize his unique abilities to the fullest extent.

In addition, consideration must be given to assuring that man's best capabilities are represented in the flight crew. The system of astronaut selection and training must produce the necessary scientists possessing skills at the technician level in disciplines other than their own and proficiency in onboard flight operational assignments.

Before a manned planetary mission is attempted, the related complexities of experiment program, space vehicle, and flight crew must be understood. The challenge today, then, lies in the definition of the scientific objectives and the development of the experiment program.

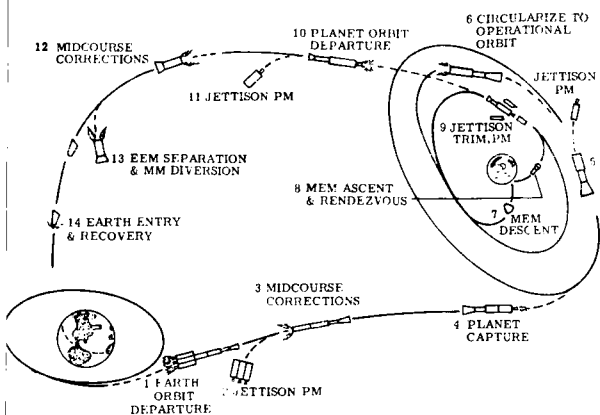


Figure 1.- Mission events sequence.

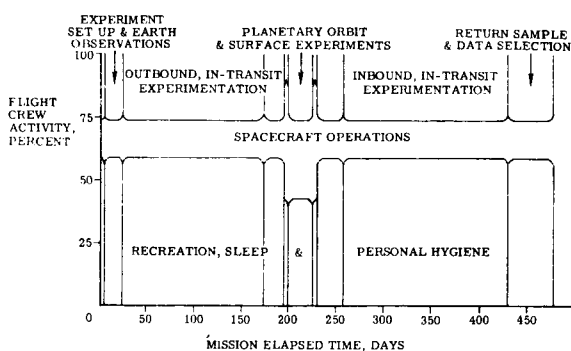


Figure 2.- Variation of flight crew activity with mission elapsed time.

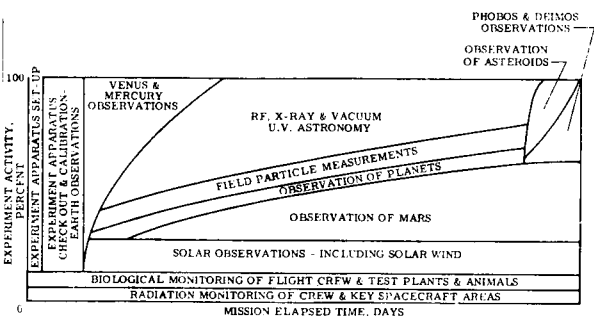


Figure 3.- Outbound in-transit experiment program.

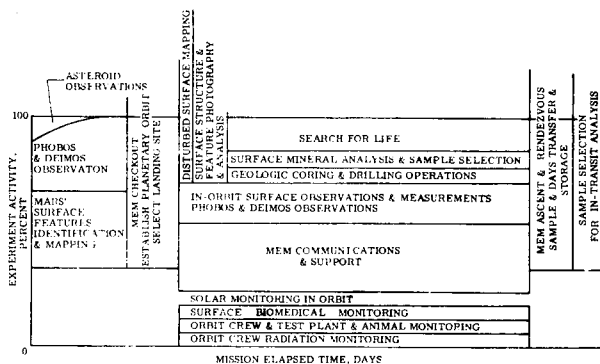


Figure 4.- Planetary surface/orbit experiment program.

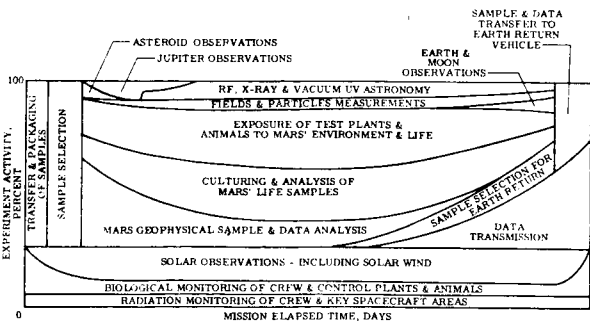


Figure 5.- Inbound in-transit experiment program.

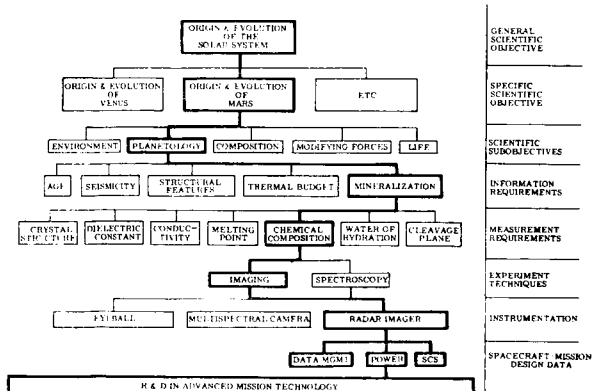


Figure 6.- Experiment planning process.

- **SYSTEMS OPERATIONS FLEXIBILITY**

| CURRENT SYSTEMS STATUS                                                                                                                       |                                         |                                     |                                   |                     | PROBABILITY<br>OF LOSING<br>ALTERNATE<br>MODE |
|----------------------------------------------------------------------------------------------------------------------------------------------|-----------------------------------------|-------------------------------------|-----------------------------------|---------------------|-----------------------------------------------|
| COMP<br>TYPE                                                                                                                                 | NUMBER OF<br>COMPONENTS<br>OF THIS TYPE | NUMBER OF<br>COMPONENTS<br>NEED (N) | NUMBER OF<br>FAILED<br>COMPONENTS | NUMBER OF<br>SPARES |                                               |
|                                                                                                                                              |                                         |                                     | 0                                 | 1                   | 0.00120121                                    |
| 1                                                                                                                                            | 2                                       | 1                                   | 0                                 | 1                   | 0.00218105                                    |
| 3                                                                                                                                            | 1                                       | 1                                   | 0                                 | 0                   | 0.00000475                                    |
| 3                                                                                                                                            | 1                                       | 1                                   | 0                                 | 0                   | 0.00012063                                    |
| 3                                                                                                                                            | 1                                       | 1                                   | 0                                 | 0                   | 0.00218105                                    |
| 5                                                                                                                                            | 3                                       | 1                                   | 0                                 | 0                   | 0.00218105                                    |
| 5                                                                                                                                            | 3                                       | 1                                   | 0                                 | 0                   | 0.00170052                                    |
| 5                                                                                                                                            | 3                                       | 1                                   | 0                                 | 0                   | 0.00364298                                    |
| 6                                                                                                                                            | 8                                       | 1                                   | 0                                 | 0                   | 0.00225991                                    |
| .                                                                                                                                            |                                         |                                     |                                   |                     | .                                             |
| .                                                                                                                                            |                                         |                                     |                                   |                     | .                                             |
| .                                                                                                                                            |                                         |                                     |                                   |                     | .                                             |
| 51                                                                                                                                           | 2                                       | 1                                   | 0                                 | 0                   | 0.00002730                                    |
|                                                                                                                                              |                                         |                                     |                                   |                     | 0.000000001                                   |
|                                                                                                                                              |                                         |                                     |                                   |                     | 0.0                                           |
| PROBABILITY OF MISSION ABORT BECAUSE OF SYSTEM 1 IS 0.026474                                                                                 |                                         |                                     |                                   |                     |                                               |
| .                                                                                                                                            |                                         |                                     |                                   |                     | .                                             |
| .                                                                                                                                            |                                         |                                     |                                   |                     | .                                             |
| .                                                                                                                                            |                                         |                                     |                                   |                     | .                                             |
| PROBABILITY OF MISSION ABORT BECAUSE OF SYSTEM 1 IS 0.000069                                                                                 |                                         |                                     |                                   |                     |                                               |
| A STATION EFFICIENCY CHECK HAS BEEN PERFORMED. CURRENT STATUS AND PREDICTED STATUS (AFTER A SPECIAL LAUNCH) ARE GIVEN IN THE FOLLOWING TABLE |                                         |                                     |                                   |                     |                                               |
|                                                                                                                                              |                                         |                                     | PRESENT                           | PREDICTED           |                                               |
| PROBABILITY OF CREW SURVIVAL                                                                                                                 |                                         |                                     | 0.969                             | 0.966               |                                               |
| PROBABILITY OF MISSION SUCCESS                                                                                                               |                                         |                                     | 0.867                             | 0.933               |                                               |
| PERCENT EXPERIMENTAL RETURN                                                                                                                  |                                         |                                     | 61.6                              | 81.9                |                                               |
| EFFICIENCY INDEX                                                                                                                             |                                         |                                     | 0.820                             | 0.906               |                                               |
| SPECIAL LAUNCH VALUE                                                                                                                         |                                         |                                     | = 4.96                            |                     |                                               |

| PHE-<br>DOMINANT<br>EXPER.PROG.<br>CHARACTERISTICS | MISSION<br>PHASE | IN-TRANSIT<br>OUTBOUND | PLANETARY STAY                 |                                  | IN-TRANSIT<br>INBOUND        |
|----------------------------------------------------|------------------|------------------------|--------------------------------|----------------------------------|------------------------------|
|                                                    |                  |                        | ORBIT                          | SURFACE                          |                              |
| EXPERIMENT<br>ACTIVITY                             |                  | OBSERVATION            | SYNOPTIC<br>DATA<br>COLLECTION | SPECIFIC<br>SAMPLE<br>COLLECTION | SAMPLE &<br>DATA<br>ANALYSIS |
| MODE OF<br>OPERATION                               |                  | AUTOMATED              | AUTOMATED                      | MANNED                           | MANNED                       |
| ROLE OF<br>MAN                                     |                  | SUPPORT                | DATA<br>HANDLING               | SAMPLE<br>COLLECTION             | RESEARCH                     |

| OPERATING<br>MODE<br><br>EVALUATION<br>FACTOR               | RIGID MOUNT  | TETHERED | SEPARATE                |                    |
|-------------------------------------------------------------|--------------|----------|-------------------------|--------------------|
|                                                             |              |          | INTERMITTENT<br>DOCKING | NO<br>DOCKING      |
| RELATIVE<br>DATA<br>QUALITY                                 | VERY<br>POOR | POOR     | FAIR                    | MODERATELY<br>GOOD |
| SPACE STATION<br>INTERFACE<br>REQUIREMENTS &<br>CONSTRAINTS | VERY SEVERE  | SEVERE   | MODERATE                | FEW                |
| MAN'S ABILITY<br>TO PERFORM<br>TASKS                        | VERY GOOD    | GOOD     | GOOD                    | POOR               |
| SAFETY                                                      | 0.9999       | 0.9900   | 0.9818                  | 0.9602             |
| RELIABILITY                                                 | 0.96         | 0.94     | 0.95                    | 0.97               |

```
graph TD; A["BASIC TECHNOLOGY<br/>• MAN'S CAPABILITY<br/>• DATA ACQUISITION & HANDLING<br/>• SYSTEMS DESIGN"] --> B["APPLIED PROGRAMS<br/>• OPTIMIZATION OF ROLE OF MAN<br/>• SYSTEMS FLEXIBILITY"]; B --> C["ADVANCED MISSION TECHNOLOGY<br/>• EFFECTIVE UTILIZATION OF MANNED/AUTOMATED SYSTEMS TO PROVIDE AUTONOMOUS EXPERIMENT AND MISSION CONTROL"]; A --> C;
```

The diagram is a flowchart illustrating the progression of technology in mission systems. It consists of three main rectangular boxes connected by arrows. The top box, labeled 'BASIC TECHNOLOGY', contains a bulleted list: 'MAN'S CAPABILITY', 'DATA ACQUISITION & HANDLING', and 'SYSTEMS DESIGN'. An arrow points from this box to a middle box labeled 'APPLIED PROGRAMS', which contains a bulleted list: 'OPTIMIZATION OF ROLE OF MAN' and 'SYSTEMS FLEXIBILITY'. Another arrow points from the 'APPLIED PROGRAMS' box to the bottom box, labeled 'ADVANCED MISSION TECHNOLOGY', which contains a bulleted list: 'EFFECTIVE UTILIZATION OF MANNED/AUTOMATED SYSTEMS TO PROVIDE AUTONOMOUS EXPERIMENT AND MISSION CONTROL'. Additionally, a direct arrow points from the 'BASIC TECHNOLOGY' box to the 'ADVANCED MISSION TECHNOLOGY' box, indicating a foundational relationship.

8

## CURRENT SHUTTLE STATUS

By C. J. Donlan, Acting Director, Space Shuttle Program, OMSF  
NASA Headquarters, Washington, D.C.

Space Shuttle Program objectives are to provide future capabilities to support a wide range of scientific, defense, and commercial uses at substantially lower costs than current space operations.

## SPACE SHUTTLE COST TRENDS

(Figure 1)

The dotted band in figure 1 defines the locus of costs for current transportation systems ranging in size from the Scout to the Saturn V. For a particular payload capability, the points on the curve show the transportation cost per pound for the lowest cost transportation system available for that payload.

The line identified as Fixed Cost/Flight is that projected for the Space Shuttle. For the maximum payload envisioned, approximately 65 000 pounds ( $\approx 29\,500$  kg), it is seen that a reduction in cost of a factor of 10 is possible. Of equal importance is that the payload weight carried by the shuttle can also be reduced by a factor of 10 and still allow the shuttle to place the lower size payload in orbit cheaper than any other available system. It is this flexibility in operation that makes the shuttle an attractive transportation system.

Study results also support the thesis that an equally important reduction in manufacturing cost-per-pound of payload can be achieved by designing the payload to make use of features supplied by the shuttle, such as a more benign environment at launch and the ability to maintain or retrieve the payload in orbit. Lower transportation costs and lower cost of the payloads transported are both essential to a continued effective space program.



# SPACE SHUTTLE COST TRENDS

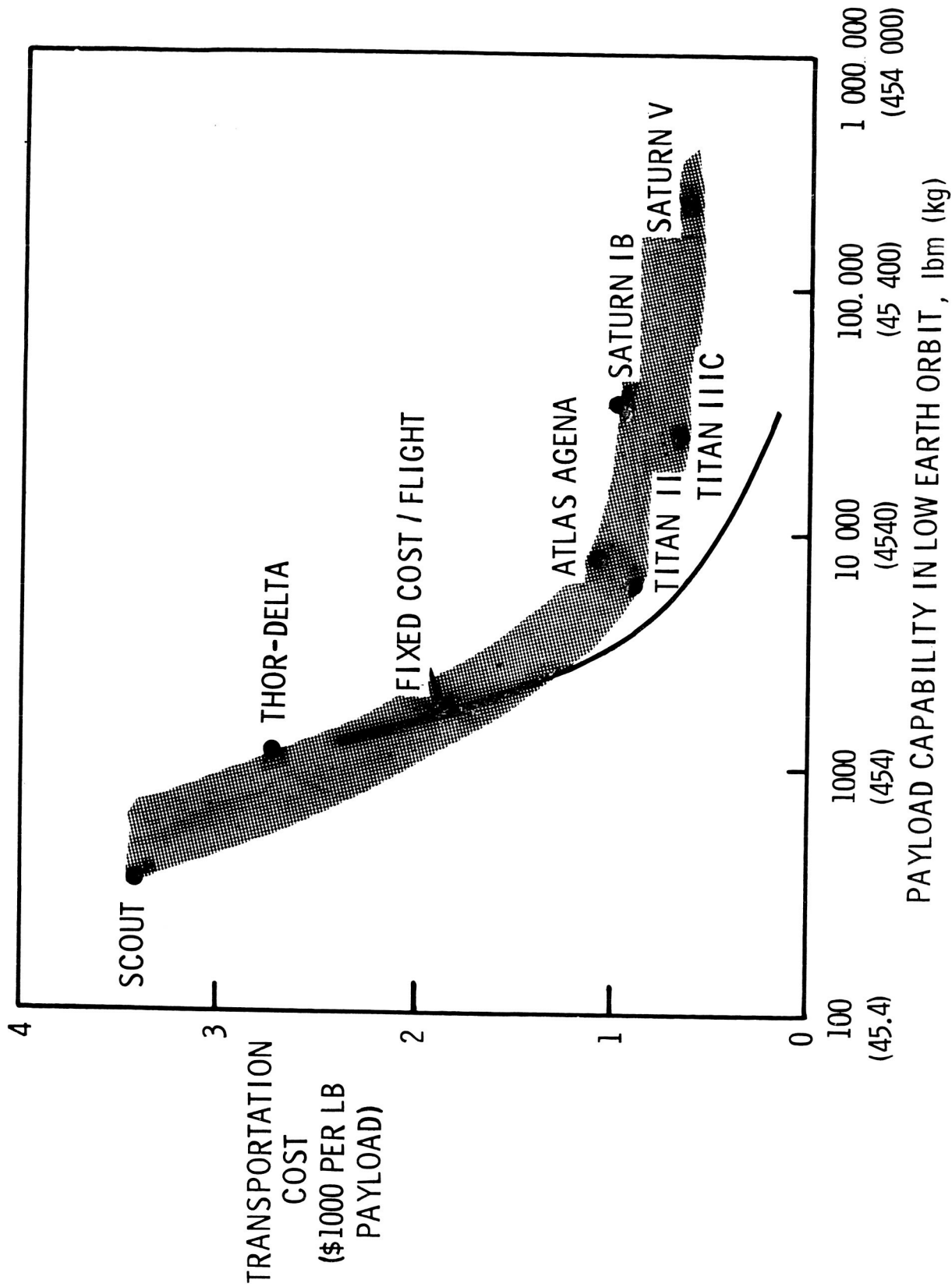


Figure 1

## TECHNOLOGY CONTRIBUTIONS

(Figure 2)

For the past 12 to 18 months, the technology has been primarily plans oriented and directed toward problem assessment. There has been much progress made relative to mobilization of personnel and resources within NASA and in industry to address the technology problems of shuttle. These programs are now well under way and are proceeding in an effective manner.

As of this time, in support of the Phase B effort, much progress has been made in the area of assessing performance requirements, engine size, and vehicle configuration. Such information has provided a basis for narrowing the range of the Phase B configuration studies and settling on more precise performance specifications.

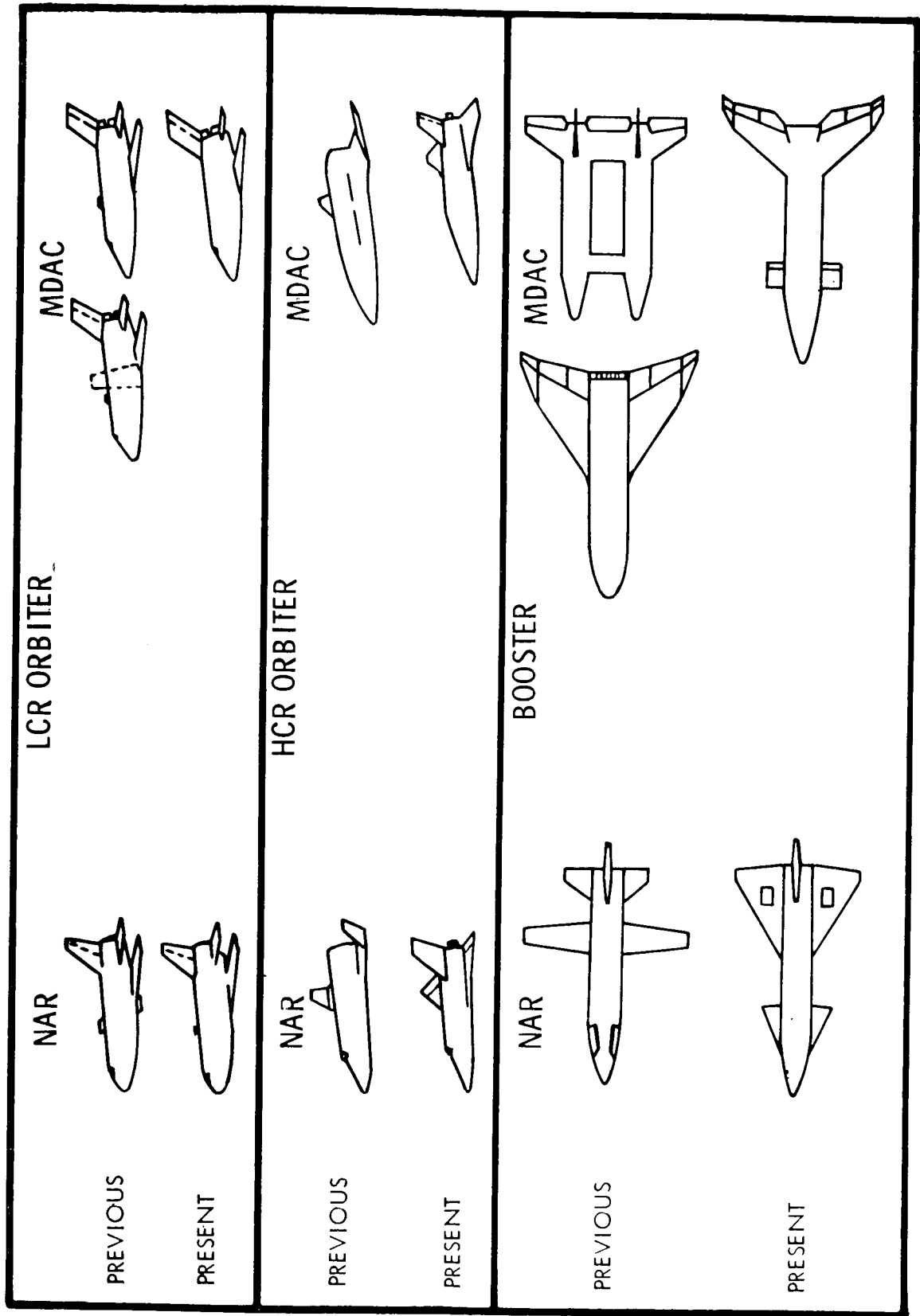
# **TECHNOLOGY CONTRIBUTIONS**

- **PROBLEM ASSESSMENT**
- **MOBILIZATION OF PEOPLE AND RESOURCES**
- **BROAD TECHNOLOGY PROGRAM PLANNED AND UNDERTAKEN**
- **DIRECT SUPPORT OF PHASE B IN SETTING**
  - **PERFORMANCE REQUIREMENTS**
  - **ENGINE SIZE**
  - **VEHICLE CONFIGURATION**

(Figure 3)

This figure shows the steps in the evolution of the low cross-range orbiter and the high cross-range orbiter and the associated boosters during the course of the Phase B studies. The straight-wing designs of both the orbiter and the booster have been dropped from the program because of deficiencies determined from aerothermal analyses and wind-tunnel studies. The high cross-range orbiter has evolved into a delta-type configuration with a single vertical tail. Both boosters have evolved into a canard-type configuration for similar reasons.

# SPACE SHUTTLE CONFIGURATION EVOLUTION



## SPACE SHUTTLE PERFORMANCE SPECIFICATIONS

(Figure 4)

This figure is a summary of the current Level I requirements for the reusable shuttle.

The cargo bay is baselined as 15 feet (4.6 m) in diameter by 60 feet (18.3 m) in length. The 60-foot dimension is perhaps the firmest dimension inasmuch as it provides space for rockets such as the Agena and Centaur stages.

It is anticipated that most missions will operate with the airbreathers in; however, as experience with the shuttle is acquired, it is feasible to consider operation with the airbreathers out, in which case, another 20 000 pounds (9075 kg) of payload can be carried in the east or south circular orbits. The design of the structural landing load is based upon a return payload weight of 40 000 pounds (18 150 kg).

The 1100 nautical mile cross range allows a return to the initial launch site in a single revolution in the event of an abort to orbit.

The life support system and fuel requirements for the shuttle while in orbit are based on 7 days of self-sustaining orbital flight. The systems will be qualified for 30 days in orbit, but the expendables for orbital flight in the excess of 7 days will be charged against payloads.

## SPACE SHUTTLE PERFORMANCE SPECIFICATIONS

- TWO-STAGE, FULLY REUSABLE SHUTTLE
- CARGO BAY - 15 ft DIAMETER × 60 ft LENGTH (4.6 m × 18.3 m)

| MISSION                              | PAYLOAD                       |                              |
|--------------------------------------|-------------------------------|------------------------------|
|                                      | AIRBREATHERS<br>OUT, lbm (kg) | AIRBREATHERS<br>IN, lbm (kg) |
| 100 n.mi. DUE EAST CIRCULAR ORBIT    | 65 000 (29 484)               | ≈45 000 (20 412)             |
| 100 n.mi. SOUTH POLAR CIRCULAR ORBIT | 40 000 (18 144)               | ≈20 000 (9072)               |
| 270 n.mi. AT 55° INCLINATION         | ≈45 000 (20 412)              | 25 000 (11 340)              |
| DOWN PAYLOAD                         | 40 000 (18 144)               | 25 000 (11 340)              |

- 1100 n.mi. ORBITER NOMINAL CROSSRANGE
- MISSION DURATION - 7 DAYS SELF SUSTAINING FROM LIFT-OFF TO LANDING
  - 30 DAYS EXPENDABLES CHARGED AGAINST PAYLOADS
- MAIN ENGINE BASELINED AT 550 000 lbf (2 446 400 N) SEA LEVEL THRUST (BOOSTER AND ORBITER)
- AIR BREATHING ENGINES USE JET FUEL
- ACCELERATION NOT TO EXCEED 3 g

Figure 4

## SPACE SHUTTLE CURRENT STATUS

(Figure 5)

Level I Program requirements have been summarized on the preceding figure.

The President's budget contains \$100 million for the space shuttle with a positive recommendation to proceed with the development of the engine for which the RFP\* was released on March 1, 1971. We are indicating an August 1971 release for the vehicle RFP should the current Phase B studies and complementary Phase A activity provide the confidence to move out into Phase C.

The schedule assumes also that the necessary approvals to proceed in this manner will be obtained in the same time period.

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\*RFP Request for proposal.



# **SPACE SHUTTLE CURRENT STATUS**

- LEVEL I PROGRAM REQUIREMENTS REDEFINED
- PHASE B STUDIES NOW IN 8TH MONTH, WILL BE COMPLETED IN MAY (6 WEEK EXTENSION FOR MAIN ENGINE STUDIES)
- PRESIDENT'S FY 1972 BUDGET CONTAINS \$100M FOR SPACE SHUTTLE AN INCREASE OF \$20M OVER FY 1971
- PROGRAM PLANNING:
  - MAIN ENGINES - RFP - MARCH 1971  
START PHASE C - AUGUST 1971
  - VEHICLE - RFP - AUGUST 1971  
START PHASE C - MARCH 1972

Figure 5

## SPACE SHUTTLE PLANNING SCHEDULE

(Figure 6)

Current planning anticipates first horizontal flight tests of an orbiter in mid-1976 with the first manned orbital flight of the shuttle system planned in the second quarter of calendar year 1978.

Because shuttle development is planned to start in 12 to 16 months, the technology program must be harnessed to provide definitive answers to the design and development teams in the next year and a half.

# SPACE SHUTTLE PLANNING SCHEDULE

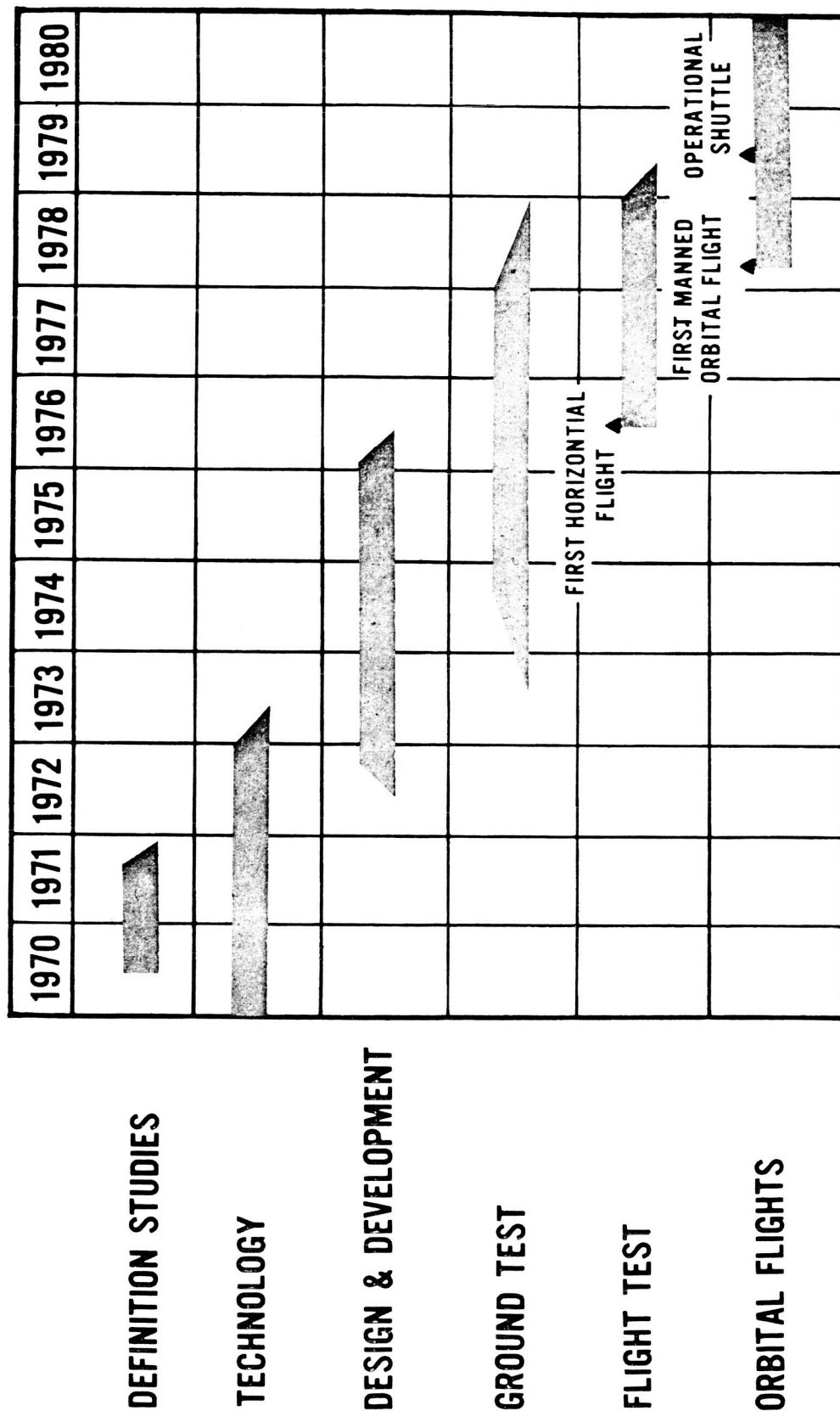


Figure 6

## TECHNOLOGY CHALLENGE

(Figure 7)

The greatest technology challenge from the point of view of the Shuttle Program Office is to generate the design data relative to advanced materials properties, aerodynamic heating, and flight loads in enough depth to allow Phase C activities to proceed with confidence and to provide hard data to support or challenge solutions relative to the thermal protection and advanced structural concepts being proposed.

The technology teams and researchers need to assure that their programs are relevant and that the results of the programs are made highly visible for assimilation into the shuttle design and development process.

# **TECHNOLOGY CHALLENGE**

- **GENERATE DESIGN DATA RELATIVE TO ADVANCED MATERIALS  
PROPERTIES, AERODYNAMIC HEATING AND FLIGHT LOADS**
- **DEMONSTRATE APPLICATION OF THERMAL PROTECTION  
AND ADVANCED STRUCTURAL CONCEPTS**

Figure 7

ENGINEERING THE FUTURE

BY

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ADA, OHIO

FEBRUARY 26, 1972

## ENGINEERING THE FUTURE

President Meyer, members of the Board of Trustees and Faculty, distinguished guests, ladies and gentlemen--it is an honor and a privilege for me to participate in the dedication of the College of Engineering Building. It is particularly encouraging to see this building coming into use at a time when technology once again has become a fashionable whipping boy. Indeed, the university itself has come in for intense criticism from all elements of our society--from the old and the young, the rich and the poor, the conservative and the radical, and from ethnic and minority groups. For each of these, the university is singled out as a special symbol of its concerns and disappointments. The older and conservative elements regard the university as a subversive force and see in its administrators too much tolerance of student and faculty challenges. Many young people and adults who are impatient for immediate social reforms are critical because they see the university as a conservative force that attempts to mold students to a role in society that they no longer regard as satisfying. The poor and certain minority groups see the university as a closed gateway to opportunity. President Jerome Wiesner of MIT put it this way, "We have achieved the dubious distinction of being regarded at one and the same time as the hothouse of revolution and propagator of the status quo."

Nevertheless, there are large numbers who are genuinely concerned and who regard uncontrolled technology especially as a major source of our social dislocations and environmental concerns. They believe that the social and economic forces that drive technological advancements cannot be directed in a way that would benefit the general welfare. They feel that technology will dominate our lives like a plague and somehow relegate men to be the unwilling servants of a large powerful, impersonal organization. They say this must be stopped.

What can be said of this assessment? There is no doubt that science and technology have helped create our present predicament by providing goods, services, and activities in modes of life and work and leisure which in the past were beyond the expectations of all but a small privileged class of our society. This has led to much social discontent and frustration. We have moved ahead in all directions with little regard for the conditioning of individuals and society to the consequences of this tremendous capability for change. We had accepted as an axiom the proposition that any technology that offered the promise of expanding our mastery over nature was desirable and essential. It has been relatively recent that this axiom has been questioned with the consequence that the relationship between technological change and man's social, biological and physical environments is now being considered. The establishment of the Environmental



Protection Agency at the federal level and similar offices at the state and local levels has done much to create a public conscience in these areas and to provide a mechanism for assessment. The positive steps being taken by business and industry to cope with these problems are even more encouraging. Gaps in our educational spectrum are also being filled with relevant information on the impact of our technological capabilities on our lives. Nevertheless, while science and technology have indeed helped create our present predicament, there is little question that effective solutions to these problems can only emerge from the same sources. Without new scientific knowledge and wise technological investments now and in the future, the problems of mankind will only increase. We must do more than merely identify and catalog the problems of our society; we must understand them and provide carefully thoughtout solutions. We must, in fact, engineer the future.

That is why I believe the study of engineering to be one of the most rewarding preparations for life that a student can undertake, even if his ultimate vocation may be in a field far removed from engineering. The study of engineering conditions a person to the nature of the technologically oriented world we live in by providing him with a comprehensive understanding of its mechanisms and by furnishing the foundation of knowledge required to shape and control its activity.

The source for this conditioning and knowledge rests in the rigor of the engineering curriculum itself. In particular, I submit there are four favorable influences at work on the engineering student during this phase of his education that will dominate his thought process for life. The first of these is a development of a respect for facts. The solution to engineering problems involves the examination of facts in a qualitative and quantitative sense. The necessity of assessing the qualitative and quantitative aspects of facts results in a continual sifting of the important and less important factors in the decision-making process. This may seem trivial but such training over a period of time does influence one's point of view. This characteristic of the engineer--an insistence on quantifiable facts--will be identifiable in his dealings with people all his life--perhaps to a fault at times. Engineers have been known to be quite boorish in such matters.

A second influence results from the involvement of the student to systematic approaches to problems or undertakings. The term "systems engineering" has become identified with such an orderly and comprehensive approach to the solution of engineering problems. The characteristic features of this approach are found in undertakings

ranging from the design of singular items such as machine tools to entire programs such as Project Apollo. Learning to think and plan in the manner implied by the systems engineering approach develops a lifelong skepticism toward accepting piecemeal or overnight solutions to complex problems.

A third influence involves the fostering of self reliance. The student of engineering finds himself exposed to many aspects of engineering activity. He develops an appreciation for details--such as the need for a stress analysis of all elements of a bridge structure or an airplane. He develops an awareness of the importance of these unglamorous but extremely vital activities to the success of the undertaking--and the realization that the completeness of such analyses are frequently dependent solely on the conscientiousness and competence of those individuals charged with carrying out that responsibility.

The fourth influence is what I call realistic communication. In engineering, probably as much as in any professional field in the university, discussions in the classroom and dialogue between faculty and student are similar to that which takes place in engineering activity in industry. The transition, therefore, from the engineering campus to the engineering industry is not like stepping into a whole new world. One

reason for this is that the engineering faculty is usually involved in some way with the active engineering profession in the role of consultants. Furthermore, their performance is readily measured by the success or failure of proposed solutions or the adoption or rejection of their ideas. Their experiences are inevitably communicated to students in the classrooms and seminars. This exposure to real life activity is most significant in formulating attitudes. It is probably not by accident that the colleges of engineering, students and faculty alike, have been amongst the most stable units in the university. I believe the four influences I have discussed have had a great deal to do with this fact.

Looking to the future, the engineering profession, I believe, will find itself more and more involved in determining its role in society. It should exercise more initiative in assessing what is good or bad about its activities. It should not let the important function of technology assessment go by default to those who see no good in technology. The engineering profession and the schools of engineering can help by studying the priority issues of our times and channeling engineering talent into new opportunities to serve society. Let me supply an example from the Space Program. President Nixon, in his recent state-of-the-union

message, indicated that the engineering and scientific knowhow that went into putting a man on the moon is now being channeled into priorities more aligned with current problems on earth.

One result of this realignment is that the capabilities we have developed for exploiting space will be heavily focused on earth resources. NASA will soon launch an Earth Resources Technology Satellite (ERTS) which will begin to furnish information required to balance man's needs against the earth's ability to meet them. Among data to be obtained are information related to agricultural crop species, crop health, types of rocks and soil, erosion, moisture content of the ground, coastal processes, surface water distribution, and water pollution. Next year, we will conduct more advanced earth resources experiments in the Skylab Program. Skylab, a modified upper stage of the Saturn V launch vehicle, will serve as the Nation's first manned earth-orbiting space station. One of the principal tasks of the Skylab Program will be earth observations and surveys.

Astronauts in Skylab will contribute in this area by selecting and calibrating sensors, interpreting measurements rather than sending raw data to earth, screening out irrelevant data, and detecting unforeseen events. An astronaut's capability to observe a large scene and to discern unusual features cannot be duplicated in current automated systems.

The Skylab Program will utilize the judgment and flexibility of men to get the best results from experiments too complex for the automated satellite. I believe this earth resources program, utilizing manned and unmanned spacecraft, will open an era of new benefits from space technology as limitless as the frontiers of space itself. The earth's survey from space may have unanticipated potential for benefitting mankind as significant as those already realized from the weather and communication satellites.

It is faith in this future for space that led President Nixon to announce on January 5 his approval of the Space Shuttle--an entirely new type of space transportation system. The Space Shuttle which features an airplane-like spacecraft will give us routine access to space at sharply reduced costs in dollars and preparation time. It will also result in greatly reduced costs of the payloads themselves. The Space Shuttle will make it possible, in the future, to launch routinely communications and weather satellites, and other advanced earth resource satellites that will allow us to monitor, and help us husband, our natural resources. Perhaps with such routine space operations made possible by the shuttle, one can hope for the development of an environmental monitoring system, international in scope, to help control our environment here on earth for all mankind.

I have used these examples of space technology because they are typical of how science and technology can be harnessed to help solve our problems and because they encompass my own sphere of knowledge. Similar extrapolations in other fields of engineering or technology can no doubt be made. I am convinced that engineering and technology not only have a bright future but that our own future, to a considerable extent, will be greatly influenced by them. It is not even necessary to redefine our terms of reference for I know of no better statement of the goal for engineering than the definition of engineering itself as contained in Webster's unabridged dictionary: "engineering--the science by which the properties of matter and sources of energy in nature are made useful to man in structures, machines, and products."